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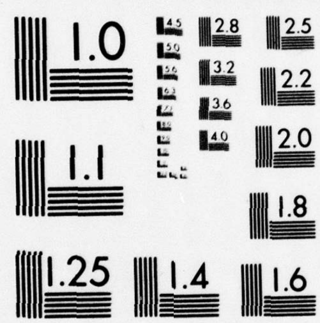
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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARDograph No. 234

Active Controls in Aircraft Design

Edited by

P.R. Kurzhals

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY
ON BACK COVER

14 AGARD-AG-234

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARDograph No.234

6 ACTIVE CONTROLS IN AIRCRAFT DESIGN,

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10 P. R. /Kurzhals

11 Nov 78

12 180 p.

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This AGARDograph was prepared at the request of the Guidance and Control Panel.

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Published November 1978

ISBN 92-835-0225-6

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

CONTENTS

EXECUTIVE SUMMARY

Active controls offer the promise of significantly increased aircraft performance and operational capability. However, realization of these gains will require major changes in both the aircraft design approach and in the implementation of the flight control system. This AGARDograph addresses related control-configured vehicle design and system considerations and summarizes representative applications of active control for fighter and transport aircraft.

Design Considerations

The basic active-control technology (ACT) or control-configured-vehicle (CCV) concept (CHAPTER 1) aims at optimizing aircraft geometry for each flight condition by considering flight control in the preliminary design process. This approach extends the traditional tradeoffs between aerodynamics, structures and propulsion to include the capabilities of a full-time, full-authority fly-by-wire control system. For example, aerodynamic stability could be reduced to neutral or negative and overall aircraft performance improved by relying on the control system to provide artificial stability. Similarly, active redistribution of dynamic wing loads can significantly reduce wing root bending moment and permit reductions in wing structural weight. Clearly, these and other active control tasks involve an unprecedented understanding of the anticipated external disturbances, aerodynamic characteristics, control system responses and compatibility problems, requiring considerable advances in our ability to describe and model such phenomena.

Nevertheless, both individual and combined active control concepts are now being demonstrated on experimental, commercial and military aircraft (CHAPTER 2). Relaxed longitudinal stability has been implemented on the F-16, tested on an F104CCV aircraft, and evaluated for transport application. Control of the aircraft center of gravity through wing fuel management is in operational service on the Concorde, and ride improvement systems have been certified for the Boeing B-747 and incorporated in the Rockwell B-1. Directional stability augmentation has seen extensive transport application and the yaw damper currently flying on large transport aircraft has progressed from a system designed to increase passenger comfort to a system which must be operating before the aircraft is cleared for flight. Maneuver load and gust load alleviation have been flight tested on a Boeing B-52 CCV and an active lift distribution control system was retrofitted into the Lockheed C-5A force. Elastic mode stabilization or flutter control, flight tested on the B-52 CCV and investigated in wind tunnel and remotely-piloted vehicle tests, has made considerable progress in the last few years but the technology for a good structural dynamics and unsteady aerodynamics model, required for design of an effective flutter mode control system, still needs to be developed. Envelope limiting, such as the angle-of-attack and normal acceleration limiter used on the F-16 to allow use of the full maneuver envelope without danger of stall-spin departure or structural damage, is now being applied to fighter aircraft and some form of envelope warning and limiting is built into most modern aircraft. Direct lift and side force control to improve maneuverability and weapon delivery have also been evaluated on several research aircraft, most recently on a modified YF-16.

Fighter Applications

Active control effects for combat and strike aircraft can perhaps best be described in terms of weight reduction achieved, aerodynamic efficiency, performance and combat score (CHAPTER 3). Related intangible gains which lead to carefree maneuvering are improved handling qualities, reduced pilot workload, and removal of flight restrictions. These benefits are due to artificial longitudinal stability, automatic configuration management, stall/spin prevention, and overstress prevention. Maneuver load control, active flutter control and gust alleviation are also considered but have relatively small gains. It is anticipated that such application of ACT to combat aircraft will provide future pilots with greatly enhanced combat effectiveness, such as 10% improvement in sustained maneuverability, 15% improvement in attained maneuverability, and 25% improvement in radius of action - combined with maneuvering free from the possibility of spinning or overstressing.

Key developments in ACT for combat aircraft include the F-16 multinational fighter, the first production aircraft to incorporate an active control system from its inception (CHAPTER 4). Principal F-16 flight control features are a quadruplex analog fly-by-wire system with fail-operative/fail-operative redundancy, three-axis stability and command augmentation, built-in self-testing capability, relaxed static stability (RSS) and automatic angle-of-attack and normal-acceleration limiting. Performance benefits for the F-16 RSS configuration with its c.g. in the range of 35-40%c, in comparison with a conventionally balanced airplane having a c.g. at 25%c, included some 200kg fuel savings and significant improvements in acceleration time and turn rate for typical combat missions. Angle of attack limitation to 25° and incorporation of an automatic "g" limiter further allows the pilot to consistently find his maximum-turn performance condition and to use it without fear of losing control of the aircraft.

Limited investigations of ACT functions have also been conducted on an F-8C aircraft (CHAPTER 5), equipped with a full-authority triplex digital fly-by-wire (DFBW) control system. The purpose of these investigations was to examine the design, mechanization, and performance of an integrated set of control laws which would be typical of those for projected aircraft employing full-time active controls. The selected control laws emphasized CCV benefits for fighter aircraft. Specific pitch axis objectives were improved handling qualities, angle-of-attack limiting, gust alleviation, drag reduction in steady and maneuvering flight, and a capability to fly with reduced static stability. Lateral-directional design objectives were improved Dutch roll damping and turn coordination over a wide range in angle-of-attack. Modern control design methodology was used to derive control laws focusing on these specific CCV benefits. Major steps in the design process involved linear modeling at some 25 flight conditions, control law synthesis with modern linear quadratic optimization techniques to satisfy conventional design requirements at each flight condition, derivation of approximate gains as functions of air data measurements, and validation of the resultant control laws on a nonlinear six-degree-of-freedom simulation. Flight tests of selected command augmentation, boundary control, ride smoothing and maneuver flap functions were also conducted on the F-8C. While the F-8C performance benefits were relatively modest, excellent agreement was obtained between flight test results and design/analysis/simulation predictions - suggesting that the development of digital CCV control can proceed with confidence.

Another interesting approach for obtaining fighter ACT design data involves flight tests of remotely piloted research vehicles (RPRV's) with active controls (CHAPTER 6) to validate highly maneuverable aircraft technologies (HIMAT). The HIMAT RPRV is a subscale closely coupled canard-wing vehicle which includes relaxed static stability, direct lift control, and a digital active control system. The maneuverability goal for the full-scale fighter aircraft was the ability to sustain an 8g turn at Mach 0.9 at an altitude of 9140 meters; equivalent performance will be demonstrated by the RPRV at an altitude of 7620 meters to match the wing loading of the full-scale aircraft. This goal and nonlinearities in the HIMAT aerodynamics placed unusual demands on the active control system. While a maximum of 10% negative static margin was used as an initial guideline for relaxed longitudinal stability and was increased to 15% later, nonlinear aerodynamics led to more than 30% static margin for some high angle-of-attack flight conditions and low Mach numbers. As a result, an angle of attack limiter was required to assure adequate excess control authority to stabilize the aircraft. Similarly, although neutral directional stability was selected as a limit for the rigid airplane, flexibility effects caused negative stability for small angles of sideslip and special provisions to prevent trimming to nonzero angles of sideslip had to be added to the relaxed directional static stability system. Some penalties were also incurred because of the active control function. Actuators and hydraulic systems were larger than those required for conventional aircraft, and the addition of wingtip ventrals was necessary to compensate for destabilizing canard dihedral effects.

The extension of ACT to propulsion/flight-control integration technology (PROFIT) is another next logical step in the designer's quest for increased performance (CHAPTER 7). Early jet-powered aircraft of the 1940's had simple mechanical flight control systems and simple turboject engines with hydromechanical fuel controls. The next generation of aircraft, introduced in the 1950's, had afterburning turbojet engines to provide greater thrust, and analog electronic stability augmentation systems to provide acceptable handling qualities at supersonic speeds. Aircraft introduced during the 1960's typically had variable-geometry inlets, autopilots and air data computers. The turbofan engine was introduced but there was little or no propulsion/flight control integration. The first step towards integration, the autothrottle, was used on some aircraft introduced in the 1970's which also saw the application of digital computers for fly-by-wire and integrated propulsion control systems (IPCS). The YF-12 cooperative control program subsequently digitally implemented the autopilot, autothrottle, air and inlet control functions. The PROFIT program extends this integration to all propulsion and flight control functions using an F-15 research aircraft. Controlled elements will include the inlet, gas generator, afterburner and nozzle on both engines for the propulsion system as well as pitch, roll, yaw control and autopilot and stability augmentation functions for the flight control system. Appropriate pilot displays and interfaces will also be developed and evaluated. Remote computation capability to extend the onboard computer capability will be provided via telemetry links. Particular concepts to be investigated include engine-inlet-nozzle integration, trajectory optimization (i.e. minimum-noise takeoff, terrain following, energy management), multivariable engine control and engine-problem detection and correction. Results of this work should provide design data for future integrated controls which optimize aircraft performance and permit reduced system costs.

<u>Transport Applications</u>

While active control applications for fighters are clearly driven by military requirements, much of the current motivation for transport applications (CHAPTER 8) comes from the civil arena where designers have turned to ACT to reduce fuel consumption and direct operating costs (DOC). Principal ACT concepts now being considered for transport applications are relaxed static stability and wing load alleviation. Analysis of a typical transport at constant aspect ratio indicates that a 50% design-wing bending-moment reduction can be obtained with existing types of controls (ailerons, spoilers, flaps). This yields some 15% reduction in wing weight and 2-3% reduction in DOC. Similarly, by relaxing longitudinal stability to just above neutral, tail size and load can be reduced with DOC gains of about 2%. These gains are roughly doubled if the aircraft is stretched with a given engine size and performance standard. For a medium-range (2000 n.m.), 200 seat aircraft, combination of these functions can provide 4-5% DOC gains with a constant payload or 7-9% if the aircraft is stretched. Since a 1% DOC reduction on a DC9, BAC-111 or B-737 amounts to a saving of about $50K per aircraft per year for a typical 3000 hour utilization, or about $1M for the 20 year life of an aircraft, such gains are highly significant.

The spectrum of ACT functions that have been investigated for transports includes maneuver load alleviation, gust load alleviation, relaxed stability, flutter suppression, fatigue life improvement and ride-quality control. An indication of the systems impact of such functions can be derived from the following examples.

As part of a program to define means for reducing the energy demands for air transportation, an active control system for commercial application in the Lockheed L-1011 airplane has been developed (CHAPTER 9). Both relaxed static stability and load relief were investigated. The combined impact of these two functions yields a 6½ percent fuel saving for the L-1011; for a fleet of thirty L-1011 airplanes operating for 10 years, that savings translates into about $45M. Based on these promising results, a load relief system providing maneuver load control and elastic mode suppression was selected for flight test. This load redistribution permits a wing tip extension of about 1½ meters per side. Associated reductions in induced drag allow a 3% fuel saving with minimal structural modifications. The load relief system, which uses integrated movements of the ailerons and horizontal tail, has been mechanized and evaluated with an L-1011 aircraft. Measured flight responses closely agreed with design predictions.

A second load alleviation system (CHAPTER 10), aimed at fatigue life improvement, has been in operational service with the entire C-5 fleet since 1975. This system is designated the Active Lift Distribution Control System (ALDCS). The ALDCS mechanization consists of an array of sensors, gains, and filters used with existing control effectors. Flight data, obtained by instrumenting 13 of the modified aircraft, closely followed the system analysis/design predictions. Maneuver and gust load incremental wing stresses were reduced by approximately 30% during normal operation and by some 20% during aerial refueling. Significant improvements in fatigue endurance are projected as a result of the ALDCS, with a conservative 1.25 life improvement factor now being used to track individual C-5 aircraft. System reliability initially predicted to be 3,000 operational hours, actually resulted in a mean time between unscheduled removals of about 1000 hours.

A last example of ACT for transports, the B-1 ride control system, involves one of the first aircraft to include CCV concepts in the early design phases (CHAPTER 11). The Rockwell B-1 has a requirement to provide a specified level of ride quality for the crew. To meet this requirement, the B-1 incorporates a Structural Mode Control System (SMCS) whose main external feature is a set of vanes near the crew station. Since the B-1 has full structural integrity with or without the SMCS operating, a fail-safe approach using dual redundancy in the sensors, electronics and actuators was employed to permit centering of the vanes in case of system failure. Tradeoff studies indicated that 4,482 kg would have been added to the fuselage to meet ride quality requirements without the SMCS. Since the SMCS weighs about 182 kg, active control permits a weight saving of some 4300 kg, a substantial active control benefit. Evaluation of the system performance in flight showed that the SMCS reduced both lateral and vertical load factors to the specified levels without degrading basic handling qualities.

Conclusion

Based on the material covered in this AGARDograph and briefly outlined here, several general observations can be made. First, in the design area, although major advances in ACT design practices have been realized, considerable work remains to be done before structural dynamic and aerodynamic models can be used with confidence in the initial CCV configuration selection. Alternate control approaches, such as parameter-insensitive systems which can deal with errors in predicted aircraft parameters, should also be pursued to permit the increased application of CCV designs. Second, in the ACT system mechanization area, cost-effective reliability and maintainability will be the principal factor in determining the extent of ACT applications. Recent trends towards distributed, fault-tolerant systems and increased integration of aircraft control functions, which permit greater commonality in control processor modules and reductions in overall control system cost, should significantly aid this process. Third, although more and more ACT concepts are now being introduced into flight systems, the implementation of true flight-critical integrated CCV configurations clearly is still a long-term goal and must be viewed as the culmination of a continuing growth in the technology of aircraft design. Active controls are now approaching the relative state of readiness of jet engines when they were first introduced into commercial service; while their near-term potential is high, the best is yet to come.

Acknowledgments

The help and support of the many individuals who participated in this assessment of active control was essential to the development of the AGARDograph. Special recognition should be given to the authors for their extensive efforts in generating the overview and application papers, and to the Guidance and Control Panel members and AGARD staff for their advice and assistance. Finally, the many stimulating and useful suggestions provided by Professor F. Haus during his review of the AGARDograph chapters were of particular value for both the editing process and for development of this summary.

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An expanded version of this summary was presented at the FMP Symposium on Stability and Control, Ottawa, Canada, October 1978.

PART I

DESIGN CONSIDERATIONS

LA CONCEPTION DES AERONEFS UTILISANT LE CONTROLE AUTOMATIQUE GENERALISE

PAR

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La littérature spécialisée en aéronautique a introduit au début des années 70 la notion de Contrôle Automatique Généralisé (CAG). S'agissait-il d'une révolution dans la conception des avions ou simplement de l'évolution traditionnelle vers des performances toujours meilleures ?. Avant de répondre à cette question il convient de faire un retour sur le vocabulaire et d'analyser quelles sont les idées plus ou moins explicitées dans cette dénomination.

On retrouve chez la plupart des auteurs, et en particulier dans le sigle anglo-saxon CCV (Control-Configured Vehicle) l'idée d'une modification fréquente des formes géométriques. La configuration est adaptée au cas de vol à chaque instant. Mais il s'agit du mouvement de gouvernes rapides dont le rôle n'est pas comparable à celui des traditionnelles sorties et rentrées de train d'atterrissage, de volets hypersustentateurs, d'aero-freins variation de flèche de voilure, etc ...

On exprime également l'idée d'asservissement (Control). Les changements de configuration évoqués se font de façon automatique, sans attendre une action particulière du pilote. Mais là encore nous préciserons de quelles sortes d'automatismes il est fait état.

On sous-entend aussi l'idée de caractéristiques artificielles : les qualités de vol que présente l'aéronef CAG ne sont pas naturelles. Ceci est une réminiscence de la terminologie associée à la philosophie des aides au pilotage : on opposait alors l'avion nu, ou l'avion de base, à l'avion complet équipé de ses amortisseurs, stabilisateurs, pilote automatique, etc ... Mais tout ceci est bien relatif : un avion ne fait pas partie du monde naturel (minéral, végétal, animal) qui nous entoure et est en soi une machine artificielle. C'est donc que l'habitude est une seconde nature : nous nous sommes accoutumés à des générations d'avions classiques et leur comportement nous est devenu naturel, toute amélioration en dehors des sentiers battus paraît artificielle.

Enfin le concept CAG implique que les commandes de vol sont l'un des éléments qui conditionnent l'architecture générale du projet d'avion (choix des solutions aérodynamiques, des structures, etc ...) et ne sont plus un simple équipement que l'on dimensionne après coup pour assurer le pilotage, ou rectifier quelque défaut.

Ainsi posées les grandes lignes de ce concept, il paraît intéressant de chercher comment il a pu naître.

QU'EST-CE QUI A PERMIS LA NAISSANCE DU CONCEPT CAG ?

Indubitablement l'origine de cette évolution ou révolution doit être cherchée dans le développement des aides au pilotage classiques : amortisseurs, auto-stabilisateurs, pilotes automatiques. On s'est ainsi habitué à introduire de l'électronique et des asservissements dans les commandes de vol pour améliorer les qualités de vol des avions et diminuer la charge de travail de l'équipage.

Toutefois dans les années 70 un pas important a été franchi dans la miniaturisation des équipements électroniques, la réduction de leur coût et l'augmentation de leur fiabilité. Ainsi ont pu se développer une informatique de bord et une cybernétique efficaces et sûres. On n'en est plus à l'époque où les postes radio étaient toujours en panne !.

Parallèlement les connaissances ont beaucoup progressé en matière de théorie des asservissements, de l'information, du contrôle optimal, et en analyse du fonctionnement des systèmes en modes dégradés. Ces deux points ont donné une très grande confiance dans des concepts fondamentalement basés sur le recours à l'électronique et l'informatique de bord.

.../...

Egalement l'aérodynamique a beaucoup progressé ces dernières années. On a vu éclore une floraison de configurations nouvelles qui ont été étudiées expérimentalement. Ainsi ont pu être connues les répartitions de pression sur des configurations nouvelles, les charges locales, les efficacités de gouvernes nouvelles, les interactions entre gouvernes, voilure, propulsion. Des modèles mathématiques expliquant le fonctionnement aérodynamique de ces configurations, y compris aux grands angles d'incidence et de dérapage et en transsonique sont en cours d'élaboration. Du point de vue de la mécanique du vol les non-linéarités, les hystéresis, les inversions d'efficacité que l'on rencontre parfois, paraissent contrôlables au moyen de gouvernes correctement asservies, ouvrant ainsi des possibilités accrues en performance et manoeuvrabilité.

C'est ainsi qu'est apparu viable le concept d'avion à Contrôle Automatique Généralisé, illustré par des études telles que celles exposées dans d'autres parties de cet ouvrage : HIMAT, PROFIT, AFTI.

QU'EST CE QUI DISTINGUE UN AVION A C.A.G. D'UN AVION CLASSIQUE ?

A notre avis c'est essentiellement qu'il a été conçu dès l'origine pour profiter des avantages des techniques du Contrôle Automatique Généralisé dont il sera question au chapitre suivant.

Ceci implique que l'avion est équipé de commandes de vol électriques.

Celles-ci diffèrent des commandes de vol mécaniques classiques par le fait qu'elles sont bâties autour d'un calculateur. Celui-ci reçoit les ordres venant du pilote (comme des commandes de vol classiques) et les informations venant de différents capteurs (gyromètres, accéléromètres, information de vitesse ou nombre de Mach, d'assiettes, d'incidence, de dérapage, etc ... comme un pilote automatique) et envoie ses ordres aux gouvernes par l'intermédiaire de vérins hydrauliques à entrées électriques. Mais les ordres élaborés par le calculateur de commandes de vol le sont à travers des fonctions de transfert complexes et il n'y a plus de relation simple entre les efforts et les déplacements sur les organes de pilotage et la position instantanée des gouvernes.

La seconde différence par rapport à une chaîne de pilotage automatique classique est que le calculateur de commandes de vol électriques a pleine autorité et peut utiliser le plein débattement de gouvernes dans tout le domaine de vol (vitesse, altitude, facteur de charge, taux de roulis, etc ...)

La sécurité du vol repose sur les redondances et le bon fonctionnement des chaînes de pilotage et non sur la déconnexion après panne ou dépassement de domaine autorisé. Sur un véritable avion à Contrôle Automatique Généralisé il ne peut pas y avoir d'espoir en un secours mécanique et un pilotage manuel (sauf peut-être dans une partie très limitée du domaine de vol). Ainsi un avion comme Concorde, bien qu'équipé d'un certain type de commandes de vol électriques ne peut pas être considéré comme un avion à Contrôle Automatique Généralisé, car la gestion du carburant est organisée de manière à toujours maintenir une marge de stabilité statique positive permettant le pilotage manuel en secours mécanique.

En revanche il ne nous paraît pas y avoir de différence de philosophie entre calculateur analogique et calculateur numérique. Seulement la complexité des fonctions de transfert à réaliser et des comparaisons et tests logiques à effectuer pour s'assurer de l'intégrité des différentes chaînes fait que les calculateurs numériques donnent beaucoup plus de souplesse aux bureaux d'études et sont de ce fait assurés d'un avenir plus prometteur.

De même nous ne ferons pas de différence de principe entre un système de commande direct et un système de commande bouclé. Pour illustrer cette affirmation nous donnerons un exemple pris sur des entrées d'air. Celles du Mirage III ont des parties mobiles qui sont directement mises en place en fonction du nombre de Mach ce qui assure une position correcte des ondes de choc sans que le système ait d'information de retour. Au contraire sur Concorde on détermine la position des chocs par des mesures de pression dans l'entrée d'air et la régulation positionne les rampes mobiles de façon à mettre en place les chocs de façon optimale. Citons encore comme systèmes à commande directe les coordinations roulis-lacet, les conjugaisons entre compensateur de profondeur et braquage de volets, les dispositifs du genre mach-trim, etc .... En ce qui concerne le Contrôle Automatique Généralisé nous pouvons admettre que le bureau d'étude choisira selon la complexité du problème à résoudre, la précision recherchée et les perturbations à craindre, un système de commande programmé (plus simple, plus fiable et moins cher) ou un système de commande bouclé.

En revanche de nombreux auteurs insistent, avant de parler de Contrôle Automatique Généralisé pour qu'il y ait entre les capteurs et les organes de puissance un calculateur électronique. Ainsi se trouvent exclus (un peu abusivement à notre avis) du champ du Contrôle Automatique Généralisé les asservissements purement aérodynamiques ou pneumatiques. Citons comme des exemples de ces techniques que l'on aurait pourtant tort de considérer comme désuettes l'utilisation de l'effet de succion de bord d'attaque pour l'optimisation de la position des becs en fonction de l'incidence sur un avion léger comme le MS 880 Rallye, l'utilisation de gouvernes auto-stables (Gianoli), les systèmes pneumatiques de régulation des moteurs SNECMA du type ATAR, etc ... (sans parler des vieux régulateurs mécaniques de Watt sur machine à vapeur !).

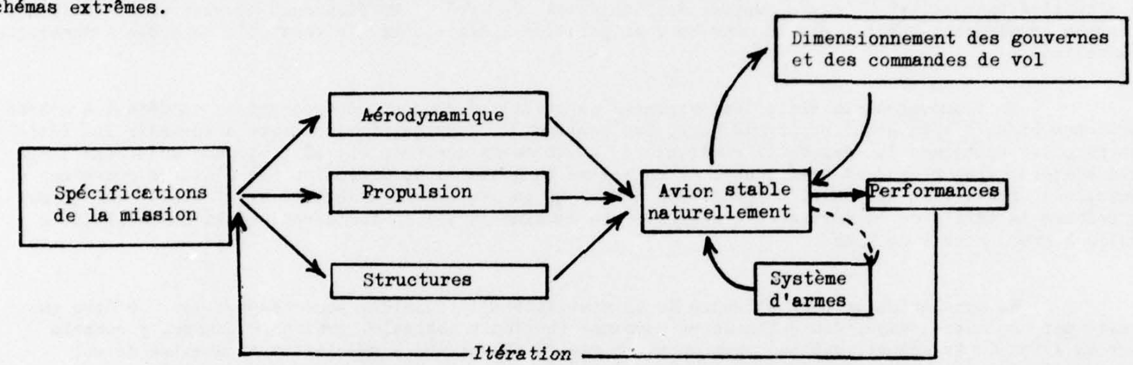
Ce qui caractérisera sans doute aussi les véritables avions à Contrôle Automatique Généralisé c'est du point de vue aérodynamique la prolifération des surfaces mobiles : séparation des gouvernes classiques en plusieurs sous-gouvernes pour des raisons de fiabilité, introduction de gouvernes nouvelles (canards par exemple), cambrure variable, géométrie variable, tuyères orientables et intégration cellule-propulsion. De plus la tendance est qu'aucune de ces gouvernes ne soit utilisée isolément mais qu'elles soient conjuguées selon des lois dont la complexité fait qu'elles échappent à la surveillance humaine, ce qui justifie le recours aux automatismes.

Enfin les appareils à Contrôle Automatique Généralisé se reconnaîtront aussi par le fait que l'organisation et la composition du poste de pilotage seront nécessairement différentes de celles des avions classiques. En effet, comme on le rappellera plus loin la possibilité d'exécuter des manoeuvres nouvelles implique de donner au pilote des organes de commande nouveaux (micro-manipulateurs par exemple). De même le fait de découpler les attitudes, les forces et les moments ne permet plus au pilote de juger de sa trajectoire par la simple observation du monde extérieur. Il faudra donc certainement disposer sur le tableau de bord d'instruments synthétiques nouveaux (généralisation des collimateurs vecteur-vitesse, variomètre à énergie totale, etc ...). On peut aisément imaginer que cela s'accompagnera d'une <u>intégration</u> des systèmes de pilotage, de guidage et de conduite de tir. Et pour terminer il faudra même que le siège du pilote permette de supporter des accélérations sur les trois axes.

Ayant ainsi brossé les grandes lignes de ce qu'est un avion à Contrôle Automatique Généralisé, il est possible de montrer comment sa conception diffère de celle d'un avion classique.

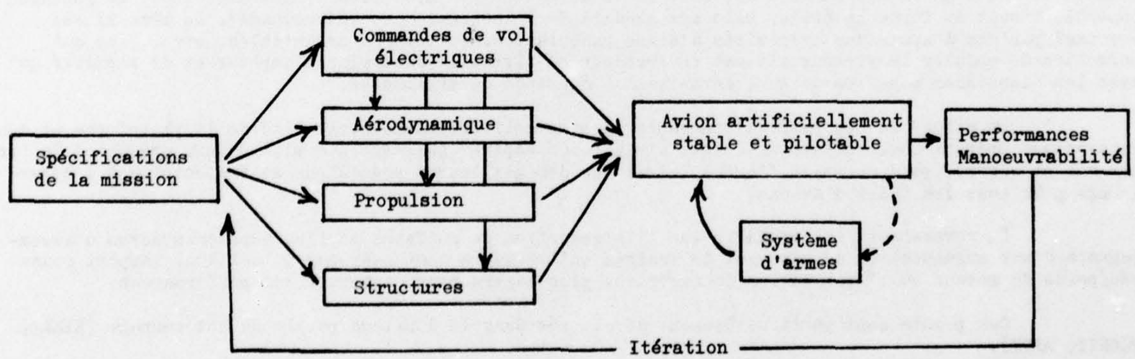
<u>COMPARAISON DES PROCESSUS DE CONCEPTION D'UN AVION CLASSIQUE ET D'UN AVION C.A.G.</u>

Il est particulièrement dangereux de se lancer dans une telle comparaison, car il n'existe pas, à notre avis, de processus unique pour concevoir un avion. Toutefois, espérant bénéficier de l'indulgence des lecteurs, nous reprendrons, pour illustrer de façon caricaturale les deux approches, des schémas extrêmes.



<u>Avion classique</u>

L'essentiel est de remarquer dans ce premier schéma que la boucle de dimensionnement des gouvernes est une boucle annexe qui réagit peu sur le niveau de performance final.



<u>Avion C.A.G.</u>

L'essentiel dans ce second schéma est de remarquer que les choix en aérodynamique, propulsion et structures sont directement influencés par le choix des commandes de vol et que le produit obtenu est artificiellement stable et pilotable. Les commandes de vol sont donc dans la boucle d'itération principale et ont un impact très important sur le niveau de performance final.

.../...

NOTA : On s'est ici limité au domaine technique et on a délibérément éliminé l'itération sur les coûts.

On pourrait faire le même raisonnement et intégrer le système d'armes dans la boucle principale.

L'existence de possibilités de manoeuvre nouvelles permettrait d'itérer sur la mission elle-même et d'augmenter les exigences contractuelles au fur et à mesure que l'on verrait les problèmes techniquement solubles.

FONCTIONS DU CONTROLE AUTOMATIQUE GENERALISE.

- Stabilité Artificielle.

C'est la fonction à laquelle on a pensé en premier.

En longitudinal, sur la plupart des configurations d'avions le recul du centrage permet d'une part, d'obtenir une plus grande portance maximale, d'autre part, à portance donnée une meilleure finesse (diminution de traînée induite et de traînée d'équilibrage).De plus en ce qui concerne la maniabilité, le recul du centrage permet de se contenter d'une surface de gouverne plus faible. On voit donc apparaître un moyen d'augmenter les performances (par amélioration de l'aérodynamique et diminution de la masse) si on n'est pas limité par ailleurs. Or l'une des limites résulte des critéres classiques de stabilité longitudinale statique et dynamique qui imposent de conserver le centre de gravité assez en avant du foyer aérodynamique. L'utilisation d'un système de stabilité artificielle permet de lever cette contrainte qui peut être spécialement gênante vers les grandes incidences (déstabilisantes sur beaucoup d'avions). En fait deux approches sont possibles. L'une que nous appellerons pré-CAG s'exprime ainsi : on accepterait un avion "naturellement" instable avec charges et artificiellement stabilisé, avec la nécessité de larguer les charges en cas de panne de commandes de vol électriques pour retrouver la stabilité "naturelle". L'autre suppose des commandes de vol suffisamment fiables pour stabiliser un avion aérodynamiquement instable même en configuration lisse : c'est le véritable Contrôle Automatique Généralisé.

En transversal-latéral, les exigences de stabilité sont en général prises en défaut à grande incidence et/ou à très grand nombre de Mach. Ceci oblige dans un premier temps à agrandir les dérives avec les problèmes de traînée , de structure et de masse que cela entraîne et dans un second temps cela s'avère malheureusement inefficace. Au contraire la gouverne de direction (et d'autres gouvernes nouvelles) reste très efficace. On peut donc concevoir un système de stabilité artificielle qui permet de réduire la taille de la dérive et d'augmenter le domaine de vol en incidence, et la maniabilité en roulis à grand nombre de Mach.

En conclusion on peut attendre de la stabilité artificielle, sous réserve de n'être pas limité par une autre cause, des gains de performance (portance maximale, traînée et masse, y compris pour un avion de transport, surtout pour un avion supersonique) une augmentation du domaine de vol (grandes incidences, protection contre la vrille) et une plus grande souplesse d'emport de charges externes qui sont en général déstabilisantes.

- Manoeuvrabilité accrue.

Comme il a été dit précédemment, l'installation de Commandes de Vol Electriques autorise la réalisation de configurations aérodynamiques très audacieuses permettant de bénéficier d'interactions favorables entre différentes surfaces. Ceci fournit une portance maximale accrue. De plus l'utilisation des combinaisons de gouvernes nouvelles (par exemple des canards) permet de découpler les forces et les moments et donne ainsi à l'avion des possibilités de manoeuvre nouvelles : contrôle direct de portance, contrôle direct de force latérale, pilotage au-delà de l'incidence de "décrochage". De même il est envisagé parfois d'avoir des extrémités d'ailes basculantes,des tuyères orientables, etc ... ce qui permettra de moduler le vecteur vitesse en grandeur et direction avec plus d'ampleur et de rapidité qu' avec les classiques manettes de gaz, aéro-freins, commande de profondeur.

On attend de ces perfectionnements unebien meilleure manoeuvrabilité de la plateforme et donc un avantage opérationnel certain en combat air-air. On espère également une plus grande précision de tir air-air et air-sol pour un avion d'armes, ainsi que des meilleures précisions de trajectoire à l'atterrissage pour tous les types d'avions.

En revanche il est probable que l'introduction de surfaces mobiles supplémentaires s'accompagnera d'une augmentation de masse et de traînée qu'il faudra compenser par un meilleur rapport poussée/poids du moteur ou l'utilisation de matériaux plus légers pour conserver les performances.

Ces points sont particulièrement développés dans la deuxième partie de cet ouvrage (HIMAT, PROFIT, AFTI).

De la même façon seul le recours à des automatismes sophistiqués permet d'assurer la stabilité, la pilotabilité et les performances des avions V/STOL. Par exemple le cas le plus critique pour les avions de transport STOL est en général consécutif à la panne d'un moteur en approche : il faut alors escamoter une partie des volets pour réduire la traînée, tout en modifiant encore la configuration pour rétablir l'équilibre transversal et latéral, le tout beaucoup plus vite que ce que pourrait faire le pilote seul. Un excellent exemple de ce genre de situation et des solutions apportées par le concept de Contrôle Automatique Généralisé est fourni par le prototype Boeing YC 14

.../...

- <u>Anti-Turbulence</u>.

A partir du moment où l'on dispose d'un grand nombre de gouvernes et où il est possible de découpler les forces et les moments aérodynamiques il devient possible de concevoir un asservissement qui permette de s'affranchir d'une partie des mouvements induits par la turbulence. Cette diminution des effets des perturbations atmosphériques peut être obtenue au niveau du siège du pilote ou en d'autres points de l'avion.

On attend d'un tel dispositif d'antiturbulence, un meilleur confort pour l'équipage, donc une plus grande efficacité opérationnelle en pénétration à basse altitude pour un avion d'armes et une plus grande précision de tir. En suivi de terrain automatique on peut espérer grâce au dispositif d'antiturbulence voler à une altitude moyenne plus basse et être donc moins détectable par les radars ennemis. Pour les passagers d'un avion de transport on obtient un plus grand confort par amortissement des modes souples de la structure. On peut également attendre une moins grande fatigue de la structure, c'est-à-dire une plus grande longévité, où à longévité égale une masse de structure réduite.

- <u>Réduction des charges statiques - Protections automatiques</u>.

Ce point rejoint un peu le précédent. Il est permis d'envisager, chaque fois que l'on n'a pas besoin de la portance maximale, de moduler la portance en envergure de l'aile ou sur les différentes parties de l'avion de façon à diminuer les charges locales et ainsi réduire les contraintes dans les structures. Ceci permet de concevoir un avion soit plus léger, soit plus durable. Cette application concerne évidemment surtout les avions de transport mais elle n'est pas exclue pour les avions de combat.

En fait il sera même plus facile pour les avions à C.A.G. d'assurer le respect de toutes les limitations de domaine de vol (vitesse, facteur de charge, incidence, dérapage) et en particulier la protection contre les vrilles que sur un avion classique. Au bout du compte la sécurité des vols et l'efficacité opérationnelle seront augmentées puisqu'il est possible de concevoir des commandes de vol électriques permettant de tirer le maximum des capacités de l'avion sans craindre de dépasser les limites autorisées (meilleure protection que par une simple alarme classique).

- <u>Anti-flottement</u>.

Deux cas sont à considérer ici.

Dans une première phase on peut constater que l'emport de charges externes s'accompagne parfois, sur les avions classiques, d'un risque de flottement "mou". Traditionnellement le problème est réglé soit par une réduction du domaine de vol de l'avion (limitation de vitesse), soit par une mise au point comportant une modification des modes structuraux par changement des rigidités locales ou des inerties ce qui se traduit toujours par une augmentation de masse. L'utilisation de gouvernes actives, faisant partie de l'avion lui-même, ou ajoutées sur le pylône de fixation des charges en question, fournira dans le cadre du Contrôle Automatique Généralisé une solution élégante à ce problème en réduisant le couplage aérodynamique-structure. Il en résulterait une très grande souplesse d'emport de charges diverses, en particulier de celles qui n'ont pas été prévues à l'origine du projet.

Dans une seconde phase on pourrait imaginer de dessiner un avion si léger et souple, qu'il aurait, déjà en configuration lisse, du flottement à l'intérieur du domaine autorisé, lequel serait supprimé par un dispositif anti-flottement. Une telle solution serait tellement audacieuse que peu de constructeurs d'avions y songent réellement. Curieusement c'est dans le domaine civil qu'une application de l'anti-flottement pourrait être faite en premier. En effet les règlements de navigabilité définissent aujourd'hui une vitesse maximale opérationnelle VMO, une vitesse maximale de calcul VD et imposent que les problèmes aéroélastiques (inversion d'efficacité de gouvernes, flottement, etc ...) soient reportés à une vitesse au moins supérieure de 20 % à VD. Il n'est pas interdit d'imaginer que cette marge de 20 % soit réduite ou annulée si un dispositif automatique supprime en permanence le risque de flottement En effet un accident ne pourrait survenir que de la combinaison d'une erreur de pilotage (dépassement de VMO) et d'une panne du dispositif d'anti-flottement, et le calcul des probabilités montre qu'une telle éventualité est assez rare pour ne pas changer significativement le niveau de sécurité du transport aérien.

- <u>Réduction des coûts</u>.

Précédemment il a surtout été question d'augmentation des performances et de la manoeuvrabilité, de réduction des masses, de meilleure efficacité opérationnelle, de plus grande souplesse d'emploi.

Un autre point doit être évoqué ici : sur les derniers avions classiques, la complexité des commandes de vol (démultiplication non linéaire, restitution artificielle d'efforts, rattrapage des jeux, amortisseur de timonerie, superposition des entrées venant des aides au pilotage, etc ...) font que elles sont extrêmement difficiles à concevoir, à mettre au point et à fabriquer, donc très onéreuses. La nécessité pour des avions d'armes de pouvoir emporter des charges très variées conduit souvent à retoucher les commandes de vol. La longueur et la souplesse des fuselages des gros avions de transport posent aussi de nombreux problèmes de précision et d'élasticité sur les commandes de vol classiques. De même la résolution des risques de flottement de charges conduit souvent à reprendre de nombreux essais de vibration au sol, à refaire des ouvertures de domaine de vol longues et onéreuses. Enfin la fatigue des structures due à la turbulence et aux manoeuvres entraîne parfois un vieillissement prématuré des cellules, d'où la nécessité de remplacer du matériel qui coute très cher.

Il n'est donc pas interdit d'espérer que lorsque les études en cours auront permis de mettre au point des commandes de vol électriques simples et fiables, une économie globale pourra être réalisée par la réduction des coûts de développement, de fabrication et d'entretien et une plus grande longévité de l'avion.

.../...

QUELS SONT LES GRANDS PROBLEMES A RESOUDRE ?

Le Contrôle Automatique Généralisé permettant une nouvelle approche de la conception des avions est l'occasion de mettre en oeuvre les progrès les plus récents dans toutes les disciplines aéronautiques.

En aérodynamique de nombreuses études sont à poursuivre pour régler les délicats problèmes liés au vol à grande incidence : écoulements décollés, écoulements tourbillonnaires, interactions entre tourbillons ou sillages et ondes de choc, retards d'établissement de portance, phénomènes instationnaires, hystérésis dans les écoulements, non-linéarité ou inversion d'efficacité des surfaces mobiles, pilotage par jet, intégration cellule-moteur, etc ...

En matière de commandes de vol, des problèmes de précision, de linéarité, de rapidité des capteurs, de puissance des calculateurs, de précision et de bande passante des servo-commandes sont à résoudre. La sécurité des vols repose sur une très grande fiabilité des chaines de commandes de vol, ce qui impose en pratique d'organiser la redondance des circuits à tous les niveaux(générations électrique et hydraulique, capteurs, calculateurs, organes de puissance) et de prévoir des tests de comparaison (surveillance de panne) en différents points.

Pour être à l'abri d'une panne totale provenant d'une cause unique (trop grande vulnérabilité) certains spécialistes proposent de réaliser une redondance dissemblable, c'est-à-dire de prévoir des chaînes de commandes réalisant en parallèle les mêmes fonctions avec des technologies différentes. Une analyse systématique sur ordinateur (mais peut-on garantir qu'on n'oubliera jamais rien ?) avec simulation des effets des pannes doit être faite et un grand nombre de cas de fonctionnement en mode dégradé sont à prendre en compte (y compris saturation de servo-commandes, ou cycles-limites).

Le réglage des commandes de vol suppose parfaitement développés des outils comme la théorie du contrôle optimal, bien connue la nature des perturbations extrêmes que l'avion est susceptible de rencontrer (statistiques météo), maîtrisées les imperfections de réalisation pratique. Mais, spécialement pour un avion d'armes, il ne faut pas oublier l'exigence d'une très grande manoeuvrabilité et la possibilité de se placer en position inusuelle (par exemple passer en éventail à vitesse nulle) sans perdre le contrôle de la machine. Tout ceci doit conduire à développer des techniques d'essais en vol, d'évaluation de systèmes et d'identification de paramètres, nouvelles et très performantes.

Des problèmes de compatibilité ne manquent pas de se poser : par exemple il faut que les attaches du moteur, ses circuits d'alimentation et de lubrification supportent toutes les accélérations, que les entrées d'air fonctionnent correctement à grande incidence et en dérapage. De même il faudra pouvoir effectuer une visée stable et tirer des engins dans ces conditions de vol.
Citons encore le risque de voir des modes de structure perturber les capteurs qui fournissent les informations nécessaires au pilotage et inversement le risque d'exciter des modes de structure en tentant de résoudre des problèmes de qualités de vol.

L'ergonomie qui avait été longtemps traitée comme le parent pauvre devient une science primordiale. Il faut repenser le problème d'observabilité : nouveaux instruments (synthèse pilotage-navigation- système d'armes) nouvelle disposition du tableau de bord, système d'alarmes adapté aux risques de sortie du domaine autorisé et aux cas de panne à envisager. Il faut repenser le problème de pilotabilité : type et position dans la cabine des commandes à la disposition du pilote, sachant que celui-ci n'a que deux mains et deux pieds pour tout faire. Cela conduira certainement à utiliser des trims et des sélecteurs de modes de fonctionnement des commandes de vol selon les phases de vol. Enfin il faut repenser le problème du confort et du champ de vision : le pilote doit pouvoir conserver ses facultés intellectuelles et physiques dans tous les mouvements de l'avion.

CONCLUSION.

L'évolution normale en aéronautique, comme dans toutes les techniques, est de chercher à améliorer constamment les performances, au sens large, des machines que l'homme conçoit et réalise. Ainsi la philosophie du Contrôle Automatique Généralisé est une évolution normale.

Elle peut apparaître cependant comme une révolution, car il ne s'agit pas seulement de progrès quantitatifs accumulés petit à petit : il y a indubitablement un changement qualitatif, une nouvelle façon d'aborder les problèmes.

Essentiellement cette approche se caractérise en donnant aux commandes de vol électriques un rôle central dans la définition de l'avion. Celles-ci permettent de se libérer de contraintes classiques de stabilité (aérodynamique, structure), mais aussi d'introduire des automatismes qui réduisent la charge de travail (intellectuelle surtout) du pilote et rendent possible l'emploi de combinaisons de gouvernes nouvelles et l'exécution de manoeuvres plus amples voire même nouvelles. Il s'agit donc d'un saut technologique important (comme l'a été en son temps le passage de la propulsion par moteur à explosion à la propulsion par réacteur, comme l'est l'introduction des matériaux composites très légers dans les structures primaires, etc ...) dont on peut espérer qu'il ne se traduira pas par une augmentation du coût des avions.

Toutefois, comme toute médaille a son envers, des difficultés théoriques ont déjà été rappelées dans les paragraphes précédents et surtout les gains que la théorie laisse espérer seront certainement partiellement amputés par les compromis qu'il faudra accepter dans les réalisations pratiques.

Ces différents aspects théoriques et pratiques sont abondamment illustrés dans les chapitres qui suivent.

CONTROL CONFIGURED VEHICLE DESIGN PHILOSOPHY

by

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Specialised papers have introduced in the early 70's the concept of Control-Configured Vehicle (C.C.V.). Was it a revolution in aircraft design or merely a traditionnal evolution towards always better performance ? Before answering this question one should analyse the wording and try to find what ideas are lying under these words.

Most of the authors, and mainly the English-speaking ones, would suggest the idea of a very frequent change in the geometrical shapes of the aircraft. The configuration is adapted to each flight condition at any time. But this implies high frequency moving of control-surfaces which cannot be compared to the classical extensions and retractions of landing gears, flaps, air-brakes or changes in the wing sweep angle.

Also the idea of a control-loop is expressed. The changes of configuration we were speaking of, are made automatically, without any particular action of the pilot. Then we will have to specify what kinds of automatic devices we are referring to.

Also there is an idea of artificial features : the handling qualities of a CCV aircraft are not natural. This is a remembrance of the wording associated with the philosophy of piloting aids : the so-called bare aircraft, or basic aircraft, was opposed to the full aircraft equipped with dampers, auto-stabilizers, auto-pilots, etc. But this is easily arguable : an aircraft is in no way part of the natural world (mineral, vegetable, animal) around us, it is basically an artificial machine. So we are accustomed to a lot of conventional aircraft and their behaviour looks like natural, whereas any improvement outside classical means seems to be artificial.

Finally the CCV concept implies that the flight control system is one of the major design elements of the project (with a large impact on the choices in aerodynamics, propulsion and structures) and not only a simple piece of equipment which has to be defined afterwards in order to provide good handling qualities or to fix some deficiencies.

This being said, it will be interesting to find out how such a concept could appear.

Undoubtedly the origin of this evolution or revolution is to be found in the development of conventional piloting aids : dampers, auto-stabilizers, auto-pilots. People were accustomed to introduce electronics and servo-loops into the flight controls to improve the handling qualities of the aircraft and reduce the crew work-load.

Furthermore, in the 70's, a large step was done : electronic equipments became much smaller , cheaper, and more reliable. Thus airborne computers were developped, efficient, safe and reliable. The time is past when radio-sets were too often unserviceable !

At the same time, our knowledge has progressed in the field of servo-loops, information and optimal control theory, as well as systems functioning analysis. This has given us great confidence in concepts which are fundamentally based on the use of electronics and airborne computers.

These last years also, aerodynamics has made much progress. A great number of new configurations have appeared which have been experimentally tested. The pressure distributions are now known on these new configurations so that local loads and stresses, control-surfaces efficiency, interactions between control-surfaces, wing and engine are identified. Mathematical modelling of the aerodynamic behaviour of such configurations, up to very large angles of attack and side-slip, even in transonic regime, is being done. In terms of flight mechanics, non-linearities, hysteresis, reversals, which might be encountered seem to be controllable. Thus an increased capability in performance and maneuverability is achievable.

Those are the reasons why the CCV concept appeared. It is illustrated by research programms which are exposed in other parts of this AGARDOGRAPH (HIMAT, PROFIT, AFTI, ..).

WHAT ARE THE CHARACTERISTIC FEATURES OF A CCV AIRCRAFT ?

For us the main point which distinguishes a CCV aircraft is that it has been designed from the very beginning to take advantage of the various CCV techniques which will be detailed in the next chapter.

First this implies that the aircraft is equipped with a fly-by-wire system.

In that case the flight controls differ from conventional mechanical flight controls in that they include a computer. This one receives command inputs from the pilot (like conventional flight controls) and various informations (pitch, roll and yaw rates, accélérations, speed and Mach number, attitudes, angles of attack and sideslip, etc.. like an autopilot) and sends output signals to the control-surfaces through electro-hydraulic jacks. Furthermore these outputs from the computer are made of complicated transfer functions and there is no simple correlation between stick forces and displacements and the variable position of the control-surfaces.

A second difference with respect to a conventional auto-pilot is that the computer has full authority and may use the full travel of control-surfaces within the whole flight evelope (speed, altitude, load factor, rate of roll, etc..).

Flight safety relies on redundancy and good functioning of the fly-ly-wire system and not on disconnection after a failure or an exceedance of the authorised flight envelope. On a real control-configured aircraft, there is no hope to fly manually through a mechanical back-up (excepted, may be, in a very limited part of the flight envelope). So, such an aircraft as Concorde, although it is equipped with a kind of electrical flight control system cannot be considered as a CCV aircraft, due to the fact that fuel transfer is organized so that the static stability margin is always positive in order to allow manual piloting in the mechanical mode.

On the other hand we would not make any difference in philosophy whether the computer is an analog one or a digital one. However the transfer functions to deal with, the logical comparisons and tests to be made are so complicated, to check the proper functioning of the systems, that digital computers are much more helpful.

In the same way, we will not make a difference in the principle for CCV applications between an open-loop and a closed-loop system. To illustrate such a statement we will give an example of two air-intake desings . The inlets of the Mirage III fighter have moving surfaces directly commanded as a function of Mach number, which results in a correct positionning of the shock waves, but the command-system does not receive any feedback information. On the other hand, in the case of Concorde the shocks position is determined by pressure measurements in the intakes and moving ramps are served in order to optimize the position of these shocks. To give other examples of open-loop systems we would also quote things like turn-coordinators, gearing between trim-tab and flaps position, such devices as Mach-trims, etc... With respect to the CCV concept, we will admit that the design office will choose, depending on the complexity of the problem to be solved, the required accuracy and the kind of disturbance to be encountered, either an open-loop system (simple, more reliable, cheaper) or a closed-loop one.

However, many authors emphasize, when speaking of the CCV philosophy, that there is an electronic computer between sensors and servo-jacks. Thus, they will exclude from the CCV field (a little bit too arbitrarily to our mind) those control systems which are purely aerodynamic or pneumatic. Let us quote as examples of these techniques, which it would be unfair to consider already absolete, the use of succion effect on the wing leading-edge to optimize the slat position as a function of angle of attack. This is done for instance on a light airplane like the Morane-Saulnier MS 880 Rallye. We would also find the auto-stable control-surface (Gianoli), the pneumatic control system of many engines (for instance the SNECMA Atar), and of course the old mechanical Watt regulator for steam-engines !

What will also probably be a characteristic feature of the control configured aircraft is, from an aerodynamics viewpoint, the great number of moving surfaces : splitting conventional control surfaces into a number of smaller surfaces for the sake of reliability, introducing new control surfaces (canards for instance), variable camber, variable sweep angle, swivelling nozzles, and integrating more closely the engine and airframe. Moreover the tendancy is that, none of these moving surfaces are used separately, all of them are mixed. The complexity of the mixing laws is such that it would not permit human monitoring which justifies the use of automatic devices.

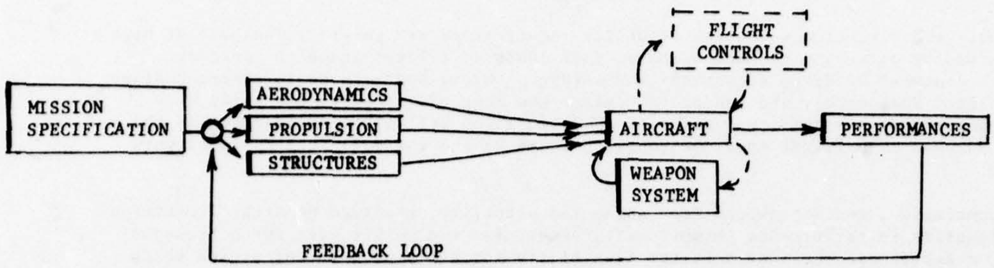
And in the end, the control-configured vehicles will be recognized by the content and the lay-out of its flight deck, which will necessarily be different from those of conventional aircraft. For the possibility of doing new types of maneuvers, which will be explained later on, implies that the pilot must be given new controls (side-stick controller for instance).

Equally the ability to uncouple attitudes, forces and moments does not allow the pilot to realize what is the flight path by simply looking outside. Surely new synthetic instruments will be needed on the panel (things like speed-vector information, total energy rate of change, etc..). It is easy to imagine that such improvements will come along with a higher level of integration of the flight control, navigation and weapon systems. Finally the pilot seat will have to be designed to bear accelerations on three axes.

Having given an overall description of what a CCV aircraft is, it is possible to show in what respect its design differs from that of a conventional one.

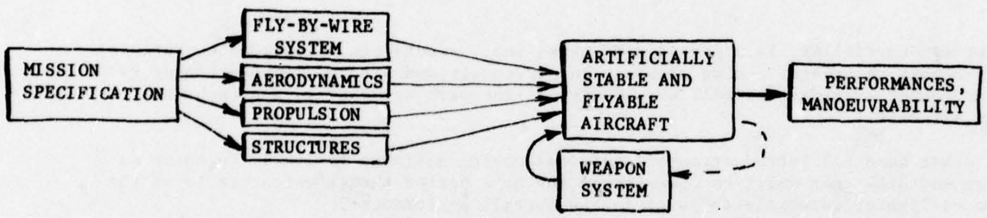
A COMPARISON OF DESIGN PROCESSES : CONVENTIONAL AND CCV AIRCRAFT.

It is a little dangerous to try to make such a comparison, for we think there is no unique process to design an aircraft. However, with the reader's indulgence, we will sketch the two approaches in a very extreme manner.



Conventional Aircraft

The important point in this first case is to notice that the loop which defines the flight controls is an appendix with little impact on the final level of performance.



CCV Aircraft

The important point in the second case is that aerodynamics, propulsion and structure choices are directly dependent on the flight control system design and that the final product is artificially stable and flyable.

The fly-by-wire system is in the main loop and has a very large impact on the final level of performance.

Notes :
- This is limited to the technical field, the cost/efficiency loop has been excluded.
- One could apply the same process and include the weapon system in the main loop.
- Increased manoeuvre capability may allow an iteration loop with the mission specifications themselves. One could add new mission requirements as long as technical problems are likely to be solved.

This being said, let us recall what kind of advantages one may expect from the CCV concept.

THE CCV FUNCTIONS.

Augmented stability.

This is the first application which was thougt of. Longitudinally, for most aircraft configurations, moving the center of gravity aft first gives a higher maximum lift and secondly at a given lift provides with a better lift-over-drag ratio (by reducing induced and trim drags). With respect to manoeuvrability, moving the C.G. aft allows for smaller control-surfaces. Thus a means appears to increase performance (by improving the aerodynamics and reducing the weight) if no other limitation comes first. However classical criteria for longitudinal static and dynamic stability introduce a limitation : the center of gravity must be kept ahead of the aerodynamic neutral point. The use of a stability augmentation system lets the design office get rid of this constraint, particularly embarassing at high angles of attack. In fact two approaches are possible. We will call the first one pre-CCV philosophy : the aircraft will be accepted "naturally" unstable when loaded with external stores and then artificially stabilized. It is necessary, in this case, to jettison external stores after an electrical flight control system failure in order ro recover the "natural" stability of the clean aircraft. The second philosophy assumes that the electrical flight controls are reliable enough to stabilize an aircraft which is unstable even in the clean configuration. That is the real CCV concept.

Laterally and directionnally, the stability requirements are generally not met at high angles of attack and/or at very high Mach numbers. That leads at a first stage to increase the fin area with a number of drag, structures and weight problems and even so in a second stage it is not sufficient. Fortunately the rudder (and other new control surfaces) are still very efficient. One can produce a stability augmentation system which will allow a reduction in the fin area and an increase of the flight envelope (angle of attack) and the roll capability at very high Mach number.

As a conclusion, one may expect from augmented stability, provided no other limitation appears first, benefits in performance (higher lift, lower drag and weight even for a transport aircraft, mainly a supersonic one) and a better flexibility for carrying external stores which are generally destabilizing.

Increased manoeuvrability.

It has previously been said that the use of a fly-by-wire system permits the design of very ambitions aerodynamic configurations. It is then possible to profit from the favourable interactions between various surfaces. This again provides with a higher maximum lift. Furthermore the use of correctly mixed new surfaces (canards for instance) permits the uncoupling of forces and moments, which gives the aircraft new manoeuvre capabilities : direct lift control, side force control, flying beyond the "stall" angle of attack. In the same way, it is sometimes envisaged to design tilting wing tips, swivelling nozzles, etc..., in order to control the speed vector in magnitude and direction, with more authority and rapidity than through conventional throttles, air-brakes and stick.

A better manoeuvrability is expected from these improvements, providing some operational advantages in air-combat. A greater aiming accuracy in air-to-air and air-to-ground shooting is also predicted for combat aircraft, as well as a better flight path accuracy in approach and landing for every aircraft.

On the other hand the introduction of additional moving surfaces is likely to cause an increase in weight and drag that shall be compensated for by a better thrust/weight ratio of the engine or the use of lighter materials to maintain the overall performance.

These points are particularly dealt with in the second part of this AGARDOGRAPH (HIMAT, PROFIT, AFTI).

In the same manner, only the use of sophisticated automatic devices gives enough stability and performance to the V/STOL aircraft. For instance an engine failure during approach results in the most critical case for a STOL transport : it is then necessary to retract part of the flaps to reduce drag, while changing also the configuration to recover lateral and directional trim all this being done much more quickly than any pilot could do. A good example of such a situation is given by the Boeing YC-14 prototype.

Turbulence alleviation.

As soon as a great number of control surfaces are available it is possible to uncouple aerodynamic forces and moments and design a feed-back system to get rid of part of the turbulence induced motion. This reduction in the effects of atmospheric disturbances may be achieved either at the pilot station or in any other part of the aircraft.

Such an anti-turbulence device is expected to provide the crew with better comfort resulting in better operational efficiency when flying at low altitudes in a fighter-bomber and a greater aiming accuracy.

In automatic terrain following mode, thanks to such an anti-turbulence device, it may be possible to fly at a lower mean altitude thus being less in sight of the enemy radars. Passengers in a transport aircraft will also be more comfortable if the airframe flexible modes are well damped. For the same reasons one can expect less structural fatigue, resulting in a longer life, or for a given life requirement a lighter structural weight.

Reduction of static loads - Automatic protections.

This point is partly tied up with the previous one. One may consider, at any time when maximum lift is not needed, the opportunity to vary lift span-wise or on various parts of the aircraft in order to reduce local loads and structural fatigue. Then the aircraft is designed either for a longer life or lighter weight. This application mainly matters for transport aircraft but cannot be excluded for fighters.

As a matter of fact with a CCV aircraft, it will be even easier to ensure respect of all the limitations of the flight envelope (speed, load factor, angles of attack and sideslip) and in particular an anti-spin protection, than for a conventional aircraft. In the end, flight safety and operational efficiency will be increased since it is possible to design a fly-by-wire system such as to allow the aircraft to reach its maximum capability without any risk of exceeding the authorized limits (that is a better protection than a classical alarm).

Flutter suppression.

Here, two cases are to be considered.

At a first stage, one will notice that carrying external stores generally results in a risk of flutter. Traditionally this problem is fixed either by limiting the flight envelope (speed limitation for instance) or by developping some modification. The structural modes are modified by changing local stiffness or inertia, which always results in an increase of weight. The use of active controls, part of the aircraft itself or added on the pylons, will provide, as part of the CCV philosophy, an interesting solution to the problem, reducing the aerodynamics structures coupling. This would give a greater flexibility to carry various stores, mainly those which were not listed at the beginning of the project.

In a second stage, one could imagine the design of so light and flexible an aircraft that it would have flutter, even in the clean configuration, within the authorized flight envelope. This would be suppressed by an anti-flutter device. Such a solution would appear so bold that very few manufacturers really would think of it.

In the civil aircraft field this second type of anti-flutter technique might be attempted first. For the airworthiness requirements define a maximum operational speed V_{MO}, a maximum design speed V_D and require that aeroelastic problems (control reversal, flutter, etc...) will not appear below a speed greater than V_D by 20 % at least. It is not unrealistic to imagine that this 20 % margin could be partially reduced or totally suppressed if an automatic device always eliminate the flutter risk. In effect, only a combination of pilot error (exceedance of V_{MO}) and system failure could result in an accident. The probability of such an event is remote enough and does not significantly change the safety level of air transportation.

Costs reduction.

In the previous paragraphs we have mainly dealt with performance and manoeuvrability improvements, weight savings, better operational efficiency and flexibility.

An other point must be emphasized at this time : on the latest conventional aircraft the flight controls complexity (non-linear gearing, artificial feel systems, play take up, control linkage dampers, etc...) makes it very difficult to design, develop and produce these equipment. They are very expensive. Furthermore the need for fighters to be able to carry various loads implies frequent fine re-tuning of the flight controls. The fuselage length and flexibility of large transport aircraft introduce also a lot of problems in accuracy and elasticity of conventional flight controls. Equally, to avoid flutter risks with stores it is often necessary to do a great number of ground vibration tests and in-flight envelope extensions, which are time and money consuming. And also the structural fatigue, due to atmospheric turbulence and manoeuvres, sometimes cause an early aging of the airframe. It is then necessary to replace a very expensive stock.

For these reasons one can hope that, when current studies have lead to the development of simple and reliable electrical flight controls, over all savings will be made by reducing development, production and maintenance costs and increasing aircraft life.

WHAT ARE THE PROBLEMS STILL TO BE SOLVED ?

The CCV concept being a new approach in aircraft design gives an opportunity of introducing the most recent progress in all aeronautical techniques.

In aerodynamics, many studies are to be carried out to fix the difficult problems associated with high angles of attack : separated flows, vortices, vortex-wake-shock waves interaction, unsteady effects, flow hysteresis, non-linearities, control reversal, jet flow control, engine-airframe integration, etc...

With regards to flight controls, questions are raised about the accuracy, linearity and rapidity of the sensors, the power of computers, the accuracy and bandwith of the jacks. Flight safety calls for the highest reliability of fly-by-wire systems : pratically that means that all circuits have to be redundant at every level (electrical and hydraulic generation, sensors, computers, jacks) and comparison checks must be organized in various points (failure monitoring). To be protected against a single cause resulting in a total failure (too great vulnerability) some specialists suggest to achieve dissimilar redundancy, that is to design parallel flight controls producing the same functions through different technologies. A comprehensive analysis (but who can guarantee that nothing has been forgotten ?) with simulation of failure consequences must be done and a great number of degraded modes must be considered (including jack-stalling, limit-cycles, etc...).

Fine tuning of the flight controls implies that optimal control theory has to be developed, extreme disturbances are to be known (meteorological statistics for instance) and that there is no unacceptable practical unaccuracies left. Specially for a fighter, one must not forget the requirement for a very high manoeuvrability which means also the possibility of reaching unusual flight conditions (for instance a zero speed associated with 90° pitch attitude). From these conditions control must not be lost.

All this will make it necessary to develop new efficient techniques for flight tests, systems assessment and parameter identification.

Compatibility problems will appear : for instance engine attachment, oil and fuel circuits must be able to bear all accelerations, air intake must work properly at high angles of attack and sideslip. Also one must be able to make a stabilized aiming and missile firing under these flight conditions. Let us state the risk that structural modes would disturb those sensors which provide the information needed by the flight controls, and symetrically the risk of exciting structural modes in trying to improve handling qualities.

Ergonomy, which has remained in a secondary position for ages, becomes now a major science. The observability problem has to be reconsidered : it will be necessary to develop new instruments (for piloting, navigation,weapon systems state) new lay-out of the instrument panel, an alarm system to be adapted to the risk of exceeding the authorized flight envelope and to the failure cases to be considered. The controllability problem has to be reconsidered : type and position of the controls in the cockpit, knowing that the pilot has only two hands and two feet ! This will surely lead to the use of trim and control mode selectors, associated with flight phases. Finally the problem of comfort and field of vision has to be reconsidered : the pilot must be able to keep all his intellectual and physical capabilities during all aircraft movements.

CONCLUSION.

The normal trend in aeronautics, like in all other techniques, is to try to improve the performance, in general, of those machines that men design and build. To that extent the control-configured vehicle design philosophy is a normal evolution.

But it may also appear as a revolution, since it does not only involve quantitative progress acquired step by step : undoubtedly there is a qualitative change, a new means to tackle old problems.

Essentially this approach is characterized by giving electrical flight controls, a central position in aircraft design. The fly-by-wire system is the means to get free of traditional constraints of stability (aerodynamics, structures) and also to introduce automatic devices, reducing pilot work-load and making it possible to use mixed new control-surfaces and perform extended and even new manoeuvres.

Thus it is really an important technological step (as the introduction of jet propulsion compared to reciprocal engines was in its time, as the present use of very light composite materials in the primary structures is). One may even hope that this progress will not result in increased aircraft costs.

Nevertheless, theoretical difficulties have already been raised in the previous paragraphs and moreover those benefits that theory shows likely to be obtained may be partially spoiled by the compromises to be accepted in practice.

These various theoretical and practical aspects are widely illustrated in the next chapters.

ACTIVE-CONTROL DESIGN CRITERIA

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SUMMARY

The question of design criteria for active control aircraft is one of the key issues involved in the design effort. If sufficient benefits are to be realized, in terms of decreased weight, reduced fuel consumption, increased performance, etc., applicable design criteria must be established which take into consideration the design improvements derived from the presence of active control systems. In this paper we will discuss briefly the definition and background of active control technology and then cover each of the functions contemplated to be performed by active control systems. The various design criteria for each will be discussed and the subject of government regulations affecting aircraft design will be touched upon briefly. This paper is based on the work presented in Reference 1, supplemented by the considerable developments of the last three years.

ACTIVE CONTROL TECHNOLOGY (ACT)

The question of just what kind of an airplane configuration satisfies the definition of an active control aircraft has been confused in the past. Several designations for this type of aircraft have been used (fly by wire, CCV, etc.), but an aircraft utilizing active controls can, in general, be identified as one in which significant inputs (over and above those of the pilot) are transmitted to the control surfaces for the purpose of augmenting vehicle performance. These inputs, derived from various sensors and properly processed, can be utilized to provide reduced trim drag and tail area through stability augmentation, reduce structural fatigue, alleviate maneuvering loads, suppress flutter, and improve ride comfort. If applied in a meaningful manner early in the vehicle design, ACT can have a significant impact on vehicle weight and geometry, thus leading to the designation of a "control configured vehicle" (CCV).

The term "fly by wire" describes a method of system implementation whereby electrical commands are used. This approach is suited to the application of active controls in that it provides an ideal interface between the basic command system and the sensor and signal processing elements. Even more advanced methods of signal transmission, such as those utilizing fiber optics, are being developed to overcome some of the drawbacks of electrical transmission systems.

One still reads in the literature items which would indicate that the active control transport will be a sudden and rather drastic innovation from the long line of transport development over the last 40 years. As a matter of fact, it is not a sudden transition, but a continuing growth in the technology of transport aircraft design. Every modern day aircraft, to some extent, incorporates some of those functions which are rather loosely tied together under the name of active control technology (ACT). It became apparent in the early twin engine transports that the pilot had difficulty exerting sufficient stick force to move the control surfaces of the aircraft. The designers rather ingeniously provided the pilot with aerodynamic tabs in order to reduce his workload and make the aircraft easier to control. As aircraft continued to grow, hydraulic-powered control systems were implemented. Although these early systems were designed in a manner which still provided the pilot with a mechanical linkage to the surface in the event of hydraulic failure, the modern day transports as well as high performance military aircraft now completely depend on the hydraulic system, and the designer (and the pilot) must rely on the reliability of the redundant systems which supply the power for the control surfaces.

Along with this reliance on hydraulic systems, the pilot has also experienced an increased dependence on the many other systems which must function properly for the economical and safe operation of the modern, high performance aircraft. In return, flying qualities and comfort have improved, reducing pilot effort and fatigue in transport aircraft, and improving combat performance and survivability in military aircraft. The pilots are slowly learning to accept the fact that certain critical conditions must be automatically detected and appropriate remedial action taken without pilot activity. In this context then, the incorporation of further active controls on the aircraft is not a sudden transition but a steady progression toward a more modern and efficient aircraft design.

Design criteria and government regulations have generally responded to the design innovations such as active control rather than leading these technical advances. It is important at this time, with active controls of various kinds becoming more and more common, that design criteria and regulations lead the effort rather than follow these new designs. Most of the immediately available active control techniques have been well explored theoretically and, in fact, have been and are being demonstrated each day on a wide variety of experimental, commercial, and military aircraft. This demonstration experience is illustrated in Table 1.

The important conclusion to be drawn from this table is that when discussing active control technology, one is dealing with a technology which in some cases is well advanced, including operational experience on in-service aircraft. Certainly if one compares this, say for instance, to the introduction of jet engines on aircraft, one would be forced to the conclusion that the relative state of readiness of active controls approaches that of jet engines at the time they were introduced into commercial aircraft. It is also important to note, however, the disparity between the status of various functions. For instance, the yaw damper is well received and in fact may be mandatory for safe handling qualities, and has many thousands of transport flight hours behind it. On the other hand, flutter control is by comparison only in its infancy. This leads to the conclusion that one must approach active control technology not as an all-inclusive blanket addition to an aircraft, but in a step-by-step procedure with each new subsystem being carefully verified on the basis of cost effectiveness, need, and reliability.

In Table 1 the experience gained in the many missiles and spacecraft, both manned and unmanned, which have flown with complete automatic control and hands-off operation is not considered. Every Apollo mission from launch to splashdown was a demonstration of active control technology. The rapidly increasing technology of remotely piloted vehicles is also quickly adding to the storehouse of knowledge on how to takeoff, land, and navigate in a hands-off, completely automatic mode. Indeed, one must consider that more than 25 years ago the first hands-off flight of an aircraft was demonstrated from takeoff to landing.

ACTIVE CONTROL FUNCTIONS

STABILITY AUGMENTATION (RELAXED INHERENT STABILITY)

Relaxed inherent stability is conventionally defined as a reduction in the stability of the short-period attitude modes of rigid-body aircraft motion. That is, reductions in inherent stability result from the reduction of aerodynamic restoring moment with respect to angle of attack or angle of sideslip or a reduction of aerodynamic damping for the unaugmented (basic) aircraft. In principle, relaxed inherent stability can also refer to reduction in stability for other modes of aircraft motion.

This is a very important departure because the basic stability parameters in both the pitch and yaw axes have established the criteria for a considerable portion of the aircraft design. It is, however, one of the prime areas of the application of active control technology. Desirability of relaxed inherent stability arises from the possibility that with smaller tail volumes significant reductions in total aircraft drag and gross weight can be realized with invariant payload and mission. This is substantiated by the results of industry design studies which show that relaxed inherent stability combined with center-of-gravity control offers the largest payoff for the aircraft in terms of gross weight reduction.

Pitch Stability

Relaxed longitudinal stability is one of the largest areas of potential benefit to be derived from the application of active control technology. We will not, in this paper, go into the details of how one implements active controls for the relaxed stability condition, but some of the design criteria involved will be discussed. First, the basic considerations influencing wing location and horizontal tail surface size and location are affected. The horizontal tail area, for instance, is normally set for a conventional design to meet stability and control requirements over the desired center-of-gravity range. Typically, the forward center-of-gravity limit tail area requirements have been set by trim capability or by control required to develop maximum lift in the landing configuration. The critical condition depends on the type of control system selected, i.e., separate trim and control surfaces or a single surface providing both control and trim. Aft CG limit requirements have generally been set by minimum levels of static longitudinal stability. For the active control relaxed stability design, the horizontal tail area may be set by either the landing case or by the pitching moment required for takeoff rotation at forward CG and by the reduced level of stability or by the pitching acceleration required for control in the presence of gusts and other external disturbances at aft CG. These points are illustrated in Figure 1. The active controlled aircraft is rebalanced with a farther aft center-of-gravity range and a smaller horizontal tail. This in itself raises the importance of sometimes obscure criteria affecting landing gear location. Tip-back tendencies and criteria for nose gear steering effectiveness in areas where aerodynamic controls are ineffective can exert a significant influence on the final configuration.

The deficiencies in inherent stability might be compensated for by augmenting C_{M_α} and C_{M_q}. The degree of instability allowable will be determined not only by increasing stabilization control power requirements, and failure conditions, but also by the variation of trim drag. As the balancing tail load changes from a down load to an up load, the longitudinal component of the tail lift vector changes from a thrust to a drag, significantly increasing tail drag. Minimum trim drag usually occurs near zero static margin, as illustrated in Figure 2. The exact center-of-gravity location for minimum trim drag is dependent on the particular configuration and even on the wing aerodynamic design.

As shown in Table 1, some experience has been gained with relaxed inherent stability. Many jet transports have augmented static longitudinal stability where the augmentation is a function of airspeed. However, the magnitude of relaxation possible with active control will change the design criteria. Perhaps one of the most disturbing ideas that accompanies these changing criteria is that the easily calculated inherent stability requirement has been replaced with a possible pitching acceleration requirement based upon the rather uncertain magnitude of airplane response required under varying conditions of flight and levels of atmospheric disturbance. Experience shows that existing levels of airplane response are satisfactory, but there is relatively little data defining minimum satisfactory response.

Flying qualities criteria may also be affected by dependence on augmentation, especially in the pitch axis. The subject of flying qualities criteria will be further discussed later.

Directional Stability

As shown in Table 1, this is the area where active control has seen the largest and most widespread application in transport aircraft. The yaw damper (an augmented directional stability and control system would more completely describe the systems currently flying on large transport aircraft) has progressed from a system which was a nice passenger comfort add-on feature to a system which must be operating in order for the aircraft to be cleared for flight. Despite this, there is probably much less to be gained by relaxed directional stability than by relaxed longitudinal stability. Currently, vertical tails are sized to provide static directional stability, dynamic lateral-directional stability, and asymmetric thrust control. Minimum control speed criteria are either critical or close to it in sizing the vertical tail on most transport designs with wing-mounted engines. Selection of the minimum control speed criteria may be somewhat arbitrary, but two things are generally considered:

a. The air minimum control speed must be less than the landing approach speed at all gross weights.
b. Ground and air minimum control speeds may dictate the minimum takeoff runway length and should be set to provide the desired capability.

With relaxed inherent stability and if asymmetric thrust control is not limiting, the tail size may be reduced to the level where stabilization control or airplane control response, as during a crosswind landing decrab maneuver, become limiting. In either case, new and unfamiliar design criteria are required.

CONTROL AUGMENTATION

The term control augmentation is applied when pilot commands are modified to produce a desired aircraft response instead of merely a control surface motion. Many variations are possible from merely quickening aircraft response to not only shaping the response but altering its character, as in attitude or rate command systems. Control surfaces affecting one or more axes may be involved as well as direct lift or sideforce control surfaces and even throttle or drag controls. Control augmentation is often combined with stability augmentation to provide the total desired vehicle flight characteristics. The resulting combination is often referred to as a stability and control augmentation system (SCAS).

Control augmentation has been around for many years, often as modes of automatic flight control systems. Systems in common use are often referred to as stick-steering or control-wheel-steering (CWS) systems. Those in use on commercial transport aircraft usually provide a rate-command, attitude-hold pitch control mode. Full SCAS systems are used in the AMST prototype aircraft to provide desirable flying qualities, primarily during STOL operation.

Control augmentation criteria will be discussed further under Flying Qualities.

CONTROL OF AIRCRAFT CENTER-OF-GRAVITY

This area of active control has also been growing rather rapidly. Some transport aircraft require a sequence of wing fuel management in order to maintain the necessary margins against flutter. Maintenance of the CG within limits on current transports also dictates certain management sequences. Systems for maintaining and scheduling an optimum CG location are included in most supersonic transport designs and are in operational service on the Concorde. It is, of course, apparent that the flight envelope and characteristics of the supersonic transport result in increased benefits from CG control compared to a purely subsonic design. The high fineness ratio of the SST configuration also enhances the ease with which this can be accomplished.

Automatic center-of-gravity control can offer significant design advantages in the following ways.

- Reduction of the design center-of-gravity range at given flight conditions may allow further reduction in the horizontal tail volume coefficient (refer to the indication of "CG range" in Figure 1).
- Minimization of total drag with respect to center-of-gravity location during cruising flight, as illustrated in Figure 2.

RIDE QUALITY

Ride quality control refers to automatic control system functions which reduce to acceptable levels the accelerations to which passengers and crew are subjected. Factors such as low wing loading, poorly damped dynamic stability, structural flexibility, atmospheric turbulence, and high-speed, low altitude flight all contribute to poor ride comfort.

Ride quality problems have tended to be secondary considerations with respect to resolution of structural load and flexibility problems. In fact, ride quality has not been a major factor in transport design, because the criteria for ride quality in the commercial environment are:

- Ride must be merely acceptable to passengers.
- Ride must be competitive with contemporary commercial aircraft.
- The aircraft must be readily controllable in turbulence.

Transport aircraft evolution has, up to now, enjoyed a history of improving ride quality due to higher flight altitudes, increased aircraft mass, and higher wing loadings. This trend may be changing, however, as wing loadings will probably increase little above today's levels and aspect ratio will increase in the search for reduced fuel consumption.

The control techniques for improving ride quality are fairly well established both theoretically and operationally. Many commercial transports have some degree of ride quality control provided by means of conventional control surfaces. The yaw damper systems of modern jet transports improve ride quality even though their fundamental purpose is to improve handling qualities.

Active control for gust load alleviation has demonstrated greatly reduced response to turbulence, thus assuring a greater comfort for passengers. A typical reduction in aircraft response to turbulence obtained during the B52 LAMS and CCV programs is shown in Figure 3. It will be noted that the decrease in response to turbulence is sensitive to the aircraft structural modes and that a uniform reduction at all frequencies is impossible. It is doubtful that ride quality design criteria will result in weight savings, so the competitive pressure to supply a smoother ride will probably dictate the control system design criteria for commercial aircraft. However, some military aircraft roles may require ride control improvement to reduce crew fatigue or even for mission completion in turbulent atmospheric conditions. The B-1 bomber is an example of this type of application.

LOAD CONTROL

Load control refers to the use of passive or automatic control functions for the purpose of regulating the net load and distribution of load applied to the aircraft structure. There are four main facets of load control. To some extent, all must be considered simultaneously to achieve a well-balanced design although some may receive considerably more emphasis than others.

Maneuver Loading

Maneuver loading is that portion of forces acting on the airframe which result from maneuvers required to maintain the aircraft on the intended flight path. The distribution of this loading over the airframe can have a powerful effect upon the shear forces and bending moments which must be transmitted at given points in the structure. The ability to tailor the distribution of maneuver loading over the airframe is maneuver load control. Maneuver load control can have a significant impact upon structural implementation and even upon configuration.

The impact of tailoring maneuver load distribution may be far-reaching. If the maximum reduction in fatigue loading is to be achieved, maneuver load control would be desirable during all maneuvering. When applied to the wing, this usually implies an "unloading" of the outer wing, thus reducing the root bending moment, as illustrated in Figure 4a. A high-wing loading transport may possibly be limited in cruise altitude by maneuver requirements such as those specified in the British Civil Airworthiness Requirements. Unloading a portion of the wing would tend to reduce maneuver capability, particularly if wing stalling occurs inboard. Thus, maneuver load control might tend to limit wing loading or dictate a new approach to wing aerodynamic design. This situation may be avoided by utilizing maneuvering flaps to increase lift on the inboard portion of the wing, Figure 4b. Additional aerodynamic and structural design considerations would still be required, along with new modes of control akin to direct lift control.

Gust Loading

Gust loading is that portion of forces acting on the airframe which result from atmospheric disturbances.

Gust-load control is accomplished by the following means:

- Controlling the aircraft in such a way as to produce a net incremental load factor which tends to cancel the net gust-induced load factor. Because of aircraft inertia, this is best accomplished with direct lift control devices.
- Controlling the distribution of the incremental load which tends to cancel the gust-induced load in such a way that their distributions are similar.
- Augmenting damping for modes excited by gusts.

The extent to which gust-load control is effective in performing all three listed functions can have a significant impact upon the structural strength and fatigue requirements.

Experience indicates that the impact of maneuver and gust-load control on reduction of structural requirements tends to be significant only when both maneuver and gust-load control are practiced simultanously. If only one of these load-control objectives is addressed, then the other source of loading often becomes critical before any significant reduction in structural requirements is realized.

Fatigue

Cyclical loading is produced by forces applied to the airframe which result in stress-level oscillations in the structure. Fatigue damage results from accumulated stress cycles at given stress levels and at critical points in the airframe. Fatigue damage control is a technique for reducing the fatigue damage rate by using active controls to reduce the number of transient cycles at the higher stress levels to which the structure is subjected during operation. The use of active landing gear systems is now also being studied to reduce ground taxi load cycles as well as improve taxi ride comfort.

The frequency range of damaging loads extends from once per 100 flights (e.g., from very "firm" landings" to the once per flight of the so-called ground-air-ground (GAG) cycle and to the characteristic frequency of the response to turbulence. The transition between the ground mean loading and the airborne mean loading of the GAG cycle accounts for as much as 80 percent of fatigue damage on the lower wing skin on some contemporary transport aircraft. Most of the remaining damage accrues from incremental loads in the 1/4- to 1/2-g range.

Since the mean-to-mean fluctuation of the GAG cycle is not amenable to control, active control offers potential reduction of longitudinal loads only for the incremental load fluctuation about the mean level of the GAG cycle. Large potential for load reduction exists for lateral loads because there is no GAG cycle effect.

The more modern approach of "rational probability analysis" coupled with careful mission analysis rather than the application of the classical, rather arbitrary approach of a discrete gust must be used in order to realize the benefits to be gained from the application of active controls to load alleviation. The obvious point here is that if careful mission analysis is applied to the calculation of the fatigue life of the aircraft and if the load alleviation control systems are assumed active during the entire life of the aircraft, the weight of the aircraft structure could be reduced for the same fatigue life. This approach is being used both on new designs and to analyze and extend the fatigue life of aircraft currently in service, although it is difficult at this time to come up with definite criteria. The problem is to accurately forecast the manner in which the aircraft will be operated in service and to decide on the missions to be considered. Conversion of an aircraft to a different role can significantly alter aircraft loading, numbers of landings, etc. The design criteria must consider whether or not such alternate or potential uses will be taken into account. However, the combination of maneuver load control plus gust load alleviation can result in reductions of load fluctuation.

Elastic Mode Stabilization (Flutter Control)

Elastic mode stabilization refers to the use of automatic control functions which alter the apparent structural mass or stiffness, or aerodynamic damping. Acceptance of this active control mode has increased dramatically in the last few years. Active mode stabilization or flutter control has been demonstrated in flight on the CCV B-52 and in wind tunnel tests. Studies are now underway which may lead to incorporation of EMS on high-aspect-ratio transport aircraft in the not-too-distant future. The nature of the control laws for achieving the required augmentation is sensitive to the unsteady aerodynamic forces and is also sensitive to the mass and stiffness distributions of the airframe. The flutter safety margins will also be influenced by the presence of other active control functions. For instance, in the case of relaxed inherent stability, it is necessary to have a relatively wide bandwidth control system to cope with the unstable short period mode roots. This control system will tightly couple with the basic flutter modes of the wing-nacelle-fuselage combinations on a large transport aircraft. This will mean that the safety margin for flutter will be a function of the control system loop gains and general design. Criteria will also have to be carefully developed to account for backup modes of operation of the flight control system. Initial applications of active flutter control will almost certainly be limited to reducing flutter speed margins and not for structural integrity within the normal operating envelope.

Other Load Limiting

Other forms of load limiting are also useful. Surface actuator capability not only limits the airplane maneuver envelope but tends to limit the maximum load on the surface itself. Many examples of load limiting are in use today on jet transports. Flap blowback or deflection limiting is in use on several aircraft to limit structural loads. Rudder deflection limiting as a function of flap angle and airspeed is also commonly employed. The "old fashioned" means of load limiting, short of pilot strength, was often accomplished by providing pressure relief valves in surface actuators, thus allowing the surfaces to blow back when aerodynamic hinge moments exceeded the actuator drive capability. Unfortunately, the margin required between hydraulic system supply pressure and relief valve opening pressure to ensure proper system functioning resulted in a design requirement for loads considerably higher than the actuator driving force. As a result, most of these systems have been replaced by automatic systems which limit surface travel as a function of airspeed, Mach number, etc. As other active control modes are used to reduce structural weight and margins, the use of these approaches will have to be considered in concert with the other control modes in a synergistic design procedure.

ENVELOPE LIMITING

Envelope limiting refers to those functions in an active control system that prevent or discourage operation of the aircraft outside its design or operating envelope.

Most modern aircraft currently have some form of envelope limit warning and envelope limiting, although not usually in the ACT sense. Envelope limit warning takes the form of stick shaker systems which warn of an approach to the stall and overspeed warning systems which warn that maximum operating speeds have been exceeded. Envelope limiting is provided by pilot strength limitations, control surface actuator capability, autopilot authority, and autopilot automatic cutoffs (ACO), for example. The limits provided by pilot or actuator strength may or may not be within the structural design envelope of the aircraft. For instance, the pilot does, in some flight regimes, have the capability of exceeding the design limit loads about all axes.

The concept of envelope limiting is now being applied to fighter aircraft to allow use of the full maneuver envelope without danger of a stall-spin departure. The somewhat similar stick pusher system approach for transports has not achieved wide acceptance among either pilots or aircraft designers, however. For transport aircraft, the incorporation of active control could supplement the present warning and limiting features with an automatic function which prevents the aircraft from entering into a forbidden flight regime. Angle of attack and sideslip limiting could avoid post-stall loads and flight characteristics problems, and reduce vertical tail loads. Overspeed limiting could reduce the required margin between maximum operating and design dive speeds, as shown in Figure 5, reducing design loads and allowing a lighter structure. The possibility of atmospheric-caused upset must be considered in establishment of minimum margins. It would then be necessary to assure that the flight control system will satisfactorily handle this job even in the backup or degraded operational modes to assure that the aircraft is operated within the criteria established for strength of the structure. G-limiting might not be desirable, as there have been several cases where the ability of an aircraft to exceed the nominal design limit load factor may have avoided a catastrophic accident following upsets at low altitudes. Since the true design limit load factor varies with aircraft loading, configuration, Mach number, etc., the use of load sensing for envelope limiting would seem to be a better approach from this standpoint.

DESIGN CONSIDERATIONS AND REGULATIONS

Key elements in bringing ACT to the point of commercial application are:
1. The ability to achieve a significant benefit justifying ACT application.
2. Availability of proven design criteria.
3. Availability of proven design practices to guide the combined application of ACT functions.
4. Limitations on ACT applications that may be imposed by regulations.

Design criteria are derived from many sources. Perhaps the most important are the manufacturer's experience and design philosophy. Studies performed or financed by NASA and DOD provide a large fund of suggested criteria and data which the designer uses in selecting his criteria for application.

For military aircraft, mandatory military specifications are usually applied to obtain what are considered to be good characteristics. In the civil or commercial world, competition usually ensures that the aircraft have the best characteristics obtainable, within reason. Safety is therefore the primary purpose of the airworthiness requirements contained in Part 25 of the Federal Aviation Regulations. Besides the U.S. FAA regulations, the designer must also consider the requirements that may be imposed by other nations on aircraft offered for sale within their territory. Among nations having specific airworthiness requirements are the United Kingdom, France, the Netherlands, Germany, Italy, and Australia.

Existing Federal Airworthiness Regulations (FARs) in Part 25 do not place significant constraints on the application of ACT. Those constraints which are imposed tend to be of the following kinds.

- Interpretations of the fundamental regulatory intent were not made in a context which included ACT.
- Practical considerations for demonstrating compliance sometimes require arbitrary maneuvers, tests, or environments which have no counterparts in normal or degraded modes of operation.
- The view of acceptable safe practice tends to be consistent with the current or recent past state of the art but not to the projected state of the art.

Existing regulations [FAR 25.21(e)] already recognize that acceptable flight characteristics may depend upon a stability augmentation system or upon other automatic or power-operated systems. This clearly admits ACT systems as well. Revisions to the regulations found necessary for ACT will probably initially take the form of special conditions for certification.

In the following paragraphs some of the important design criteria and regulatory problems affecting the implementation of ACT are discussed.

RELIABILITY – SAFETY

The reaction of most designers when faced with consideration of ACT is to raise the question of reliability and safety. The benefits and ultimate success of ACT depend strongly on the reliability of ACT and associated systems. The extent to which inherent characteristics are augmented depends on the level of degradation to be considered in the design. If it is decided to consider a complete loss of ACT functions, then adequate inherent characteristics must be provided, thus limiting the benefits obtainable from ACT. Further, reliability analysis is mistrusted and often misunderstood. Reliability is simply a probability of occurrence of a certain event within a specified interval. The accepted approach to this problem is to define undesired events, such as loss of an aircraft, and calculate their probability of occurrence within a meaningful period, such as a flight of some duration. A maximum value of this probability can be set as a design goal, with the value in proportion to the seriousness of the event. This approach is in fact taken in the design of primary structure, though the probability is not generally calculated.

It is apparent that safety must not be compromised, and that the probability criteria for catastrophic failure will not be compromised. Significant sudden structural failures are very rare. In most cases, structural damage is a result of fatigue and can be detected before a hazardous level of degradation is reached. The impression persists, though, that this is not the case with some other aircraft systems. Failures or malfunctions occur in autopilot, electrical, and hydraulic systems with little or no warning. The required level of overall function reliability is achieved in control and vital power systems by increasing redundancy for those functions that do not have the required reliability. For example, controllability of modern jet aircraft is dependent on the integrity of hydraulically powered controls. Reliability for safety of flight is provided by multiple hydraulic systems. After some number of failures, it is, of course, advisable to terminate the flight at the nearest suitable airport in order to minimize exposure time in a nonredundant configuration.

One difference, however, is that failures of presently utilized active control functions do not usually result in reductions in structural capability under normal flight conditions, whereas proposed ACT functions will, in effect, replace primary structure. This does not necessarily mean that these functions must be as reliable as the basic structure, however. An assessment of situation severity and a list of means available for reducing risks presented by failures in ACT functions is given in Table 2. There are three principal means of controlling the risk:

- Control system redundancy
- Actuation and/or surface authority distribution
- Reduced operating envelope.

The ultimate levels of functional reliability will be required only for those functions upon which safe termination of the flight depends. Category IIIa autoland systems are presently achieving this reliability, but for only a short exposure period during each flight. This short time allows the objectives to be achieved using only two autopilot systems. Figure 6 shows the required single system MTBF as a function of the number of systems required to achieve a probability of complete failure of not more than 1×10^{-9} during a 3-hour flight.

The problems with reliability are likely to occur within the sensing, computing, and display functions which are today largely restricted to flight guidance and control systems (FGCS). Typical MTBF values for functional failures of these systems are on the order of 300 to 800 hours. These numbers are expected to improve with the usage of digital elements in the systems. Although Figure 6 shows that the overall reliability requirement may be satisfied with a not unreasonable number of redundant systems, the use of three systems would require a ten-fold increase in reliability. Characteristic systems for this application will include multiple-channel command paths in which failures will be annunciated, thus providing the pilot with system degradation information enabling him to take corrective action prior to total system failure. Ultimately, however, improved reliability goals and techniques must be derived and imposed, but must always include a sensible system failure mode and annunciation capability.

An associated problem is an apparent FAA requirement for determining that safety-related systems are functioning prior to dispatch. Difficulties in determining sensor status have prevented taking credit for automatic cutoffs (ACO) in limiting the consequences of autopilot hardover failures, in some cases. This may require design of systems which can be satisfactorily checked on the ground.

Safety is presently established in a manner whereby critical functional elements of the system can be specifically identified in a reliability block diagram and the reliability of each element is available. The computation of the reliability of the avionics elements which contribute to the flight safety of a control configured vehicle will be significantly more complex. Not only are there many more elements, but the software is an additional facet which must be evaluated. Accomplishing the failure and probability analyses of these complex systems is a major task in itself. In some cases, failure analyses have been required to prove that certain types of failures were extremely improbable, which in itself may be a nearly impossible task.

It is extremely doubtful that first generation ACT applications on transport aircraft will be flight critical. Until further experience with new ACT modes is obtained, allowance will be made for failure of these modes and adequate inherent safety provided. This will reduce the severity of the reliability criteria for first generation systems.

RELIABILITY — ECONOMICS

The economics referred to here is that of dispatch reliability, not maintenance costs, although the latter are certainly important.

A typical design goal for dispatch reliability is that, mechanically, the aircraft shall be capable of departure within 15 minutes of the scheduled time 99 percent of the time. This goal is very stringent and is currently being achieved consistently by only one transport aircraft, the DC-9. The design of this aircraft emphasized simplicity and reliability, whereas the design of later aircraft has emphasized performance, with a resulting increased complexity.

This dispatch goal produces a desire to have your cake and eat it, too. The benefits of more complex systems are desired but it is also desirable to allow dispatch with as many things as possible inoperative or missing. It is common to find flight manuals and minimum equipment lists filled with information for covers, doors, and fairings missing, or for hydraulic pumps, yaw dampers, Mach trim systems, autopilots, antiskid, and thrust reversers inoperative. The benefits to be obtained from, and therefore dependency on, some systems are limited by the criteria for inoperative dispatch. In some cases, the ACT system redundancy requirements dictated by safety considerations may severely penalize dispatch rates unless further redundancy is added for dispatch inoperative allowance. This will, in turn, increase the total system acquisition and maintenance costs.

The goal of 1-percent delay rate is typically allocated among the various aircraft systems as shown in Figure 7. The pilot controls and FGCS are allotted 0.005 and 0.10 percent, respectively. The small size of these percentages does allow some increase without having a major impact on delay rate, but the accompanying impact on maintenance and spares availability may be significant. As ACT systems are introduced that are dispatch critical, continued improvement of built-in-test-equipment (BITE) and trouble-shooting means will be required to quickly pinpoint and correct faulty system elements.

FLYING QUALITIES

Design criteria for flight characteristics, or flying qualities, seem to be in good shape, judging by pilot acceptance of the wide body jet transports. However, it should be realized that these criteria were derived and applied ten years ago. While most of the basic criteria were expressed in terms of conventional open-loop characteristics, many can be applied to ACT aircraft. Much more work has been done on pilot-model-in-the-loop criteria, which generally do not require such conventional modes of control and response.

The FAA regulations concentrate on classical stability characteristics, primarily static, and on steady-state control requirements. For example, the regulations require that aircraft have positive static longitudinal stability. In practice, static longitudinal stability is demonstrated by showing that a pull force is required to maintain air speeds below trim speed and that a push force is required for speeds above trim speed. The intent of the rule is to insure flight safety by reducing pilot workload and fatigue, and by causing the aircraft to return to trim after inadvertent inputs or external disturbances. At the same time, modern aircraft are equipped with pitch attitude/rate control modes, mechanized through the autopilot, which do not have the property of static stability.

While some would argue that static stability, like many other classical criteria, is out of place in the modern world of ACT, it can be shown that the application of augmentation without concern for the properties intended by classical criteria can produce undesirable characteristics. Consider, for example, an inherently stable airplane with the augmented control modes referred to as control wheel steering and autothrottles. These modes can be switched off, in which case the aircraft reverts to an unaugmented mode of control. The response of this aircraft to a pilot-induced upset is shown in Figures 8a, 8b, and 8c. In Figure 8a, both systems are off. The aircraft has both static and flight path stability, as shown by its tendency to return to trim airspeed and flight path after the upset. With the autothrottle on but CWS off, the flight path and pitch attitude diverge following the upset, as shown in Figure 8b. When the CWS system is turned on, as shown in Figure 8c, the divergence in attitude (an instability) is changed to neutral stability, in that a new trim state is obtained. It can be seen that a system which is a great help to the pilot in actively managing the aircraft can become a hindrance to controlling the airplane in periods of inattention.

It should now be apparent that a new problem is introduced if the designer is unconstrained by tradition. He must decide which states should be controlled by which of the controllers available to the pilot. Traditionally, the stick has been geared to the elevator and the throttle lever to the powerplant. In the discussion above,the stick controlled pitch rate and the throttle lever controlled airspeed. Considering only the kinematics of the airplane, pitch seems an odd choice of a state to control, as only the projection of the gravity vector depends on pitch attitude. Angle of attack would seem a much better choice. The virtue of pitch attitude is that it is easily measured. Further, it is generally a reasonable approximation to angle of attack, which is difficult to measure directly. However, in the presence of a system which maintains constant speed, pitch attitude is not a good approximation to angle of attack, though it is a very good low frequency approximation to flight path angle. It seems equally odd to link throttles to airspeed, when virtually any student of aeronautics can quote the formula for rate of climb.

After 75 years of manned aircraft flight, the argument over the proper modes of longitudinal control still rages. It appears that more thought and research is needed to permit a rational selection of the number of controls to be made available to the pilot and what states are to be controlled. A possibly promising control mode is a blend of angle of attack, pitch attitude, and pitch rate command. This is essentially a flight path command system, which should work well in combination with an airspeed command system (autothrottles).

This freedom in specifying modes of control and response will obviously cause difficulties in applying many of the conventional criteria. In many cases, however, the criteria can still retain some utility when applied to ACT systems. The intent of the criterion must be determined in its original context, and the intent applied in the new context. There is a further problem for the flying qualities specialist in specifying criteria before the nature of the control augmentation has been specified. This means that he must formulate criteria which are completely general, a seemingly hopeless task.

A better approach is for the handling qualities and controls specialists to work interactively in designing a control system which provides good flying qualities. If the controls analyst uses optimal control theory, the flying qualities specialist can provide an explicit statement of his requirements in the form of a model. This model can then be used in explicit or implicit model following design to produce a control system which provides good flying qualities. The flying qualities specialist should then, using whatever criteria he has, analyze the augmented aircraft to verify that it does, indeed, have the desired flying qualities. If problems occur, it may be necessary to change the model or the cost function in the optimal control algorithm. The model should be made as nearly like the unaugmented airplane as possible, with only those changes needed to achieve the desired flying qualities. This should result in a minimum number and size of the controller gains. It can easily be seen that there will be substantial difficulties in the model following design if, for example, the model and airplane differ greatly in load factor response (n_z/α) and there is no direct lift control surface.

Flying qualities criteria must be defined not only for the normal operating conditions, but also for failure modes. There will be some degree of degradation following component failures affecting the control system, which may or may not be acceptable. Whether they are acceptable depends on how frequently they occur and the magnitude of the degradation. For example, a failure which would cauls the loss of the aircraft is no more acceptable than a catastrophic structural failure. A probability often associated with this frequency is 1×10^{-9} per flight, which corresponds roughly to never. The MIL-F-8785B has a good approach to this problem: it specifies three allowable levels of flying qualities as a function of liklihood of occurrence. The MIL-F-8785B probabilities, which are given in Table 3, are much too high for civil transport applications. Suggested probabilities for civil transport applications are given, reflecting a much more conservative bias. These suggested numbers have no concrete foundation, except for the requirement that catastrophic failures occur no more often than once in 10^9 flights.

STRUCTURES

The major impact of active control technology will be on the determination of the external load levels and stiffnesses for which the structure is to be designed. Active controls will change the magnitude, distribution, and frequency of occurrence of the loads for which the structure is designed. Other areas affected include the selection of critical conditions, calculation of external loads, both steady state and dynamic, and flutter characteristics. However, it is likely that present structural design criteria can be applied largely unchanged to the next generation of aircraft. Most of the present maneuvers, load factors, gusts, safety factors and flutter margins will be satisfied with the active systems operating.

One area which will be impacted by active control technology is the methodology required for external loads and flutter analyses. State of the art methods will take us a long way, but as dependency on active systems increases, more exact solutions will be necessary. Since effective structural stiffness and damping will be augmented by the active control system responses, techniques which account for the complete dynamic transient response of distributed loads as well as the total body will likely be required in both the frequency and time domains. Since the design loads are normally obtained at or near the extremes of the aircraft flight envelope, it may be necessary to account for certain nonlinear properties in these analyses. Advances in the field of unsteady transonic aerodynamics may then be needed for efficient system design.

The area of aircraft structural fatigue life prediction will also be impacted by active controls. Because the application of active controls provides a more efficient distribution of external loads at the aircraft design conditions but does not change the total load in steady 1g flight, a change in the historical relationship of structural design level stresses to 1g stresses will result. Also, depending on the ranges of response of the active controls, i.e., dead bands, etc., the in-flight fatigue damage cycles and peak-to-peak stress relationships will be altered. Therefore, future fatigue evaluations will assume greater importance, necessitating increased use of mission analysis to determine the effects of active controls on the resulting fatigue spectrum, as mentioned previously.

Elsewhere in this paper, it is suggested that the basic maneuver requirements remain unchanged, although the design conditions might be reduced through the use of envelope limiting which would alter criteria or requirements for margins between the operating and design envelopes. Total maneuvering loads would therefore be relatively unchanged, with active controls providing only an optimized redistribution of this load. In contrast, active control systems can alter the magnitude as well as the distribution of gust-induced loads. The magnitude of the resultant load and stress reductions must be balanced against the desired flying qualities and ride control functions of an active control system to achieve an overall practical optimum design.

New design criteria must be developed for ACT system degraded/inoperative conditions, if these conditions are allowed. System failure modes and effects must be determined, and their probability of occurrence calculated. Special criteria and conditions which account for these failures in a logical and rational manner must be established. The primary consideration in the creation of new active controls design criteria should be to at least maintain the present level of safety and integrity of the aircraft. Therefore, it is extremely important that all proposed new criteria be balanced against present criteria on a probability basis. For example, if an active control system could be shown to have the same level of integrity as structural components, the present structural fail-safe load criteria could logically be applied to the system. Obviously, the establishment of a total fail-safe criterion for these new systems will not be this simple. A high level of interchange and cooperation between the air transport industry and government research and certification agencies will be required if realistic and effective design criteria are to be obtained. This interchange has already begun with the establishment of special criteria for present load alleviation controls such as yaw dampers and control surface limiting devices and is continuing in the current investigations of wing load alleviation.

The increased dependence on load alleviation systems places increased importance on these criteria. It is not clear at this time what changes, if any, should be made in the structural design criteria for degraded active control system performance in system failure modes.

CONTROL SYSTEMS

The criteria for detail design of conventional control systems are predominantly developed by the manufacturers. These include instructions regarding design to provide safety, ease of maintenance, and to prevent incorrect assembly, for example. The implementation of active controls will necessitate the expansion of these rules to include much more sophisticated applications.

One area receiving considerable attention is that of establishing a math model of the airframe and deriving design criteria for establishing parameter perturbation analyses on the model. This is an area that has received considerable attention in missile and launch vehicle control system design. Unsteady aerodynamics and structural dynamics are among the principal problem areas. The accuracy of existing prediction methods is doubtful for optimum ACT system design. Recent studies are quantifying the likely magnitude of prediction errors (e.g., ref. 20) and more work will need to be done before full confidence can be placed in a new vehicle with a "flight critical" control system before first flight. A related problem is the variation in structural dynamic and aerodynamic parameters due to changes or differences in fuel and payload distribution that may occur during one flight as well as between flights, along with the variation of airspeed, altitude, and Mach number encountered. The insensitive flight control system approach may prove to be the best way to handle this variation in parameters.

The active control system will also be much more demanding on control system components which are subject to wear. Because of the higher gains required by the active control system, control system components will have to meet tighter specifications, and remain within these specifications throughout the useful life of the control system. This may require new design criteria for components such as hydraulic valves and actuators whose phase and gain characteristics are affected by wear. It will also require tighter tolerances on control system bearings in order to prevent low amplitude, fatigue causing, limit cycle oscillations. At the same time, the automatic controllers must handle out-of-tolerance conditions. These conditions can occur due to manufacturing tolerances, aging, wear, material failures, off-nominal power supplies, and dynamic characteristics caused by changes in environmental conditions. It is expected that digital implementation will improve the end-to-end tolerance problem significantly.

As noted earlier, transport aircraft design has been evolutionary rather than revolutionary and this trend is expected to continue as active control concepts are implemented. It is unlikely, for example, that the manufacturers will commit to fully flight critical active control systems without going through a proving period on similar systems which are not flight critical, i.e., a complete system failure will not create an unmanageable situation from the flight crew's point of view. Some experience has been accumulated in the lateral-directional axes (yaw-damper) and the longitudinal axis (Mach trim) with favorable results. Systems which allow the designer to further implement active control functions will need to operate similarly, i.e., they must exhibit excellent availability, be free of nuisance failures and interruptions, operate with little or no attention from the crew, require no more ground and preflight testing than today's yaw damper, and system elements must fail obviously and passively.

As such flight control systems become more complex, built-in test equipment (BITE) takes on greater importance as a means for improving safety, operational reliability, and maintenance costs. The design requirements for built-in test equipment must include not only static end-to-end checks of the control system but dynamic checks as well. The BITE requirements should include the capability for these status and performance checks by continuous on-line tests, inflight pre-engage operational status tests, channel comparison monitoring, and ground maintenance tests. The inflight tests must be capable of detecting failures to the functional system level. The ground checks resulting from inflight failures must isolate these failures to the line replaceable unit (LRU) level. The complexity of the systems as compared with the level of capability of average maintenance personnel will require very stringent design requirements to preclude faulty maintenance and provide ease of fault isolation and correction. It is important to note that the background of missile control system experience will do little to help formulate design criteria associated with many hours of continuous operation.

As one of the special conditions in the transport certification procedure, it is specified that the airplane will operate safely for at least 5 minutes with the primary electrical system inoperative. The addition of ACT functions will add to the electrical load which must be supplied under emergency conditions. Adequate and reliable emergency power sources will be required for the functions which are critical for continued flight, and compatibility of power supply and switching transients must be provided.

CONCLUSIONS

It is clear that a great deal of work remains to be done in the area of detail design criteria and design practice. It is also apparent that the overall improvement that can be achieved by going to active controls is, with but a few exceptions, not being held back by current regulations and basic design criteria.

The area where the most work needs to be done is in the detail design criteria of the control system itself. The problems center around the derivation of reasonable criteria for the design of advanced flight controllers. Other problems are the achievement of the reliability goals and production of hardware which can be maintained and manufactured at costs comparable to the rest of the aircraft critical components. The determination of the relative magnitudes of inherent vs augmented qualities may then be the result of minimum operating cost trade studies.

As this work progresses, more ACT functions will be proven to be both reliable and practical, and will be incorporated into advanced aircraft designs.

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TABLE 1

ACT FUNCTION APPLICATION EXPERIENCE

ACT FUNCTION	AIRCRAFT	STATE OF READINESS	PAYOFF & TRADE DATA	SYSTEM MECHANIZED	FLIGHT TESTED	OPERATIONAL EXPERIENCE
Relaxed Inherent Stability Augmentation	Military	→	→	→	→	
	Experimental	→	→	→		
Center of Gravity Control	Military	→	→	→	→	
	Commercial	→	→	→	→	
Ride Quality Control	Military	→	→	→		
Yaw Damper	Military	→	→	→	→	
	Commercial Transport	→	→	→	→	
Maneuver Load Control	Military	→	→	→		
Gust Load Control	Military	→	→	→		
	Commercial Transport	→	→	→	→	
Fatigue Damage Control	Military	→	→	→	→	
Flutter Control	Military	→	→	→		
Envelope Limiting	Military	→	→	→	→	
	Commercial Transport	→	→	→	→	

TABLE 2

DEGRADED SITUATION SEVERITY AND MEANS AVAILABLE FOR MODIFYING RISKS PRESENTED BY FAILURES

FUNCTION		SEVERITY OF SITUATION WITH FUNCTION DEGRADATION	MEANS AVAILABLE FOR MODIFYING RISKS PRESENTED BY FAILURES
Relaxed Inherent Stability Augmentation		Moderate-Very	Redundancy + Authority distribution Reduced operating envelope CG management
Load Control	Maneuver	Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
	Gust	Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
	Fatigue Damage	Negligible	Reduced operating envelope
Flutter Control		Very-Extreme	Redundancy + Authority distribution Reduced operating envelope
Ride Quality Control		Negligible-Moderate	Redundancy + Authority distribution Reduced operating envelope
Envelope Limiting		Negligible-Moderate	Redundancy Reduced operating envelope
CG Control		Negligible	Reduced operating envelope

TABLE 3

LEVELS FOR FAILURE STATES

Level	Probability of Encountering, per flight: MIL-F-8785B	Suggested
1	—	—
2	$<10^{-2}$	$<10^{-5}$
3	$<10^{-4}$	$<10^{-7}$
worse than 3	—	$<10^{-9}$

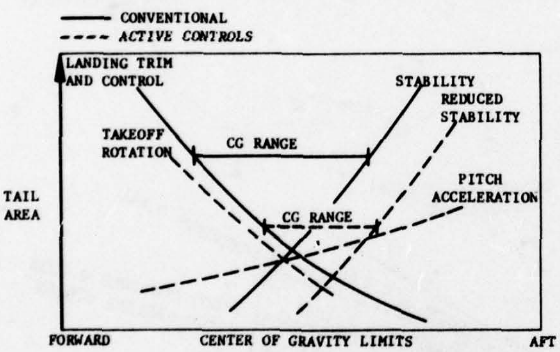

FIGURE 1. HORIZONTAL TAIL AREA REQUIREMENTS

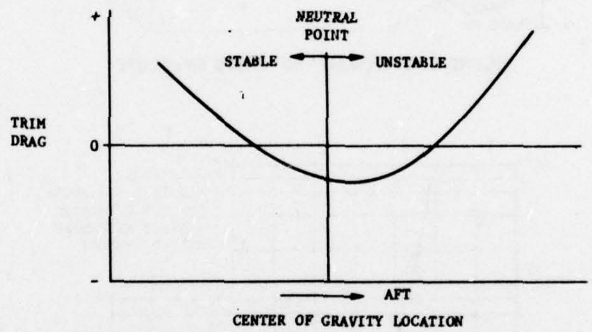

FIGURE 2. TRIM DRAG

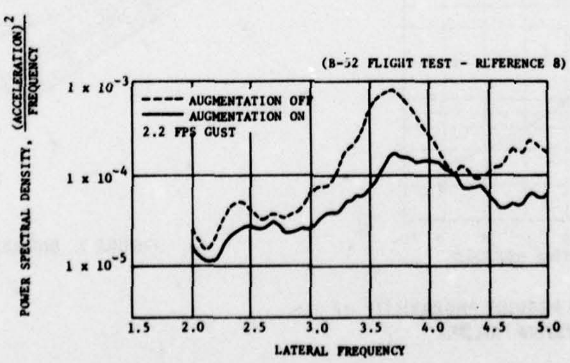

FIGURE 3. RIDE QUALITY IMPROVEMENT

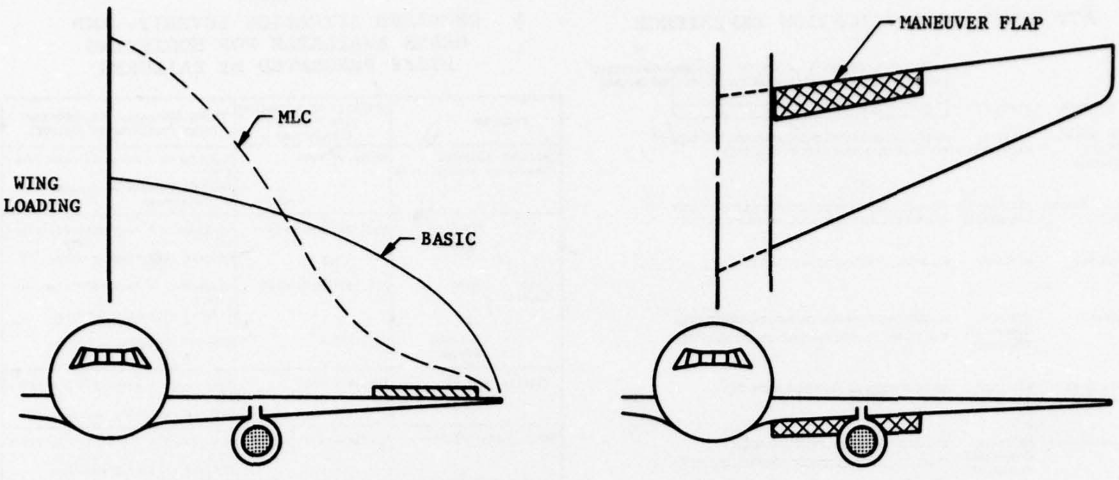

FIGURE 4. MANEUVER LOAD CONTROL

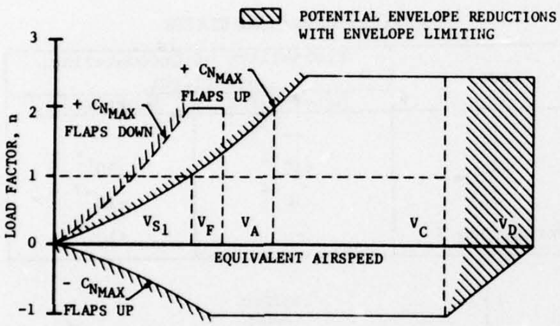

FIGURE 5. DESIGN MANEUVERING ENVELOPE

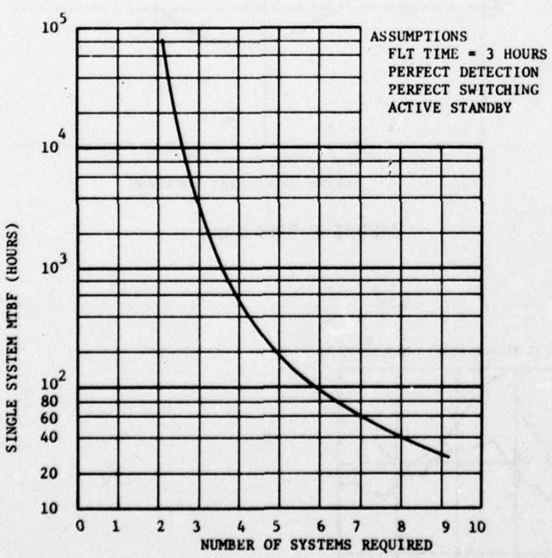

FIGURE 6. SYSTEMS REQUIRED TO PROVIDE PROBABILITY OF 10^{-9} FOR COMPLETE SYSTEM FAILURE

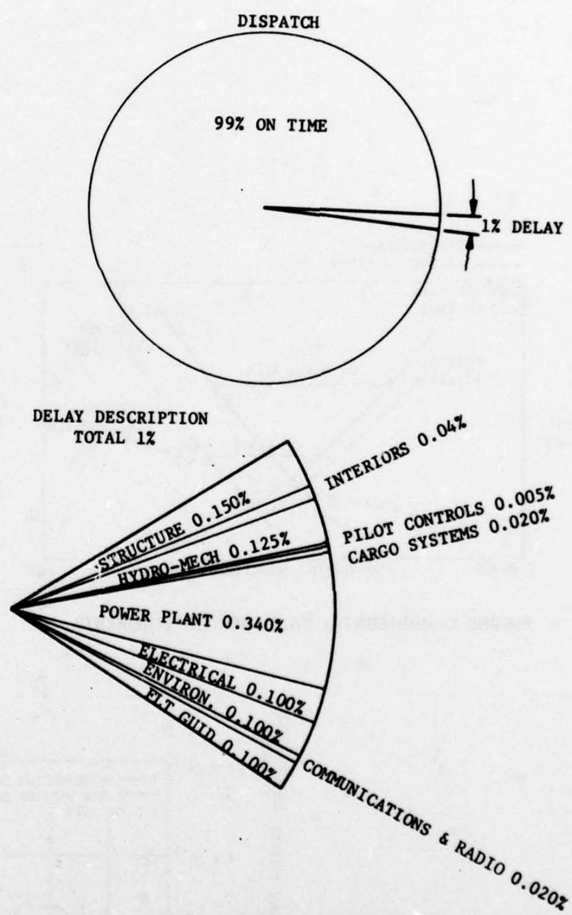

FIGURE 7. DISPATCH RELIABILITY

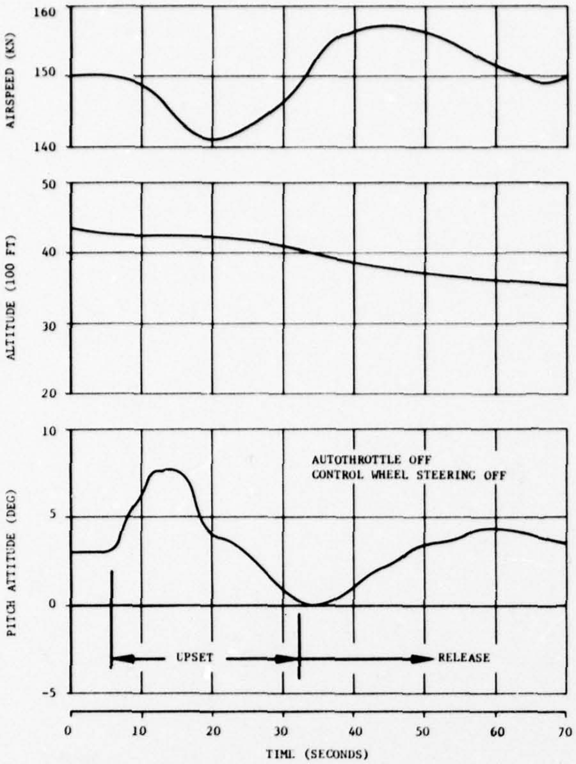

FIGURE 8a. APPROACH LONGITUDINAL STABILITY

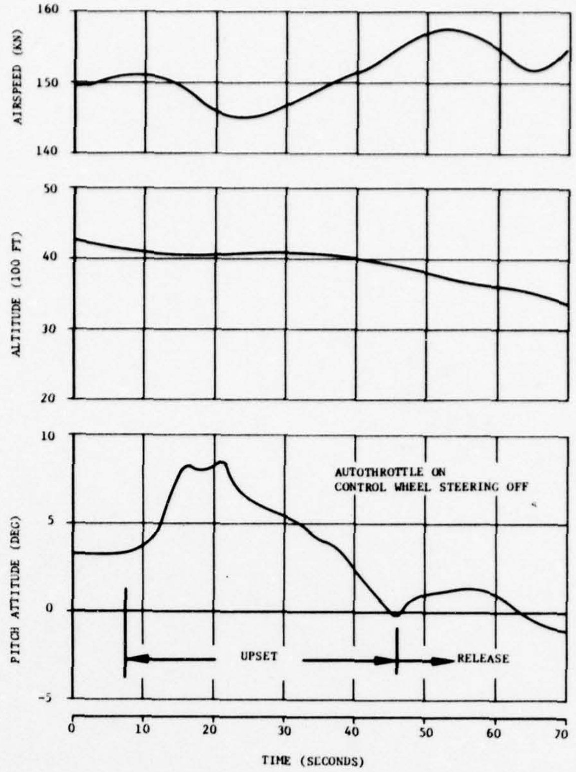

FIGURE 8b. APPROACH LONGITUDINAL STABILITY

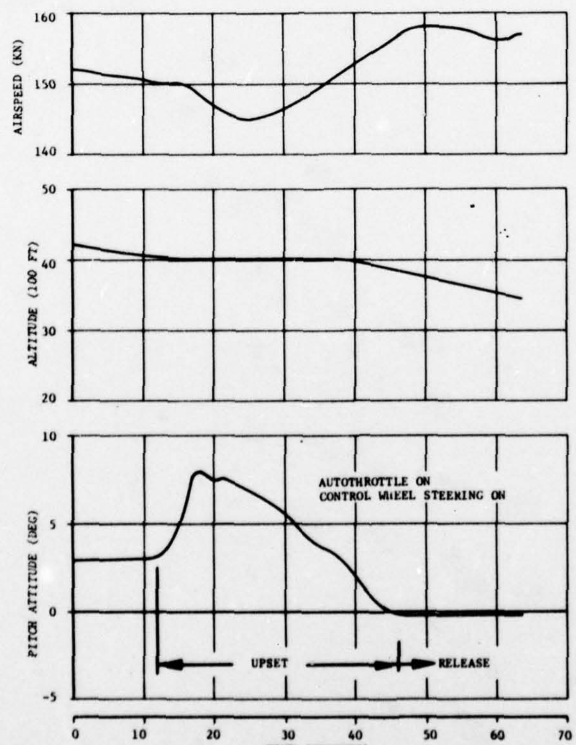

FIGURE 8c. APPROACH LONGITUDINAL STABILITY

PART II

FIGHTER APPLICATIONS

CONTROL-CONFIGURED COMBAT AIRCRAFT

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SUMMARY

The effects of Active Controls Technology on combat aircraft, in terms of weight reduction achieved, performance and handling improvements are reviewed. It is shown that very significant improvements in performance can be achieved with artificial longitudinal stability coupled with automatic operation of combat flaps. The adoption of spin prevention and automatic manoeuvre limitation will give carefree manoeuvring. The combination of the performance and handling improvements will lead to greatly increased operational capability.

The engineering features of a full-time fly-by-wire system to achieve these ends are discussed briefly.

LIST OF SYMBOLS AND ABBREVIATIONS

A.C.T.	active controls technology
A.T.R.	attained turn rate (lift limited)
A.D.D.	airstream direction detector
$\bar{c}$	reference mean chord
C.A.P.	combat air patrol
C.C.P.	combat correlation parameter
c.g.	centre of gravity
C_{l_β}	rolling moment due to sideslip derivative
C_m	pitching moment coefficient
C_{n_β}	yawing moment due to sideslip derivative
F.B.W.	fly-by-wire
H	pitot pressure
Ixx	moment of inertia in roll
Izz	moment of inertia in yaw
L.C.D.P.	lateral control departure parameter
kg	kilograms
L/D	lift/drag ratio
M	Mach number
m	metres
M.L.C.	manoeuvre load control
n	normal acceleration factor
n_{z_α}	normal acceleration per unit incidence
p	rate of roll
P	static pressure
q	dynamic pressure
$\dot{q}$	angular acceleration in pitch
r	rate of yaw
S	wing reference area
S.D.S.P.	stall departure and spin prevention
S.E.P.	specific excess power
S.T.R.	sustained turn rate
U.K.	United Kingdom
W	weight

1. INTRODUCTION

Theoretically, the adoption of Active Controls Technology may be used either:-

. to reduce aircraft weight, maintaining a specified performance

or

. to improve performance, with a defined power plant

In practice the benefits are usually achieved in terms of both weight reduction and performance gains, but are realised in full only if ACT is adopted at the inception of the design and used to influence the aerodynamic, structural and systems configurations to achieve maximum efficiency. This is what is meant by a control-configured aircraft or, more specifically:-

A control-configured aircraft is one which utilises the aerodynamic forces and moments produced by its movable surfaces, activated by commands from aircraft motion sensors, to augment, alleviate or redistribute the total aerodynamic forces in such a way as to improve aerodynamic, structural or operational efficiency.

In this Chapter, project studies made by British Aircraft Corporation (Military Aircraft Division) of the effects of ACT on strike and combat aircraft are reviewed. The results are illustrated in terms of:-

- weight reduction achieved
- aerodynamic efficiency (e.g. maximum lift and lift/drag ratio)
- performance (sustained manoeuvrability, fuel economy, take-off and landing distances)
- combat score

These are the tangible gains, which can be expressed in numerical terms. Also discussed are the effects of:-

- improved handling qualities
- reduced pilot work load
- removal of flight restrictions

which lead to "carefree manoeuvring". The effects of these are much more difficult to translate into numerical values, but they are no less important.

The benefits listed above are due to the adoption of:

- artificial longitudinal stability
- automatic configuration management
- stall departure and spin prevention
- overstressing prevention

The effects of manoeuvre load control, active flutter control and of gust alleviation on combat aircraft are also considered briefly.

The engineering of a full-time fly-by-wire control system is discussed briefly, the preferred configuration reflecting the outcome of recent BAC studies.

2. ARTIFICIAL LONGITUDINAL STABILITY

2.1 Application to Reduced Aircraft Size and Weight

Traditionally the tailplane is sized and the c.g. range located on a combat aircraft by the following criteria:-

- an adequate stability margin at aft c.g.
- ability to trim maximum usable lift at forward c.g.
- ability to lift the nosewheel on take-off
- ability to trim in manoeuvring flight at high supersonic speed/high altitude

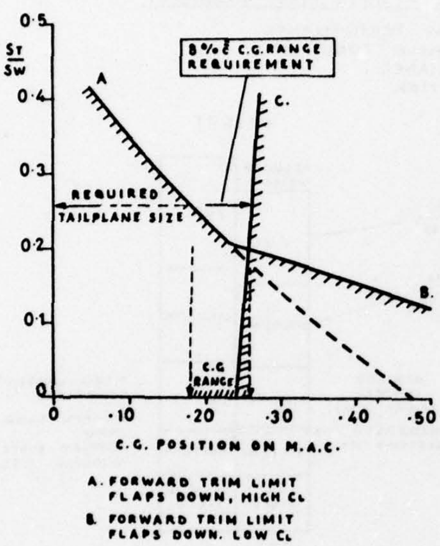
A typical tailplane sizing diagram of a conventional supersonic strike aircraft is illustrated in figure 1a. The rate of exchange between the aft c.g. limit and tailplane area is proportional to the tailplane lift slope, diminished in proportion to the downwash gradient in which it is situated and reduced by aeroelastic losses. Minimum longitudinal stability usually occurs at high subsonic speed on conventional (aft tail) configurations because of increasingly adverse downwash and aeroelastic effects with increasing subsonic Mach number; with a low tailplane underwing stores cause additional degradations due to increased downwash and reduced dynamic pressure in their wake. The rate of exchange between the aft c.g. limit and tailplane size is therefore small, tailplane effectiveness as a stabiliser being possibly as low as 10% of its rigid, free air value because of these adverse effects. Although large tailplane loads are generated, only a small proportion of the tailplane lift capability is used at aft c.g.

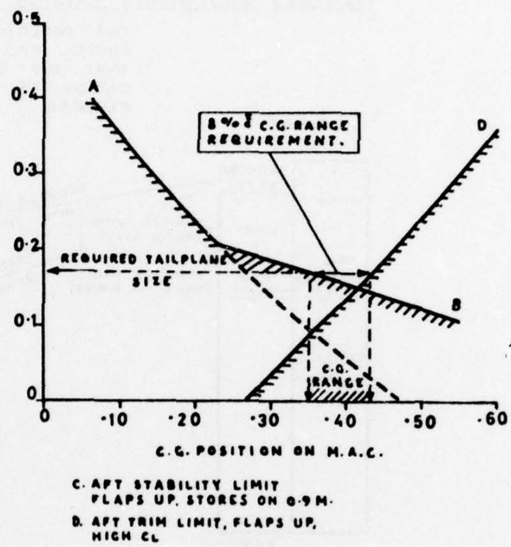
When the constraint of natural stability is removed, the full lift capability of the tailplane may be used to trim the unstable wing-body moment at aft c.g., leading to the revised tailplane sizing diagram shown in figure 1B. The design case at aft c.g. is now determined by the balance of pitching moments. The maximum lift of the tailplane, generating a nose-down moment, must be capable of overriding the maximum nose-up wing-body pitching moment within the usable range of incidence at any Mach number. Obviously an exact balance of pitching moments at the design point leaves no margin. Some residual pitch-down capability must be provided for initiating recovery from high incidence and for resisting roll-yaw inertial coupling moments i.e.

$$\Delta C_m = \frac{(I_{zz} - I_{xx})\ rp}{q\ S\bar{c}}$$

The numerical values are defined by experience as the onset of severe departure at minimum combat speed.

TAILPLANE SIZING





Comparison of figures 1a and 1b shows that artificial longitudinal stability allows 35% reduction in tailplane area and is associated with a rearward c.g. shift of 17%$\bar{c}$ in the case illustrated. The same total c.g. range of 8%$\bar{c}$ is retained.

Rearward location of the c.g. range results in a favourable variation of tail load to trim. In the example case, as illustrated in figure 2, trimmed lift is increased by 12%, both with flaps up and with flaps down. Whether the wing is sized for combat manoeuvrability or for airfield performance, this increased trimmed lift allows wing loading to be increased pro-rata. A reduction of more than 12% in wing area is in fact possible because weight can be reduced by taking advantage of the reduced drag, leading to reduced fuel requirements and reduced engine size. The resulting drag and weight savings are illustrated in bar-chart form in figure 3. In this example case the design mission was a low-low strike mission at 0.7M. Take-off distance, radius of action and weapon load were held constant. Savings of 9% in drag and weight were demonstrated. Additional benefits in manoeuvre performance would have resulted also from the reduced trimmed induced drag associated with rearward c.g. location. A planform comparison of the conventional design and the CCV variant is shown in figure 4.

FIG 2 EFFECT OF C.G. POSITION ON MAXIMUM LIFT

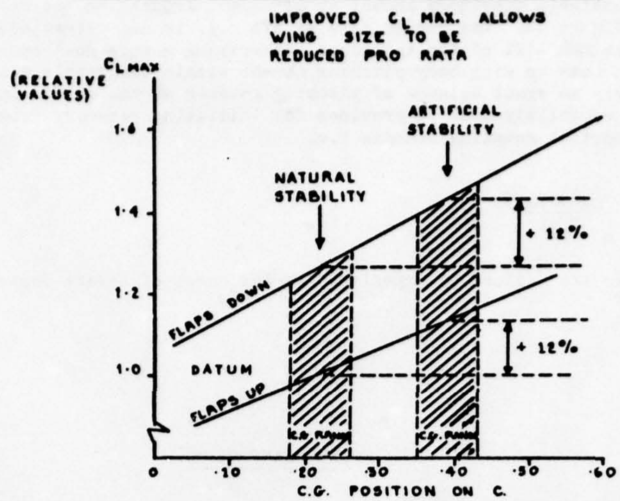


FIG 3 DRAG AND WEIGHT SAVINGS WITH ARTIFICIAL STABILITY.

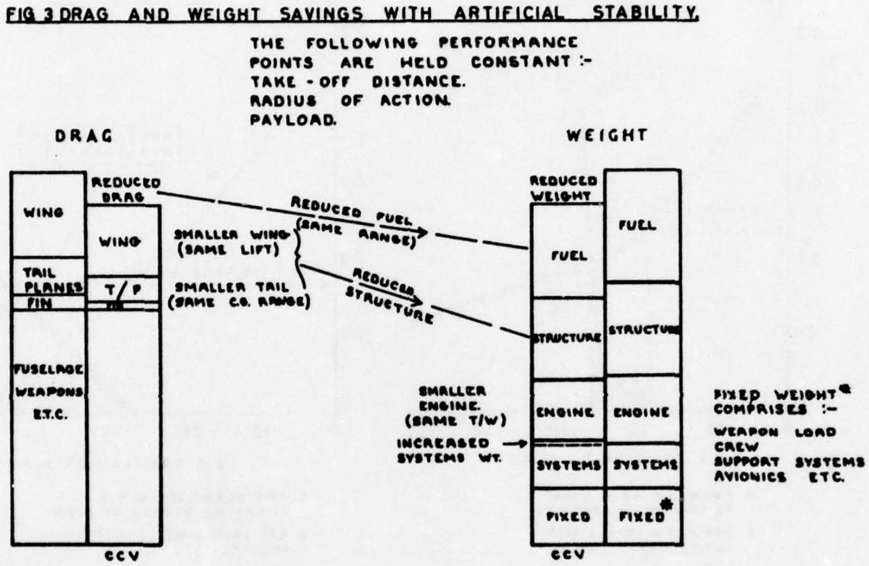
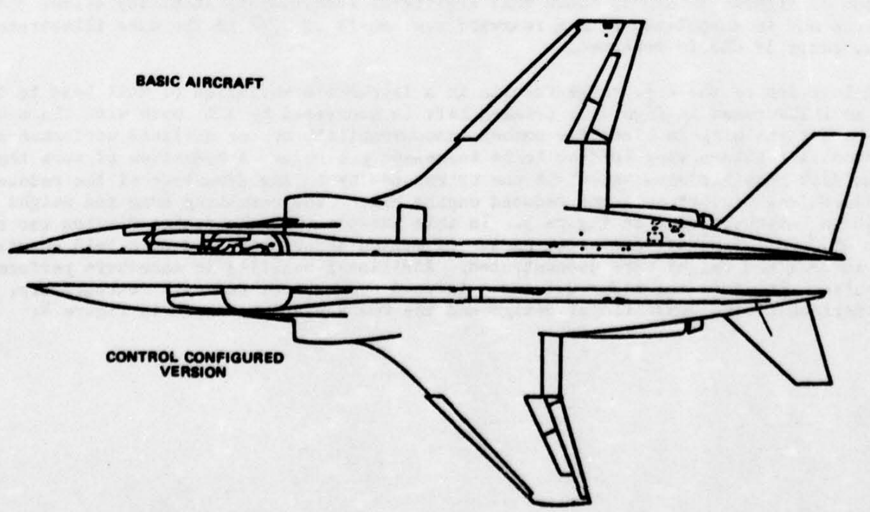


FIG 4 EFFECT OF ARTIFICIAL STABILITY ON AIRCRAFT SIZE



It is interesting to note that this figure of 9% weight saving agrees very closely with the figure quoted in reference 1 for a similar exercise based on the Phantom. However, this is coincidental because the weight saving is strongly dependent on the fixed weight fraction and on the design mission. The smaller the fixed weight fraction, the larger are the structural, fuel and power plant contributions which are all susceptible to scaling down by the adoption of ACT. Again, in the example case there was no direct benefit (fuel saving) resulting from reducing induced drag, but only from profile drag. In an air superiority mission where large proportions of the fuel would be expended in loiter and in sustained manoeuvring, there would be a premium on reducing induced drag. Finally, it should be noted that a significant weight saving can be realised only if the engine is in the formative design stage and can be scaled to suit the airframe. In designing around a fixed engine the weight saving is much smaller, typically 3%, but of course performance is enhanced.

The weight saving depends strongly also on the datum configuration. The figures of 35% reduction in tailplane size and 12% improvement in trimmed lift apply to a fixed wing design without combat flaps with a conventional tailplane on an arm of $1.5\bar{x}c$.

The results of limited studies on two other configurations are shown in figures 5 to 7. Figure 5 illustrates the variation of trimmed lift and lift/drag ratio with c.g. position and incidence, subsonically, for a tailless delta configuration with similar aerodynamic characteristics to the Mirage III or F-106. With artificial longitudinal stability and blown elevators, lift for take-off and landing ($20^\circ\alpha$) is increased by 30%; at a given total C_L of 0.8, lift/drag ratio is increased by 50%. Alternatively, as depicted in figure 6, wing size could be reduced by 22%, maintaining the same total lift and c.g. range.

The variation of trimmed maximum lift with c.g. position and wing sweep for a variable sweep strike/combat aeroplane is shown in figure 7. Retaining the tail sized for natural stability, trimmed lift could be increased by 30% by rearward c.g. location and reduced wing sweep. Alternatively the tailplane could be re-sized (reduced by 33%) and trimmed lift increased by 11%.

The cases illustrated in figures 5 to 7 were not used directly for project design purposes, so no figures for the resulting weight savings were available. Nevertheless they serve to illustrate the effects of artificial longitudinal stability on alternative configurations.

60° DELTA

FIG 5 VARIATION OF TRIMMED LIFT/DRAG RATIO WITH C_L (α) AND C.G. POSITION [SUBSONIC]

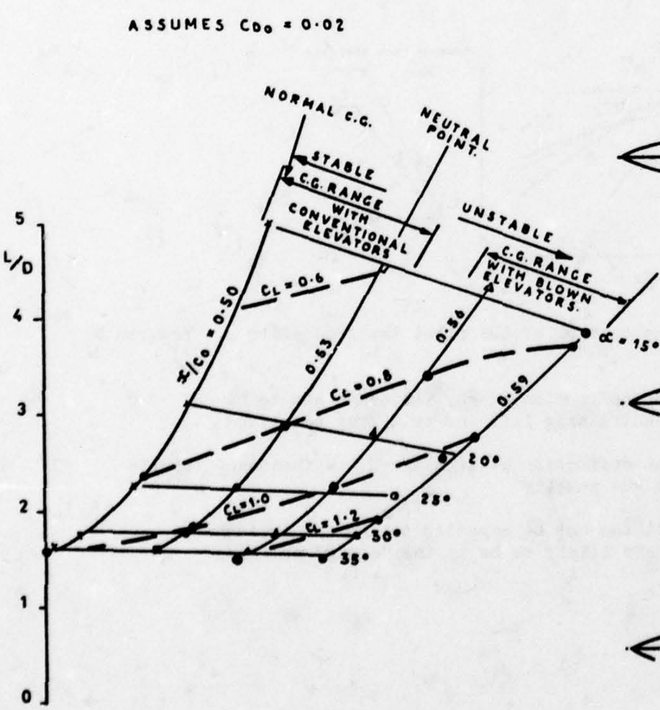


FIG. 6 EFFECT OF ARTIFICIAL STABILITY ON WING SIZE 60° DELTA

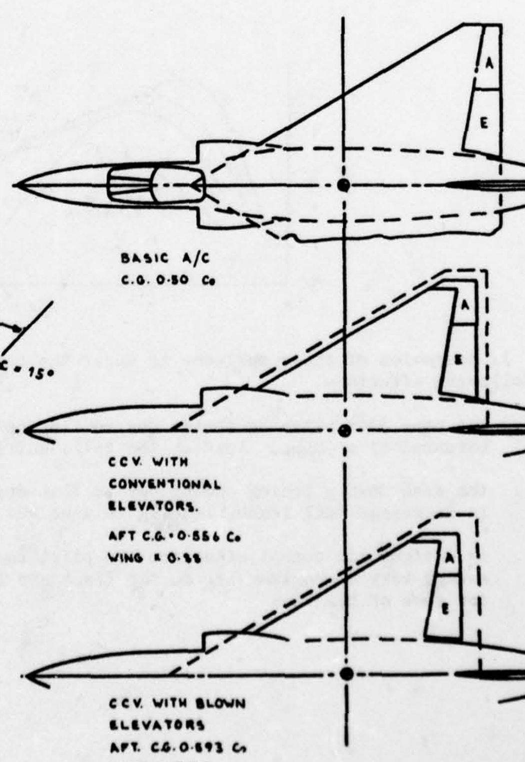
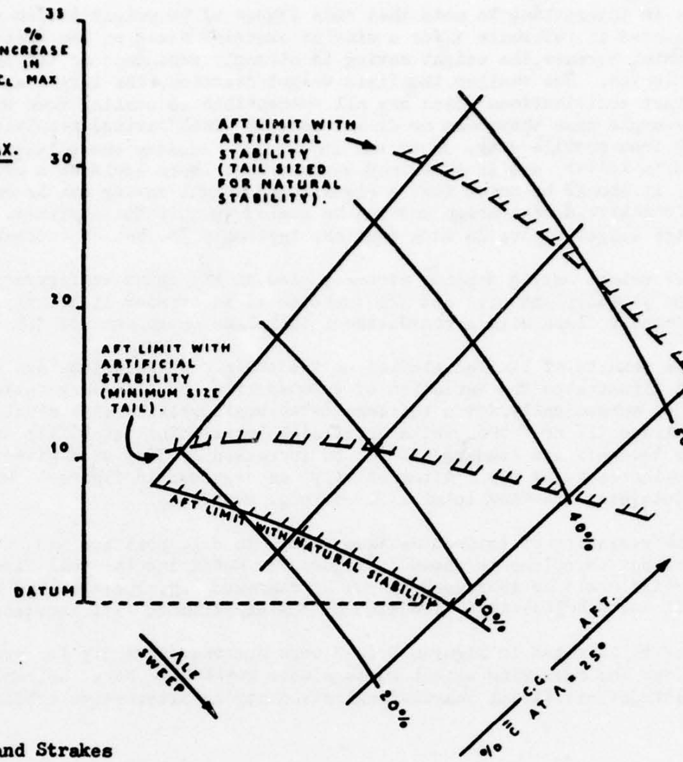


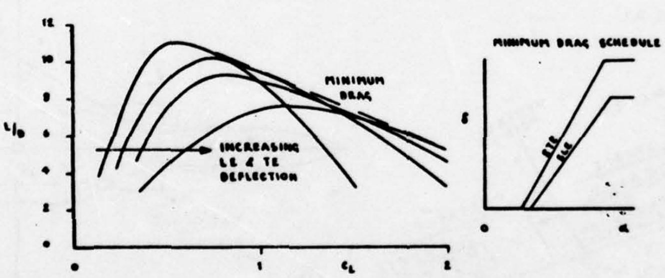
FIG. 7 Y-G A/C VARIATION OF TRIMMED C_L MAX. WITH SWEEP AND C.G. POSITION.



2.2 Integration of Manoeuvre Flaps and Strakes

Combat high lift devices can be used at subsonic speed not only to augment lift, but to reduce drag. Deflection of both leading edge and trailing edge high lift devices entrains a profile drag penalty but reduces lift-dependent drag at high lift at subsonic speeds. A typical family of L/D curves is shown in figure 8; the minimum drag envelope is indicated and the associated deflection schedules are depicted in the inset diagram.

FIG.8. EFFECT OF COMBAT HIGH LIFT DEVICES ON LIFT/DRAG RATIO AT SUBSONIC SPEED.



If operation of these surfaces is under the control of the pilot their benefits are reduced by the following effects:-

- **the nose down pitching moment due to flap deflection at any incidence has to be balanced by a down load on the tail, entraining lift and trim drag penalties**
- **the nose down pitching moment due to flap deflection at high speed low incidence results in increased tail loads, leading to a weight penalty**
- **in a tight air combat situation the pilot can not be expected to monitor incidence, except very approximately, so the flaps are likely to be in the "wrong" position for some of the time**

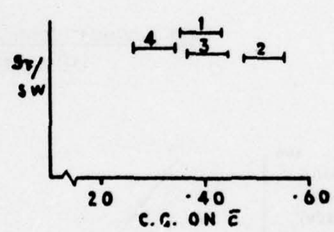
If the combat high lift devices are scheduled automatically with incidence, then their pitching moments may be used to advantage to trim wing-body lift, with an appropriate aft c.g. location. Further, tail loads are minimised and the aircraft operates on the minimum drag polar at all times. In this case, the c.g. range is located from trim drag considerations rather than being situated precisely between the down-load and up-load limits of the tailplane. It may be therefore that the full lift capability of the tail is not utilised at one or other of the limits. If the flaps are operated at all times in the automatic mode (i.e. they are never down at low incidence, even at low speed) then large downward lift on the tail is avoided in all flight situations. However the mainwheel/aft c.g. relationship must be preserved so nosewheel lift on take-off remains as the one design case for down-load on the tail.

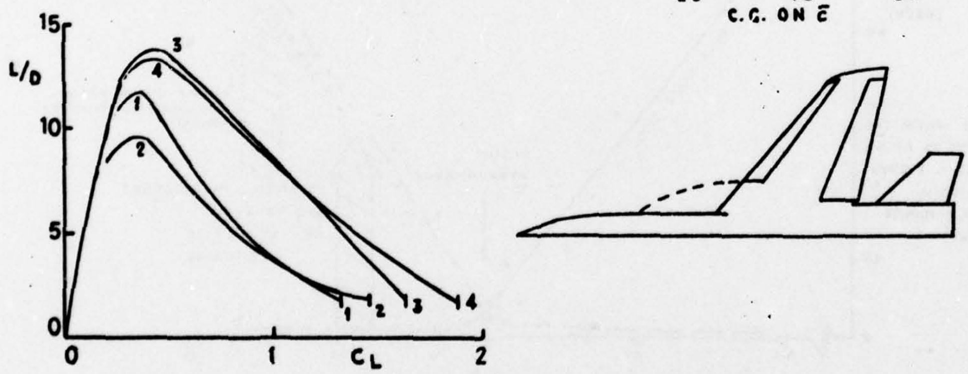
Leading edge strakes may be employed to advantage on low aspect ratio wings to augment lift and reduce lift-dependent drag at high incidence. They also achieve an orderly progressive flow breakdown, enhancing lateral stability and roll damping and enabling deeper buffet penetration. Because they produce vortex lift, proportional to incidence squared, their contribution to longitudinal stability is non-linear. With natural stability, the tail must be sized and the c.g. located to ensure stability at the worst point, namely at high incidence. With non-linear pitching moments therefore, stability must be excessive at moderate incidence, resulting in significant lift and drag penalties due to trimming. With artificial stability, on the other hand, and the c.g. range located to minimise trim drag, using scheduled flaps, the non-linear pitching moments are no longer an embarrassment. At low and moderate lift coefficients tail loads are small. With increasing incidence, as vortex lift from the strake develops it is balanced by up-loads on the tail up to the point where the maximum (tail-off) wing-body moment is balanced by maximum usable tail lift.

Figure 9 shows a selection of possible combinations of natural and artificial stability, manual and scheduled flaps with and without strakes, with associated c.g. ranges, tailplane size, maximum L/D and maximum lift values. The superiority of the integrated solution with scheduled flaps and artificial stability is evident.

FIG. 9 EFFECTS OF ARTIFICIAL STABILITY AND SCHEDULED COMBAT FLAPS ON LIFT/DRAG RATIO, MAXIMUM LIFT, C.G. LOCATION AND TAIL SIZE

CASE	STABILITY	FLAPS	STRAKES	C.G. LOCATED FOR
1	NATURAL	MANUAL	NO	MIN. TAIL SIZE
2	ARTIFICIAL	MANUAL	NO	MAX. LIFT
3	ARTIFICIAL	SCHEDULED	NO	MAX. L/D
4	ARTIFICIAL	SCHEDULED	YES	MAX L/D.





2.3 Performance Benefits

2.3.1 Air Combat

Mathematical modelling and piloted simulation of air to air combat has been used to evaluate the effects of aircraft performance parameters on combat success. The most commonly used performance parameters are:-

- **sustained turn rate (thrust-limited) - STR**
- **attained turn rate (lift-limited) - ATR**
- **specific excess power - SEP**

Various companies and agencies have evolved different combinations of these performance parameters to correlate with alternative combat scoring criteria. The Combat Correlation Parameter (CCP) currently favoured by BAC for low altitude subsonic combat, when both aircraft are aggressive, is proportional to the product

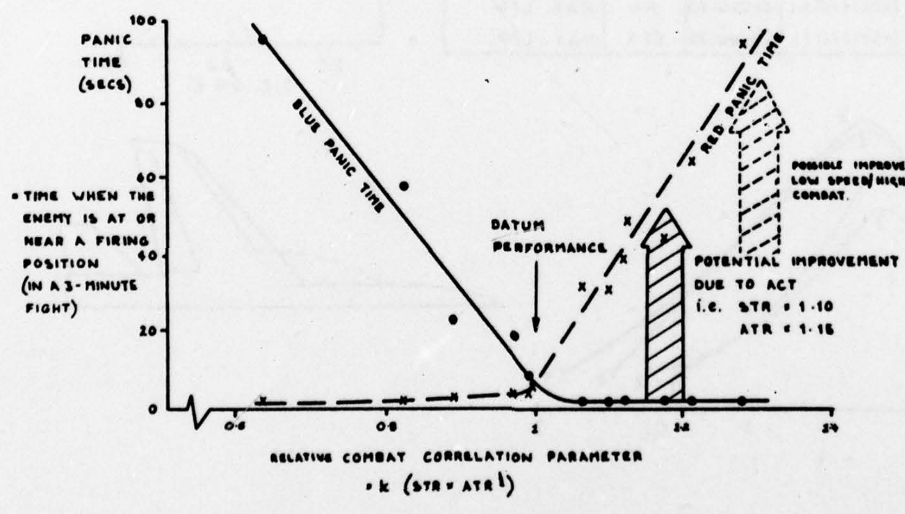
$$STR \times ATR^{\frac{1}{2}}$$

Currently also the favoured scoring criterion is "panic time" being the time during which an aircraft is in imminent danger of being shot down and must take immediate evasive action; in other words his adversary is at, or close to, a firing opportunity.

Sustained turn rate at a given speed is (approximately) proportional to sustained normal acceleration which is equal to the product of Thrust/Weight ratio and Lift/Drag ratio. Attained turn rate is proportional to *maximum* lift divided by *wing* loading. So the *effects* on combat correlation parameter at constant thrust/weight ratio and wing loading can be deduced directly from the changes in lift/drag ratio and maximum lift due to ACT.

Taking the example from the previous paragraph where the use of ACT has been shown to augment maximum $^{L}/_{D}$ by 10%, maximum lift by 15%, the combat correlation parameter is increased by 18%. Figure 10, based on BAC combat simulation, shows the relationship between "panic time" and the combat correlation parameter for a number of one versus one 3-minute combats at low altitude, starting at high subsonic speed. When both aircraft have equal performance (CCP = datum value) neither is at risk for any significant time. However in a closely matched situation any performance advantage is turned into an overwhelming tactical advantage. The 18% improvement due to ACT, applied to the "blue" aircraft puts the "red" aircraft into a "panic" situation for 60 seconds - $\frac{1}{3}$ of the time.

FIG 10 COMBAT CORRELATION FOR STUDY AIRCRAFT
(LOW LEVEL ENGAGEMENTS)



This 18% advantage due to ACT is based on the improvement in maximum L/D, that is on the assumption that at the mean combat speed the sustained manoeuvre limit coincides with the lift coefficient for maximum L/D. This is likely to be so only if the speed is kept high. Frequently in combat the speed "unwinds" to 0.6M or lower. In this case, taking for example an aeroplane with a wing loading of 400 kg/m^2 and thrust/weight ratio = 1 at 0.6M/5000ft the sustained turn rate is improved by 20% and combat correlation parameter by 30%, giving a "panic time" of about 100 seconds (figure 9).

ACT thus changes a stalemate situation to one of clear dominance for the ACT-equipped aeroplane.

2.3.2 Fuel Economy

In a battlefield air superiority mission a typical breakdown of fuel usage is:-

Flight Phase	Fuel % Total Fuel	Used % Take-Off Mass
Combat	40	10
Cruises	30	7.5
Loiter (CAP)	20	5
Reserves etc	10	2.5

Benefits in maximum lift/drag ratio yield proportional decrements in thrust and fuel requirements for loiter. If the combat task is defined in terms of sustained manoeuvres, a similar yield is obtained. In cruise phases the benefit is halved, 10% on maximum L/D resulting in 5% fuel saving. Taking the fuel breakdown above and 10% improvement in maximum L/D, the fuel saved due to application of ACT becomes:-

	% Total Fuel		
Combat	4		
Cruises	1.5		
Loiter	2		
Reserves	0.5		
TOTAL	8%	=	2% mission take-off weight

The fuel saved can be used either to increase performance yielding

20% improvement on combat endurance
or
25% improvement on radius of action
or
40% improvement in CAP time

Alternatively, maintaining the same mission performance it could be used to reduce mission take-off weight. In the design stage the weight saving is greater than 2% because of the reduction in structure and systems weights associated with the lower design weights, the cumulative effect being expressed in the growth factor. A typical growth factor for this class of aeroplane with fixed engine and wing loading is 2, yielding a weight saving of

4% in mission take-off weight

due to the improvements in aerodynamic efficiency, conferred by ACT.

Additional bonus points are:-

8% reduction in take-off distance
2% reduction in landing distance
3% increase in 1g SEP

3. CAREFREE MANOEUVRING

3.1 Stall Departure and Spin Prevention (SDSP)

Most combat aeroplanes currently in service can not exploit their full lift capability, being limited by lateral/directional misbehaviour and divergence before the wing stalls. This can manifest itself in a number of ways such as wing drop, wing rocking and yaw-off which, if uncorrected, lead to a spin. At the best, a spin takes an aeroplane out of the combat: at the worst it results in loss of the aircraft and possibly the pilot.

Aerodynamic palliatives, such as combat high lift devices and strakes can postpone the onset of the limiting phenomena and provide a more controlled development of separated flow on the wing. Increased fin size and forebody shaping can be employed to delay or eliminate directional instability, but a stable forebody in sideslip produces a propelling moment when rotated in yaw, leading to a dangerous fast, flat spin tendency. These measures can remove the slippery slope, but the cliff edge is still there beyond. If spin prevention can be provided, the cliff edge is inverted; it becomes a safety barrier and pilots can exploit the full lift capability of their aeroplanes without fear of going "over the

edge". This is one feature of "carefree manoeuvring".

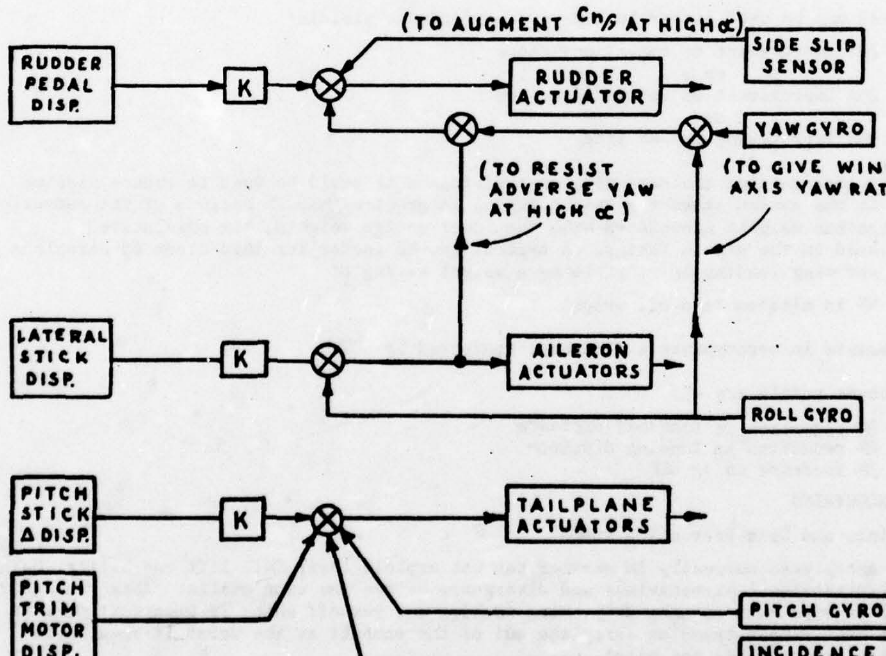
Stall departure and spin prevention involves:-

- **augmentation of lateral/directional stability as necessary to ensure adequate dynamic Cn_{β} * at the incidence limit**
- **lateral control co-ordination to ensure positive directional stability with bank angle constrained (LCDP)* at the incidence limit**
- **augmentation of yaw and roll damping as necessary to ensure adequate dutch roll damping at the incidence limit**
- **restriction of the pilot's roll and yaw control authority as the incidence limit is approached to prevent large inertia coupling moments being generated in manoeuvres**
- **increasing incidence feedback in the longitudinal control loop as the incidence limit is approached, leading to apparent "hyperstability" at the limit**

The level of stability provided, about all axes, must be sufficient not only to contain pilot-induced manoeuvres, but also to resist external disturbances such as entering the wake of another aircraft. The latter is a frequent occurrence in air to air combat and has been the trigger action of some spinning accidents.

Figure 11 shows the block diagram of a stall departure and spin prevention system.

FIG 11. SIMPLIFIED BLOCK DIAGRAM OF AN ELECTRONICALLY SIGNALLED FLIGHT CONTROL SYSTEM INCORPORATING A STALL DEPARTURE AND SPIN PREVENTION SYSTEM.



WHERE

K IS THE RESPECTIVE CONTROL INPUT GAIN SCHEDULED WITH DYNAMIC PRESSURE & INCIDENCE TO IMPROVE AIRCRAFT HANDLING AND TO PERFORM THE STALL DEPARTURE AND SPIN PREVENTION FUNCTION.

The benefits of such a system are:-

- improved usable lift capability up to the lift limits of the wing
- improved "pointability" - that is flight over a wider range of attitude, beyond the stall, which can provide additional missile firing opportunities in combat
- elimination of spinning accidents which currently account for a large proportion of combat aircraft losses
- removal of pilot anxiety

The effects of the first three of these may be evaluated in combat simulations and expressed in numerical terms. Improved usable lift alone can provide (figure 12) 20% improvement in attainable turn rate. Elimination of spinning accidents yields a direct economic reward. Removal of pilot anxiety can not readily be translated into numerical terms. It couples with automatic configuration management (ie.scheduled operation of combat flaps, wing sweep etc) in relieving the pilot of the task of continuously monitoring incidence and air speed and operating secondary controls to obtain best performance and stay within limits. This allows him to concentrate on manoeuvring to best advantage and operating the weapons system, undoubtedly leading to increased combat success.

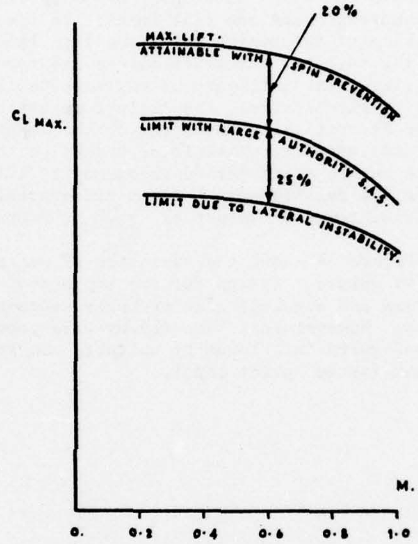


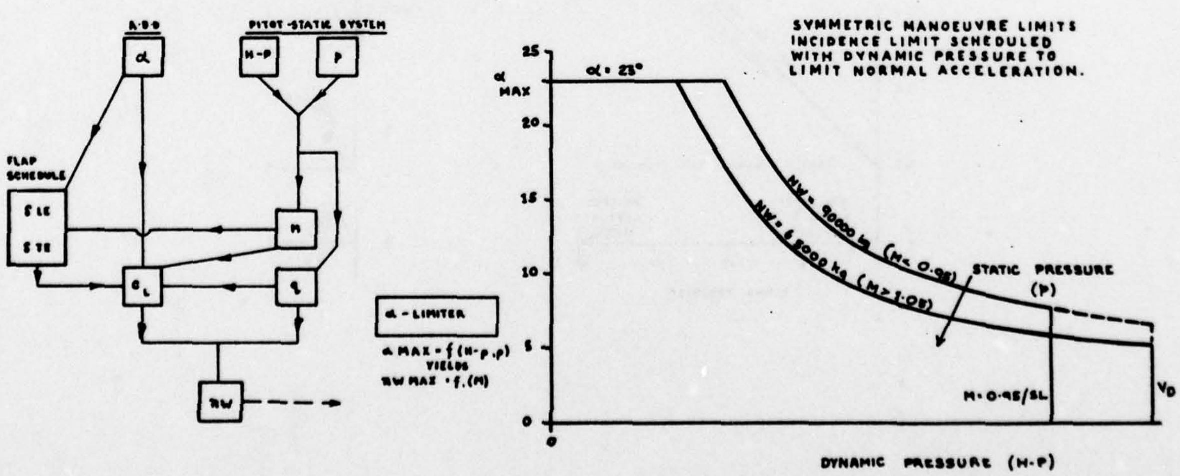
FIG 12 MANOEUVRE LIMITS EFFECT OF "FLY-BY-WIRE"

3.2 Overstressing Prevention

3.2.1 Symmetric Manoeuvres

Spin prevention involves limitation of incidence. An incidence limit which is related to dynamic pressure (H-P) and static pressure (P) provides a means of limiting normal acceleration; in fact, more precisely, it will limit the product of normal acceleration and aircraft weight (nW) which is more desirable, giving a constant proportion of the design nW in a given configuration. With automatic configuration management the flaps will be scheduled automatically with Mach number and incidence, so their effects on lift coefficient can be included in the computation of nW without direct inputs from flap position sensors. The flow diagram of the computation of nW is shown in figure 13. together with an illustration of the scheduled incidence limit.

FIG 13 OVERSTRESSING PREVENTION



Inability to overstress the airframe will contribute to carefree manoeuvring. It does however raise the following questions regarding definition of the maximum usable normal acceleration in relation to the design limit:-

- should the limit be defined with the traditional factor of (usually) 1.5 between service limit and proof load?

  or

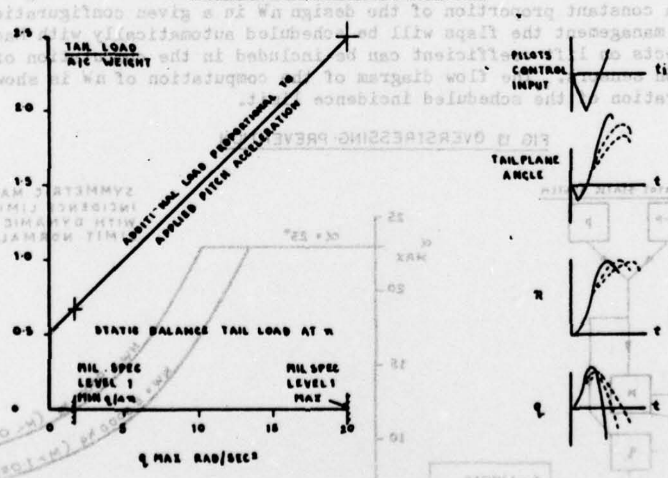
- should the limit be set somewhere between the placard limit and the proof limit to allow some margin for *extreme manoeuvres (e.g. collision avoidance)*?

- will the inability to break the aeroplane cause pilots to apply high g more frequently, to the detriment of fatigue life?

These questions can not be answered immediately; experience is needed with development systems. It is clear however that any overstressing prevention system must be incapable of being overridden. This does not mean that it can never be allowed to fail. If the pilot is warned of the failure of the automatic system he can revert to a safe "placard" limit; this will be acceptable provided that the probability of system failure combined with an extreme manoeuvre is sufficiently low to be ignored. This should allow the integrity requirement for such a system to be several orders lower than that of the basic flight control system.

A further way in which the control system can be used to prevent overloading is to safeguard tail loads. Design requirements (such as Mil. Spec. 8785) specify alternative longitudinal control "histories" to generate adverse wing and tail loads. In the specified pull-up/push down manoeuvre with the normal 'g' peaking at the design value the tail load on a naturally stable aeroplane is dependent primarily on the tail-off stability margin and the maximum rate of operation of the longitudinal control surface; the influence of any autostabilisation is small. With instability in pitch and full-time fly-by-wire control, the picture is entirely different. The worst tail load in the checked manoeuvre is critically dependent on the magnitude of the nose down pitch acceleration applied at peak normal acceleration, which is dependent on the longitudinal control law. Handling qualities criteria which relate short period frequency to lift slope (n/α) might be expected to provide some guidance on the relationship between pitch acceleration and normal acceleration ($\dot{q}/n$) but these allow a very wide range of values of $\frac{w^2 P}{nz_\alpha}$, proportional to $\frac{\dot{q}\ max}{\Delta n}$, ranging from 0.28 to 3.6 (13 to 1). *Figure 14 shows the variation of design* tail load with the value of $\dot{q}/\Delta n$ over the Level 1 range of values. Design for the top end of the range is to be avoided, in the interests of weight saving and simulation is probably necessary to decide at what point down the scale control becomes sluggish. However full-time fly-by-wire provides the ability to limit the maximum pitch acceleration to safeguard tail loads by suitable control laws. No longer is it necessary to design for the worst "ham-fisted" pilot input.

FIG.14 EFFECT OF ANGULAR ACCELERATION IN PITCH APPLIED AT PEAK NORMAL ACCELERATION ON DESIGN POSITIVE TAIL LOAD.



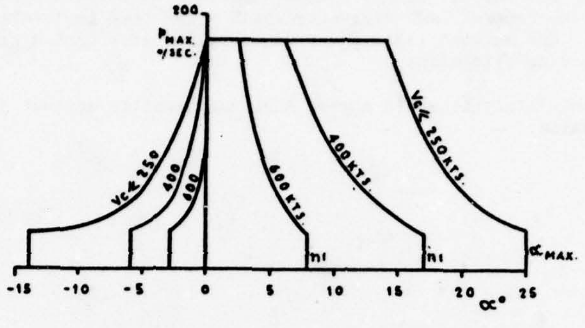
3.2.2 Asymmetric Manoeuvres

In rolling pull-out and push-over manoeuvres, severe loading and handling problems can arise if full roll control is used at extremes of positive and negative 'g'. Examples are:

Fin and Rear fuselage loads
Store attachment loads
Wing internal tank pressures
Autorotational rolling

By scheduling the demanded rate of roll with incidence, dynamic and static pressure, the maximum rate of roll can be controlled as a function of normal acceleration (nW, in fact); the law can also be made a function of CAS or Mach number if so desired. A possible schedule is illustrated in figure 15.

FIG 15 DEMANDED MAXIMUM ROLL RATE SCHEDULED WITH INCIDENCE AND DYNAMIC PRESSURE TO PREVENT OVERSTRESSING AND MISHANDLING.



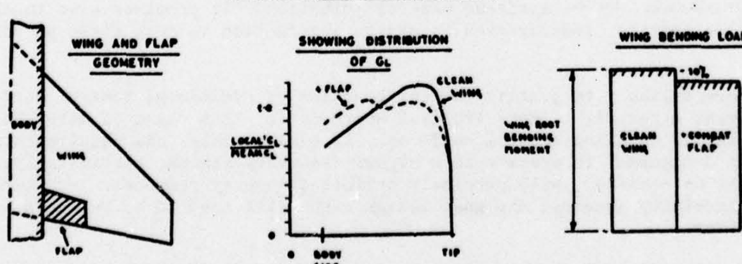
Removal of any restrictions on the use of roll control, together with unrestricted use of longitudinal controls over the complete range of normal acceleration, incidence, air speed and altitude offers carefree manoeuvring to fighter pilots of the next generation.

4. LOAD ALLEVIATION, RIDE CONTROL AND FLUTTER CONTROL

4.1 Manoeuvre Load Control (MLC)

Figure 16 compares the spanwise load distributions and resulting wing root bending moments, with and without a part-span combat flap on a low aspect ratio swept wing. Deflection of the inboard flap at high 'g' "flattens" the spanwise loading resulting in a reduction in wing root bending moment of about 10%. This could result in a weight saving of about 1 %, worthwhile in its own right. However if this weight saving is to be realised, the flap must be scheduled with normal acceleration, rather than incidence, which is likely to result in drag penalties in low altitude, high speed flight. Further, this MLC application requires a combat flap of relatively small spanwise extent compared with a flap designed to minimise drag in manoeuvring flight, the latter being scheduled with incidence rather than 'g'. Overall, it is considered that combat flap design will be dominated by performance considerations rather than MLC as such and will yield a greater benefit by means of the wing area reduction and fuel savings, discussed in earlier paragraphs.

FIG 16 MANOEUVRE LOAD CONTROL



4.2 Ride Control

The use of incidence-scheduled or normal acceleration-scheduled flaps, operating in the reversed sense to that used for lift augmentation and drag reduction, offers a means of reducing the effective lift slope of a wing, alleviating gust response. At first sight enormous reductions would appear to be possible. For example a 25% chord full span trailing edge flap on a low aspect ratio swept wing has a value of a_2/a_1 of about $\frac{1}{8}$; therefore with a flap: incidence gearing 4:1 a 50% reduction in lift slope would be realised in steady state conditions. In practice, when the dynamic response of the sensors and controls, the compensating longitudinal control inputs and the aeroelastic losses on the flaps are taken into account, the benefit is very much smaller.

Figure 17, taken from a BAC study, shows the percentage reduction in r.m.s. normal acceleration obtained versus system gain, both for an ideal system free from lags and for practical systems with realistic lags and filters. The improvement obtained is only about 10 - 15% reduction in r.m.s. 'g' compared with 60% for an "ideal" system. Further improvement could be obtained only by a large increase in the control loop frequency which would lead to interactions with structural mode frequencies. The maximum improvement was obtained at a system gain of about 50% of that required to eliminate wing lift slope.

From this study it would appear that the benefits may not justify the cost and complexity of such a system.

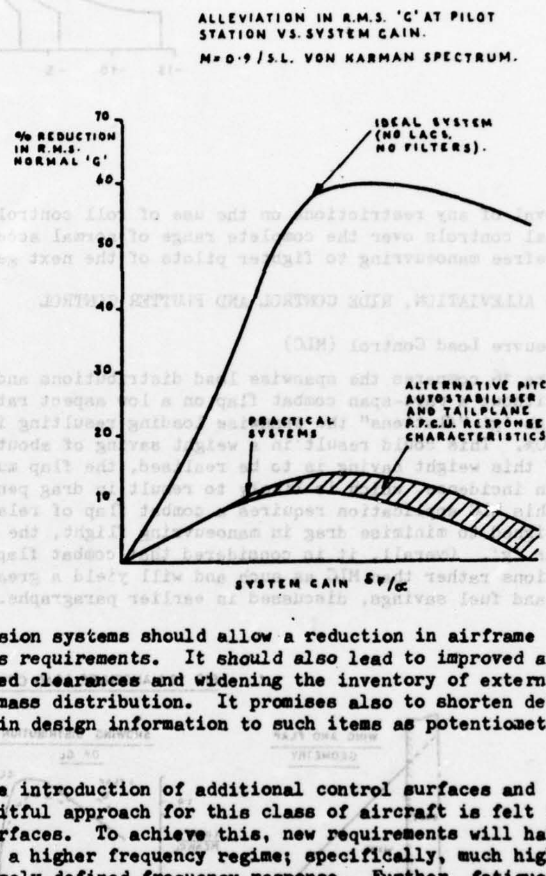


FIG 17 ACTIVE RIDE CONTROL SYSTEM

4.3 Active Flutter Control

The application of active flutter suppression systems should allow a reduction in airframe weight due to relaxation of structural stiffness requirements. It should also lead to improved aircraft effectiveness and versatility by increasing speed clearances and widening the inventory of external stores, due to reduced sensitivity to airframe mass distribution. It promises also to shorten development time scales by confining critical inaccuracies in design information to such items as potentiometer settings.

These benefits are unlikely to justify the introduction of additional control surfaces and actuators on small combat aircraft. A more fruitful approach for this class of aircraft is felt to be flutter suppression via the existing control surfaces. To achieve this, new requirements will have to be put to the actuator designers, to operate in a higher frequency regime; specifically, much higher rates of operation must be achieved, with precisely defined frequency response. Further, fatigue implications must be carefully assessed and gust design cases will need to be reviewed with the airworthiness authorities.

The integrity requirements for active flutter systems need not be as high as those of the basic flight controls; a restricted flight envelope could be acceptable after first failure. However with the proposed design approach the integrity of the computing and actuation will be identical to that of the basic flying controls and therefore sufficient. An adequate degree of redundancy will have to be provided in the additional sensors (accelerometers) which have to be provided to measure the structural responses.

Although the development required in the actuator field is likely to prevent immediate adoption of active flutter control for the next generation of combat aircraft, enough work must be done at least to ensure that flutter characteristics are not degraded by structural mode interactions with the flight control system. So, in fact we have already set foot on the road towards active flutter control without consciously taking that step.

5. IMPLEMENTATION OF FULL-TIME FLY-BY-WIRE

5.1 Degree of Redundancy

The most attractive application of active controls to combat aircraft - artificial longitudinal stability, combined with automatic configuration management - requires a full-time, full-authority fly-by-wire control system for its implementation. This implies complete reliance on the continuous operation of the essential sensors, computers, actuators and their associated power supplies, for flight safety. In unstable flight, catastrophic failure will occur with a few seconds of runaway or "freezing" of the controls. Mechanical or direct electrical reversionary links are useless in this situation; standby power supplies must come instantaneously "on line" in failure situations: tolerable interrupts are measured in milliseconds.

It is assumed that degradation of flight safety on the introduction of full-time FBW is intolerable: we strive continuously to improve flight safety. Therefore a reasonable objective to set "on paper" is a safety rate due to flying control failures an order of magnitude better than that of the current generation of combat aeroplanes. The latter is of the order of one aircraft lost in 10^6 flying hours, putting the target safety rate at:-

$$1 \times 10^{-7}$$

Failure rates of current single channel systems are of the order of 1×10^{-3} which yields loss rates (assuming loss of all but 1 channel results in loss of control) of:-

3×10^{-6} for a triplex system
4×10^{-9} for a quadruplex system

The former figure being insufficient, the latter possibly being "over-kill".

Improvement over the current flying controls safety rate may not be a valid target however, because it neglects, for instance:-

- aircraft saved due to safer handling (e.g. spin prevention) with fly-by-wire

- self-monitoring capability of digital systems, allowing continued flight on a single channel following certain types of failure

Another line of attack is to improve the reliability of components and sub-systems to allow the target safety rate to be achieved with a lower level of redundancy. Not the least of the problems of this approach however is the testing time required. For example reference 2 quotes 2.6 years of continuous testing to establish 90% confidence in 1×10^{-4} reliability.

Overall therefore it seems that quadruplex redundancy is required for the first generation of full-time fly-by-wire systems, but there is hope of reducing this to triplex in future generations.

5.2 Choice of Sensors

The achievement of optimum handling qualities for all operational tasks over a wide flight envelope is likely to require a mix of aircraft motion sensor signals from:-

Rate gyroscopes
Accelerometers
Airstream Direction Detectors (ADDs)

In failure states loss of some of these sensors may be acceptable, but for continued controlled flight motion sensing about all axes must be provided; this demands quadruplex (at least) signals from the essential sensors.

Multiplex ADD installations require individual corrections to be applied to each sensor to compensate for local flow and cross-axis effects; these corrections are, in general, functions of Mach number. Multiplex accelerometer installations require individual corrections to be applied for local acceleration effects due to angular rates and accelerations; they are also subject to individual local effects of structural motions, which are extremely sensitive to positioning. For these reasons the correction and consolidation of sensor signals in multiplex installations of these types of sensors is very complex. Multiplex rate gyroscope installations on the other hand are relatively insensitive to such effects and for this reason are the preferred type of sensor for the "hard core" of the control system. The only question is whether adequate handling qualities can be achieved for safe flight with only angular rate feedback signals. BAC studies indicate that acceptable handling in pitch can be provided with an appropriate blend of proportional, + integral + differential pitch rate signals with both stable and unstable aerodynamics, although for satisfactory handling over the complete speed range incidence and, possibly, normal acceleration signals are required. On the roll axis, rate demand control is now conventional and the question of static stability does not arise. In yaw, stability augmentation by rate feedback alone is believed to provide a sufficient level of stability to prevent divergence should a failure of sideslip or lateral accelerometer signals occur in flight at high incidence, where natural stability is inadequate; recovery to low incidence, (natural stability) may be necessary to prevent longer-term divergence.

5.3 Signal Processing

A few years ago there was considerable debate over the question of analogue or digital computing for future flight control systems. Now the choice seems clear; digital computing is almost universally favoured for the following reasons:-

- **easier implementation of complex control laws e.g. non-linear gearing schedules**
- **ability to generate integral and differential functions of sensor inputs**
- **better interlane accuracy**
- **exact, repeatable performance, insensitive to environment**
- **reducing size and cost**

Potential hazards of digital systems are the questions of software integrity and susceptibility to electro-magnetic interference, particularly lightning strike. These problems have to be seen to be overcome - the former by rigorous checking procedures, the latter by adequate testing, before digital systems can be declared airworthy.

5.4 Actuation

The integrity established upstream in the quadruplex sensing and computing lanes must not be allowed to drain away at the interface with the power controls. A number of possible solutions for systems which can survive a single hydraulic system failure combined with an electrical lane failure have been investigated. The solution favoured by BAC at the present time is duo-triplex first stage actuation, with conventional, duplicated hydraulic supplies. Each triplex first stage is served by one hydraulic system. Single hydraulic failure leaves three electrical lanes, served by the live system, operative, with ability to lose one of those lanes without loss of control. The type of actuator preferred is the failure absorption actuator. In this type a failed lane is not de-selected but overridden by the live lanes. This leads to considerable simplification compared with positive rejection schemes with acceptable degradation of performance following failures.

Other alternatives, such as triplex hydraulic systems or separate control surface redundancy, which may superficially appear simpler solutions have, on deeper investigation, found to be more complex and heavier than the solution of single multiplex actuator operating each of the conventional minimum number of control surfaces.

5.5 Compromises Necessary

One of the limitations encountered in the definition of control laws for fly-by-wire systems is the compromise necessary to cater for the wide range of loadings and configurations. A typical strike aircraft with fuselage and wing-mounted stores spans very wide ranges of mass (2:1), moments by inertia (4:1 in $\frac{I_{zz}}{I_{xx}}$), longitudinal, lateral and directional stability with different external store loads. The control laws must perforce be a compromise because it is impossible to provide optimum handling qualities in all configurations. Inevitably also, manoeuvre restrictions must be based on the worst stability characteristics.

It would be possible to use signals from the stores management system and fuel system as inputs to the flight control system to provide some form of "configuration sensing" for scheduling the control gains. However multiplex discrete signals would be required from each source and the number of these leads to formidable integrity problems.

This is an area which warrants further study; alternative solutions should be sought.

6. THE JAGUAR FLY-BY-WIRE PROGRAMME

To pave the way for the next generation of combat aircraft, a national (UK) research programme into full-time digital fly-by-wire, based on a Jaguar airframe, has been launched. The objectives are to demonstrate:-

- **confidence in the airworthiness of full-time digital fly-by-wire**
- **control of longitudinal instability**
- **stall departure and spin prevention**

The quadruplex, digital flight control system and the associated power supplies have been defined to meet the same safety rates as for an operational service aircraft.

Engineering features of the fly-by-wire Jaguar are indicated in figure 18.

The fly-by-wire Jaguar will, it is believed, be the first aircraft, other than the space shuttle, to fly with a digital full-time fly-by-wire control system. It is expected to provide the confidence to go ahead with full-time digital fly-by-wire control in the next generation of combat aeroplanes.

FIG 18 JAGUAR FLY BY WIRE ENGINEERING FEATURES.

4-PLEX DIGITAL COMPUTING AND SIGNALLING.
4 PLEX PRIMARY SENSORS (RATE GYROSCOPES)
2 PLEX SECONDARY SENSORS (A.D.D.'S, ACCELEROMETERS)
DUO-TRIPLEX FIRST STAGE POWER CONTROL ACTUATORS.

TWIN ELECTRO-PUMPS
UPRATED E-D HYDRAULIC PUMPS
INCREASED THRUST TAILPLANE P.F.C.U.'S.
EXTRA BATTERIES
UPRATED T.R.U.'S.

MINOR MODIFICATIONS TO EXISTING SYSTEMS

WING L.E. STRAKES AND AFT BALLAST FOR UNSTABLE FLIGHT.

7. SUMMARY OF BENEFITS AND RISKS

The application of Active Controls Technology to combat aircraft offers significant benefits in efficiency or performance. Particularly in the air superiority role it will provide future pilots with greatly enhanced combat effectiveness, for example:-

10% improvement in sustained manoeuvrability
15% improvement in attained manoeuvrability
25% improvement in radius of action

combined with carefree manoeuvring, free from the possibility of spinning or overstressing. An aeroplane so equipped will be a fighter pilot's aeroplane par excellence.

There is promise also of some additional benefits due to the application of active flutter control and ride control systems.

Undoubtedly there are development risks, as there have been with all technical innovations. For this reason the Jaguar fly-by-wire programme has been launched. Additionally, great care will be needed to control costs, not only of development but of procurement and of ownership over the total life span of the aircraft.

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The writer wishes to thank his colleagues, in particular Messrs. K. Carr, R. F. Dell, B. Gee, I.Jones & C. G. Lodge for their contributions to this Chapter.

F-16 MULTI-NATIONAL FIGHTER

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SUMMARY

The F-16 Multi-National Fighter is the first production aircraft developed in which an Active Flight Control System was incorporated from its inception. In the past, the design of a flight control system was undertaken after the basic aircraft aerodynamic design was set and was used mainly to improve handling qualities. This usually involved little more than augmenting pitch and lateral-directional damping. As aircraft handling and performance requirements increased, so did complexity of the flight control system. The desire to obtain uniform aircraft response to pilot commands results in command augmentation systems being used in the flight control system. Since these systems required large authority surface commands to achieve the desired response, the requirement for highly reliable electronic systems was generated and achieved. The achievement of this reliability has allowed the application of an Active Control System in the F-16. This paper presents a summary of the F-16 Multi-National Fighter Flight Control System. The basic functions of the Flight Control System are discussed as well as the unique features such as Relaxed Static Longitudinal Stability (RSS), Fly-By-Wire (FBW), and Side-Stick Pilot's Controller (SSC). In addition, the basic philosophy behind the selection of the Flight Control System functions and unique features as well as flight test results and future applications are discussed.

SYMBOLS

A.C.	aerodynamic center
A_n	normal acceleration
C_D	drag coefficient
C_L	lift coefficient
$L_{\alpha_{WB}}$	lift of the wing body due to angle of attack
L_{WBT}	total lift of the wing-body-tail
L_{α_T}	lift of the tail due to angle of attack
L_{δ_T}	lift of the tail due to deflection
LH	left-hand
$M < 1$	Mach less than one
$M > 1$	Mach greater than one
MAC	mean aerodynamic chord
P_T	total pressure
P_s	static pressure
RH	right-hand
RSS	relaxed static longitudinal stability
SM	static margin
T.E.	trailing edge
W	weight
α	angle of attack
β	sideslip angle
$\dot{\theta}$	pitch rate
δ_e, δ_H	horizontal tail deflection

INTRODUCTION

The relaxed static stability (RSS) concept is incorporated in the F-16 Multi-National Air Combat Fighter (Figure 1). The fighter, developed by General Dynamics, is equipped with an active fly-by-wire flight control system. The configuration of the flight control system for the F-16 evolved from extensive evaluations conducted on a six-degree-of-freedom, real-time, piloted flight simulator and from aircraft system performance flight evaluations on the YF-16 prototype aircraft. This flight control system is a full fly-by-wire control system; no mechanical linkages are used between the cockpit and the integrated servoactuators that power the primary control surfaces. The significant capabilities and features of the F-16 flight control system are:

- o Longitudinal relaxed static stability (RSS): allows the airplane to be balanced to achieve improved range and maneuvering performance.
- o Three-axis command and stability augmentation: provides precise control and excellent handling qualities.
- o Fail-operative/fail operative redundancy: provides a high degree of flight safety and a high probability of mission success.
- o Full fly-by-wire control system: provides maximum flexibility for tailoring flying qualities.
- o Automatic angle-of-attack limiting: allows the pilot to aggressively use the maximum capability of the airplane without fear of inadvertent loss of control.
- o Built-in self-testing capability: ensures flight control system flight readiness with minimum downtime for maintenance actions.

The primary flight control system interfaces directly with the secondary flight control and air data systems. Three-axis flight-path control is provided by the primary flight control system through movement of the primary control surfaces, whereas the secondary flight control system provides high-lift, aerodynamic braking, and improved maneuver performance through the movement of lift- and drag-modification devices. The air data system provides the aerodynamic intelligence.

Flight path control is achieved by actuating the all-movable horizontal tails for pitch and roll control, the partial-span, wing-mounted flaperons for roll control, and the conventional rudder for yaw control (Figure 2). Maneuver capability at high angles of attack is enhanced by a full-span leading-edge flap, automatically programmed as a function of Mach number and angle of attack. The result is a variable, near-optimum camber that maintains effective lift coefficients at high angles of attack, thereby providing a higher maximum lift capability, improved buffet characteristics and improved directional stability. Symmetrical downward deflection of the flaperons and fixed deflection of the leading-edge flap provide increased lift for takeoff and landing.

SYSTEM MECHANIZATION

System mechanization of the F-16 flight control system is accomplished by applying the basic system elements as shown in Figure 3. Pilot control of the primary control surfaces is accomplished without mechanical linkage by using displacement-type, force-sensing control stick and rudder pedals, and through pilot-initiated trim commands in each of the three axes. The relationships of flight control system input parameters, functional characteristics, levels of redundancy, and control surface actuation mechanization are shown in Figure 4. Four independent electronic branches (quadruple-redundant system) with quadruple redundant (quadrex) pilot sidestick controller sensors accept pilot command inputs. Feedback of quadrex airplane motion sensors (rate gyros and accelerometers) provide static and/or dynamic stabilization. The quadrex system is protected through automatic failure detection and correction mechanization to provide fail-operative performance following two like electronic failures. The prime electronic assembly is the flight control computer, which processes, gain-adjusts, filters, and amplifies signals to command the five integrated servoactuators used to power the primary control surfaces. Gain scheduling of commands in the various control axes, as required for a particular command, is performed as a function of static pressure, impact pressure, or the ratio of impact pressure to static pressure. These functions are supplied in quadrex form by the air data system, which senses triple sources of total pressure, static pressure and angle of attack.

System Self-Testing Capability. A built-in self test capability in the primary flight control system ensures flight readiness of the system with minimum downtime for maintenance actions. Self-test sequencing and monitoring is performed manually, automatically, or in combination. Manual self-testing requires test inputs from stick, pedals, switches, etc. Automatic (or semi-automatic) self-testing is more extensive and involves the initiation of system inputs from within the system. Three types of tests are available: (1) a safety-of-flight control confidence check, (2) a mission essential flight control confidence check, and (3) a complete test for maintenance purposes. The flight control warning and semi-automatic self-test system is shown in Figure 5.

Redundancy Concept. System tolerances to various types of failures are of primary concern when a fly-by-wire control system is used without a mechanical backup system. Electronics, actuators, electrical power, and hydraulics are the four basic areas of concern. Each of these areas has been carefully considered and provided with adequacy redundancy to assure mission success. A two-fail operative redundancy concept is employed in the F-16 (Figure 6).

Electronic Redundancy. In all areas, except air data sensing, where three pneumatic sources are used to obtain four electronic signals, four separate electronic branches serve the flight control system. The function schematic diagrams for the pitch, roll, and yaw axis are shown in Figures 7, 8, and 9. As can be seen from these figures, any two consecutive like-failures would not cause degraded performance and, therefore, would meet MIL-F-8785B Level 1 handling qualities. The effect of a third like-failure depends on the particular characteristics and location of the failure. The third failure results in either no degradation or zero command to one integrated servoactuator, depending on whether the failure is a hardover or a null failure. For the null failure, the aircraft would provide at least Level 3 handling qualities since the aircraft can be flown with any one surface at zero command.

Actuator Redundancy. F-16 is equipped with integrated servoactuators which make it the first production aircraft to use this unique design concept. Combining the command servo and the power actuator in one package reduced the weight, volume, and cost of the actuator package. In addition, it simplified the logistics problem since the flaperon and horizontal tail use the same integrated actuators.

Integrity of the integrated servoactuator is superior to that of any tandem surface actuator now in production. Figure 10 presents a schematic diagram of this unique device that combines the basic inherent reliability of conventional, integral, mechanical-position feedback with multiple-command capabilities of redundant electrical commands. The integrated servoactuator internally detects and corrects one failure (the second electrical or first mechanical servovalve); and the pilot can, at his discretion, arm another monitor in the flight control computer which will compare the actuator with an electronic model. If a failure is detected, a fail-safe command to the computer will cause the integral, mechanical feedback to command the servoactuator to a predetermined neutral position.

Electrical Power Redundancy. The adoption of the fly-by-wire approach necessitated special design considerations to assure uninterrupted electrical power to the flight control system. It was necessary that the power redundancy be consistent with the electronic redundancy; thus, four independent isolated power supplies were required for a quadruple system. The arrangement of these four independent isolated 26-V, single-phase, 800-Hz power supplies is shown in Figure 11. This schematic illustrates the arrangement of the two ac generators, a 40-KVA primary generator and a 5-KVA standby generator, that are employed to provide a redundant and reliable electrical power source for the control system. Each power supply receives inputs from each of the main 28-VDC buses and integral battery. The dc power is supplied from two 115/220-VAC-28-VDC converters. Each converter may receive input power from primary generator or from the standby generator. The emergency power unit is driven by engine high-pressure bleed air or hydrazine in the event of improper generator voltage or frequency, generator failure or engine loss.

Each power supply includes the basic 26-V, 800-Hz inverter; a 24-V battery, a battery charger, input power source selection logic; status annunciation logic; and self-test logic. Multiple input power sources preclude loss of a supply because of a malfunction in the input power system. The 24-V battery prevents power interruptions during primary power source switching operations.

In summary, the flight control system electronics receives four-branch, uninterruptable, regulated electrical power regardless of transient voltages or fault conditions elsewhere in the aircraft system.

Hydraulic Redundancy. There are two independent hydraulic systems. In addition, the emergency power system can pressurize the primary system with its hydraulic pump. A block diagram of this hydraulic power subsystem arrangement is shown in Figure 12. Level 2 handling qualities will still exist after loss of one hydraulic system.

Air Data Redundancy. The air data probe supplies two sources of total and static pressures. The third source of total and static pressure is obtained from the air data side probe. Consecutive loss of two air data sources will cause the flight control system to revert to standby gains (fixed values of static and impact pressure). Level 3 handling qualities will result.

Three sources of angle-of-attack data are provided; two single-conical, air-flow-detector-type angle-of-attack transmitters (one on each side of the aircraft) and a differential pneumatic signal from the air data side probe. The side-mounted angle-of-attack transmitters contain quadruple-redundant rotary voltage differential transformers, thereby making the angle-of-attack signal to the flight control computer two-fail-operative for electrical failures. The effect of two mechanical angle-of-attack transmitter failures is dependent on the actual failure modes.

Redundancy Management. One of the most critical items in multiple-redundant flight control system design is redundancy management. Redundancy at the system level introduces considerations that do not arise in nonredundant systems. In the case of systems with three or more channels, a more complex redundancy management involves signal selection, failure detection, failure isolation, and recovery from system faults. The redundancy management approach selected for the F-16 has been proven in laboratory and flight tests. The basic concept is an extension of the F-111 technology, which demonstrated high operational ability.

An active signal selector is the heart of the electronic redundancy management technique used for the F-16. The selector will select a good signal from quadrex redundant signals in each of the three control axes (pitch, roll, and yaw). As indicated in Figure 13, the quadruple-redundant system employs four independent signal branches, i.e., each input signal source (pilot, inertial sensor, etc.) designated Branches A, B, C, and D. Each of the four branch signals is processed independently and sent to the signal selector in each of the four branches.

The active selector approach was chosen because (1) excellent quality signals can be achieved, (2) the required clamping circuits are not complicated, and (3) the third consecutive failure to the active selector will not cause a hardover command (can be only good or passive). This selector concept incorporates a middle-value-select voting scheme along with a standby redundancy feature. In the normal (no-failure) condition, Branch D in each selector is switched to a standby condition, and the signal selector functions as a middle-value voter for the three remaining branch signals, i.e., the middle value of Branches A, B, and C is selected while all four branches are continuously monitored. Should there be a failure, the failed branch would be detected by its monitor and switched out in each branch signal selector. At the same time, Branch D would be switched from standby status to active control in place of the failed branch, provided Branch D had not failed. After the second failure, the signal nearest zero is selected and passed to the servoamplifiers.

SYSTEM SELECTION AND RATIONALE

Two principal factors influence the F-16 active flight control system design: (1) the impact on airplane design, and (2) the design considerations involved in defining the flight control system configuration. There are many trade-offs associated with the airplane and its control system definition; in this section, the more important and significant factors are discussed.

The major features that were selected for the F-16 are:

- Relaxed Static Stability
- Fly-By-Wire
- Minimum Displacement Controllers
- Angle-of-Attack and Normal Acceleration Limiting.

Relaxed Static Stability (RSS). For the F-16 fighter aircraft, maneuverability, weight, and range are the paramount requirements that influenced the selection of the relaxed static stability (RSS) concept as a basic CCV approach for designing the aerodynamic configuration. Basically, the relaxed static stability concept involves (1) balancing the airplane to optimize performance and maneuverability, (2) providing sufficient aerodynamic control to avoid departures due to aerodynamic and/or inertial disturbances, and (3) tailoring the flight control system to optimize handling qualities. Differences between an airplane with relaxed static stability and a conventionally balanced airplane are shown in Figures 14 and 15.

In the subsonic flight regime (Figure 14), the conventionally balanced airplane is shown to have the resultant wing-body lift acting slightly aft of the center of gravity, the nose-down moment produced by the wing-body lift component is balanced by a download on the horizontal tail, and the tail lift subtracts from the wing-body lift. For the RSS-balanced airplane the wing-body lift acts forward of the center of gravity and requires an up-load on the horizontal tail to balance the nose-up pitching moment of the wing-body. In addition, the lift on the tail of the RSS (Relaxed Static Stability) airplane is additive to the wing-body lift, thus resulting in a higher total lift.

In Figure 15, the same information is shown for the supersonic flight condition. In this case, both the conventionally balanced and the RSS airplanes have the wing-body lift vector acting aft of the center of gravity. Because the RSS airplane has a center of gravity farther aft than the conventionally balanced airplane, the nose-down pitching moment due to the wing-body lift requires a much smaller down-load on the horizontal tail for trim. The result is a higher obtainable lift for the RSS airplane in the supersonic flight regime.

Further insight into the differences between a conventionally balanced airplane and one employing RSS can be obtained by an analysis of Figure 16 which illustrates a typical aerodynamic center (a.c.) variation in Mach number. A conventional aft-center-of-gravity limit is shown on this plot. Normally, this limit is based on maintaining some minimum positive static margin, nominally three to five percent. When an airplane is balanced in the conventional manner, large static margins are inherent in the supersonic flight regime because of the significant aft movement of the airplane aerodynamic center. A typical RSS aft limit is also shown on the same illustration. It can be seen that the subsonic static margin is much less positive for the RSS configuration.

The performance benefits derived by adopting the RSS concept are basically attributable to (1) higher trimmable lift coefficients, and (2) lower trim deflections with their attendant drag and tail load reductions. Typical performance benefits that can be expected from reduced static stability are shown in Figure 17. Performance gains for a typical air-combat mission are indicated by comparing a conventional balanced airplane having a 25 percent MAC c.g. with those of an RSS airplane having a c.g. in the range of 35 to 40 percent.

Fly-By-Wire. When the RSS concept is employed, the basic airplane static stability must be augmented in the subsonic flight regime where negative static margins are present. Basically, an unstable airplane is flown as a stable airplane, and the stability and command augmentation system permits normal pilot control techniques while masking the unstable free airframe. If mechanical components are used to transmit pilot commands to control surfaces, electrical components are still involved to implement the command and stability augmentation system. It follows that the retention of mechanical components for transmission of pilot stick commands is unjustifiable (weight and cost-wise), since an unstable airplane cannot be controlled in flight without the benefit of a full-time operating stability and command augmentation system. This places increased demands on electronic reliability. However, once the demands on electronic reliability and the commensurate demands on electrical and hydraulic power requirements are met, the obvious step is to take advantage of this reliability by incorporating a full fly-by-wire flight control system.

The decision to employ a full fly-by-wire flight control system in the F-16 airplane was primarily based upon obtaining improved airplane performance through the use of RSS. Therefore, the selected flight control system concept was a natural outgrowth of a redundant electronic control system required for the augmentation system in an unstable (i.e., RSS) airplane.

Minimum Displacement Controllers. The application of fly-by-wire flight control system permits a simple implementation of the pilot's control stick and rudder pedals. Particularly, the fly-by-wire approach to the flight control system is compatible with force-sensing devices for accepting pilot input commands. Additionally, results of YF-16 trade studies on conventional center location versus a side location and displacement versus force-sensing mechanizations favored the force-sensing, side-stick controller. Advantages of the force-sensing, rudder pedal and side-stick controller combination are (1) no linkage dynamics or friction felt at the controller, (2) no linkage balancing problems, (3) enhanced system survivability, (4) greater freedom in airframe design compatible with the high "g" cockpit arrangement, and (5) potential for weight, space and cost savings.

The pilot's side-stick controller (Figure 18) used in the F-16 is equipped with physically and electrically isolated transducers so that forces (approximately 1/8-inch total displacement) on the stick grip generate quadrex electrical signals to command pitch and roll inputs to the flight control system. The force-sensing element contains quadrex transducers in both the pitch and roll axes.

Like the pilot's side-stick controller, the pilot's rudder pedal assembly is a minimum-deflection force-sensing device. In appearance this assembly is similar to a pair of conventional rudder pedals and the pilot achieves directional (yaw) control through the translation of either pedal (approximately $\pm$ 1/2 inch) to generate quadrex electrical signals by use of a linear variable differential-type transducer. These signals are summed with other gain-adjusted signals in the flight control computer and sent to the rudder integrated servoactuator.

Angle-of-Attack and "G" Normal Acceleration Limiting. Considerable time and effort has been spent during the course of the F-16 development to determine the optimum aircraft configuration to enhance its performance as a fighter. Since turn performance is one of the main indicators of fighter performance, we have leaned heavily on this parameter to develop the configuration. Turn performance is a function of both speed and normal acceleration. In turning flight, speed is a function of drag and of thrust to weight ratio, and normal acceleration is a function of wing loading, angle of attack, and speed. Once the desired wing loading and thrust to weight ratio have been established for the design configuration, the maximum turn rate capability can be determined.

Increased turn rate can be obtained at a given flight condition by increasing load factor or by holding load factor and bleeding off energy (speed). The maximum usable lift of an aircraft occurs at that angle of attack where the speed is decreasing faster than the lift is increasing and consequently both normal acceleration and turn rate fall off. Increases in angle of attack above this value not only result in excessive speed loss, with reduced turn rate, but put the aircraft in an area in which it is subject to loss of control. Since loss of control is not normally considered a useful tactical maneuver, increasing the angle of attack beyond the maximum usable value puts one more task on the pilot that detracts from his ability to visually track his opponent-Monitoring Angle of Attack.

Maximum F-16 turn performance occurs in the 24-26 degree angle of attack range; therefore, the angle of attack limiter is set at 25 degrees. Incorporation of the angle of attack limiter in the F-16 allows the pilot to consistently find his maximum turn performance condition and use it without having to take his concentration off his opponent for fear of losing control of the aircraft. The only other operational factor which requires the pilot to monitor the aircraft state during combat conditions is the aircraft structural limits. Incorporation of the "g" limiting feature frees the pilot from having to monitor normal acceleration and thus leaves him totally free to concentrate on his opponent.

We firmly believe that incorporation of the angle of attack and "g" limiting features has given the pilot an aircraft that he can use up to its maximum performance capability without fear of loss of control or overstressing the aircraft. We feel that these kinds of devices do not limit performance. They actually enhance it by allowing the pilot complete freedom to use his maximum performance conditions.

SUPPORT AND OPERATIONS

The F-16 flight control and air data systems will achieve high sortie rates with low support cost because supportability has been designed into them. Design emphasis was placed on obtaining high reliability, low maintenance cost, ease of operation and support, safety, high survivability and parts standardization. These goals were achieved by using parts with high reliability, reducing maintenance time, reducing vulnearable areas, maximizing use of off-the-shelf components and standard parts, and by the incorporation of functional redundancy.

Reliability. Reliability of the flight control and air data systems has been achieved through the emphasis placed on those design considerations given in the principal areas of redundancy, the use of high reliability parts and the incorporation of environmental control measures.

Multiple redundancy, backup systems and alternate modes of operation are provided for those highly critical functions that could, if lost, result in loss or damage to the aircraft. Some examples of this type of feature are the following:

1. Quadrex fly-by-wire electronics and sensors in all control axes.

2. Triplex integrated servoactuators on each of the five primary and control surfaces.

3. Triplex sources for air data parameters.

4. Dual hydraulic power sources with emergency pumping capability.

5. Normal and standby 115-VAC power generation.

6. Dual converters for 28-VDC power supply.

To preclude safety-related failures, the F-16 flight control system has designed-in multiple redundancy from input to output.

Equipment life and reliability of the F-16 electronics are significantly improved by the interface of the environmental control system with the electronic equipment. Results from F-111 experience and from trade studies conducted during the F-16 preliminary design phase show that operation of electronic equipment at temperature extremes increases failure rates and broad temperature excursions cause cyclic stress and induce mechanical failure due to fatigue. The environmental controls identified below are built into the F-16 to prevent such failures.

1. Cooling air at a constant supply temperature of $35^{o}F \pm 2^{o}$ for subsonic flight conditions.

2. Cooling air at a supply temperature of $0^{o}F \pm 5^{o}$ for supersonic operation.

3. Airflow to equipment regulated by a flow control set which provides a near constant cooling effect with exhaust air temperatures that do not exceed $+150^{o}F$.

4. Positive equipment-bay ventilation with cabin exhaust air to minimize variations in bay temperatures resulting from aerodynamic heating effects and intermittent equipment operation.

The F-16 cooling-air-supply temperature characteristics which result in reduced temperature variation and reduced potential for thermal shock are depicted in Figure 19.

<u>Maintainability</u>. Maintenance costs and maintenance downtime of the flight control and air data system are designed-in by providing readily accessible equipment, providing built-in tests, eliminating adjustment requirements and simplifying repair procedures.

<u>Accessibility</u>. The F-16 access doors and covers to the flight control and air data system equipments are optimized as a result of YF-16 maintenance experience and evaluation of F-16 equipment maintenance requirements. Much of the mechanical and electronic equipment is functionally grouped in compartments with large, hinged access doors. As a result, the need to remove in-the-way components is virtually eliminated.

<u>Fault Isolation/Self Test</u>. Historically, the flight control system is one of the most important contributors to maintenance cost; therefore, this system has received special design attention for maintainability improvement. As a result, the system incorporates extensive built-in test capability for automatic fault-detection and self-test. Also, system components are readily accessible under large, fast-opening, hinged doors and are readily replaceable without adjustment. This accessibility, illustrated in Figure 20 was demonstrated during prototype flight test when the flight control computer was removed, replaced and checkout ready for flight in only 21 minutes. The flight control system incorporates extensive use of fault isolation and self-test. It detects malfunctions, identifies which axes and channels are affected, identifies which servoactuator has malfunctioned, utilizes failure indicators on the flight control computer, includes special fault-isolation logic for use by maintenance personnel, and includes automatic self-test to verify proper system operation.

<u>Simplified Repair and Support</u>. The primary flight control system is repaired by replacement of interchangeable components without a need for system alignment or adjustment after repairing any component. The extensive fault isolation and self-test capability reduces flight control system maintenance personnel skill requirements and improves the probability of successful system repair. The repair of electronic components at the intermediate level by replacement of circuit boards also reduces maintenance personnel skill level requirements. Because of the built-in test capabilities, the need for special test equipment for troubleshooting and functional check is eliminated. Also, no maintenance platforms are required since all preflight, turnaround, and post-flight inspection

items are accessible from ground level. The overall result of these economies is a flight control system requiring a minimum of support equipment. This reduction in support equipment decreases support cost and facilitates deployment of the airplane to, and its operation from austere bases.

System Safety, Survivability and Vulnerability. The principal features of the flight control system contributing to these characteristics are delineated briefly below.

System Safety. The F-16 is designed to be a safe and reliable aircraft. Many features are incorporated in the airplane to prevent and detect hazardous situations and to reduce accidents. Special emphasis was given in the design of the flight control system to minimize the probability of loss-of-control accidents by (1) permitting the use of high angles-of-attack without divergence, and (2) automatically limiting the maximum angle-of-attack to maximum usable combat values. The tall vertical tail provides directional control at maximum angles-of-attack and the automatic aileron-rudder coupling provides spin resistance. Thus, the pilot can fly the F-16 to the limits of his physical ability and the aircraft's structural and maneuver capability without fear of losing control. The features enhance F-16 flight controllability and are illustrated in Figure 21.

Survivability/Vulnerability. The F-16 flight control system contains designed-in survivability/vulnerability features that enhance supportability. These features provide improved performance in the important areas of damage tolerance and damage repairability. Through damage tolerance, the F-16 airplane and its control system can sustain hits and damage, return to base, and thus place minimal support demands for replacement aircraft. Through damage repairability features, minimal requirements are imposed for repair assets in terms of materials and manpower.

The damage tolerant design of the flight control system is the result of vulnerability reduction through proper materials selection, redundancy, and separation of critical components; shielding of critical components, concentration of critical components, etc. The fly-by-wire flight control system is quadruple-redundant and the hydraulic system is dual-redundant. Both systems are separated to the maximum extent. In addition, the emergency power system is a redundant source of hydraulic and electrical power should other aircraft power sources be destroyed. This emergency power system assures flight control during air-start of jet-fuel-starter-assisted engine start attempts. And, finally, the F-16 can be successfully controlled and landed with the engine shut down.

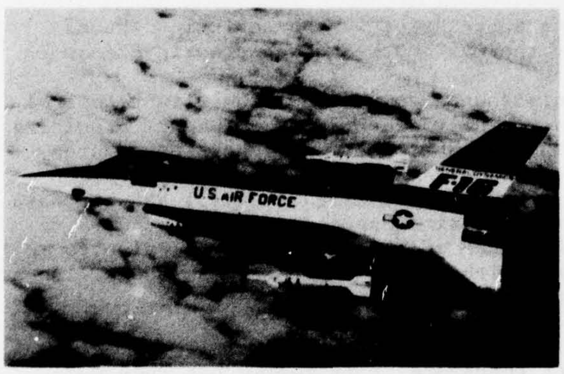

Figure 1 The Multi-National F-16 Air Combat Fighter

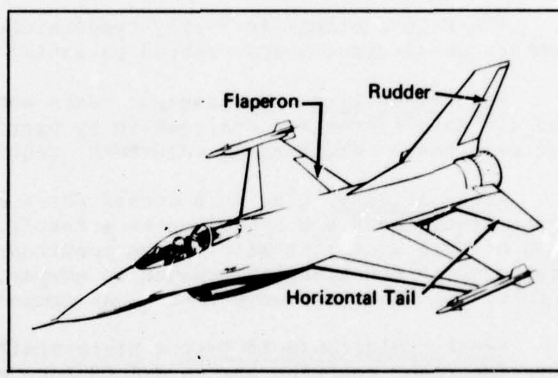

Figure 2 Primary Control Surfaces

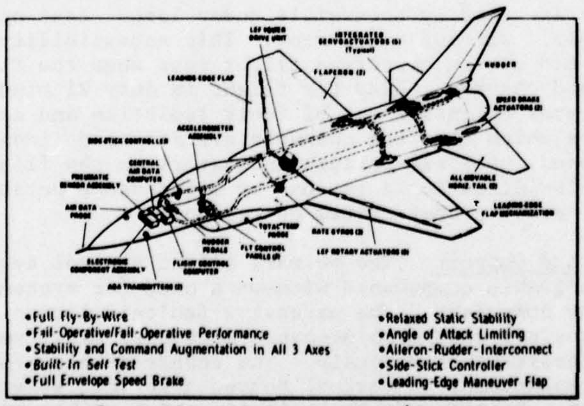

Figure 3 F-16A Flight Control/Stability and Control Features

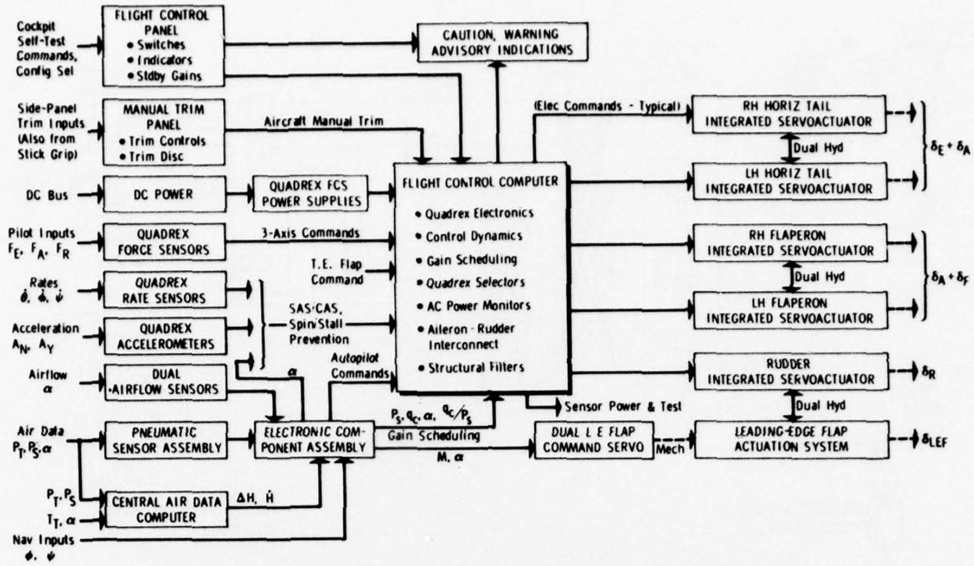

Figure 4 Flight Control System Functional Schematic

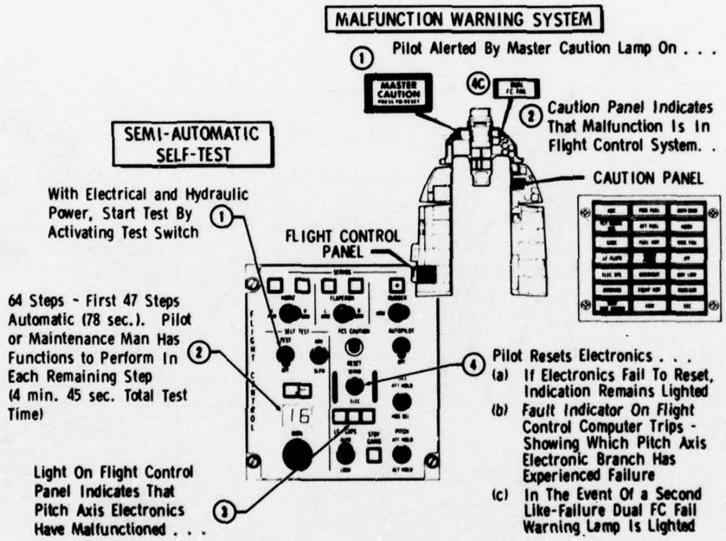

Figure 5 Flight Control Warning and Semi-Automatic, Self-Test System

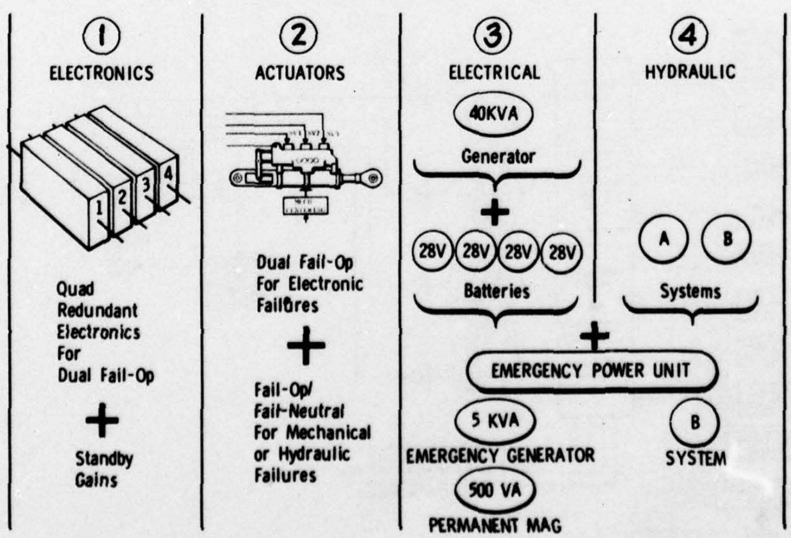

Figure 6 Flight Control System Redundancy Concept

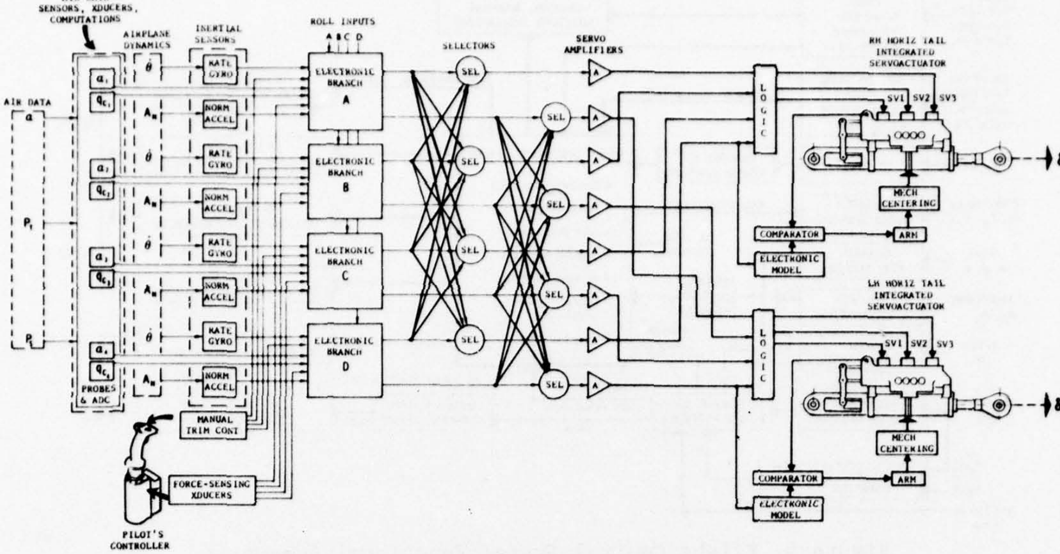

Figure 7 Pitch Functional Schematic Diagram

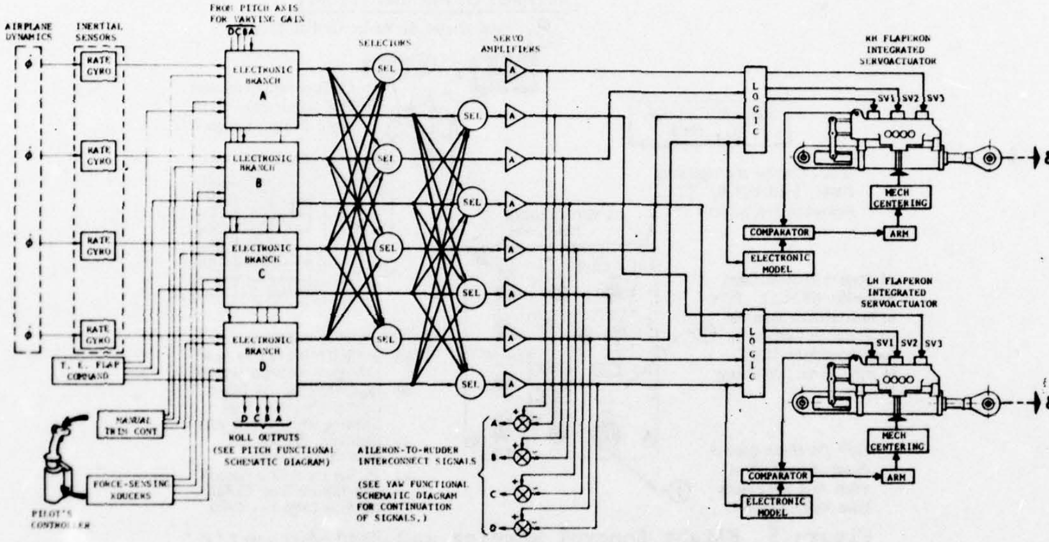

Figure 8 Roll Functional Schematic Diagram

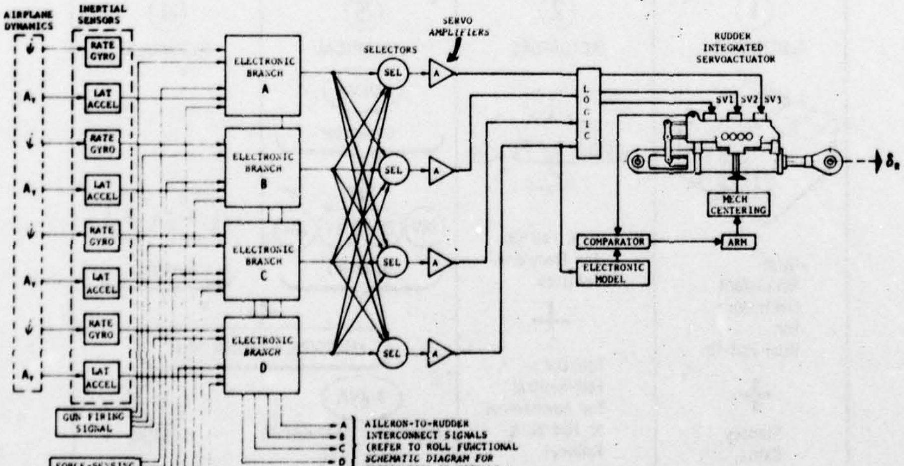

Figure 9 Yaw Functional Schematic Diagram

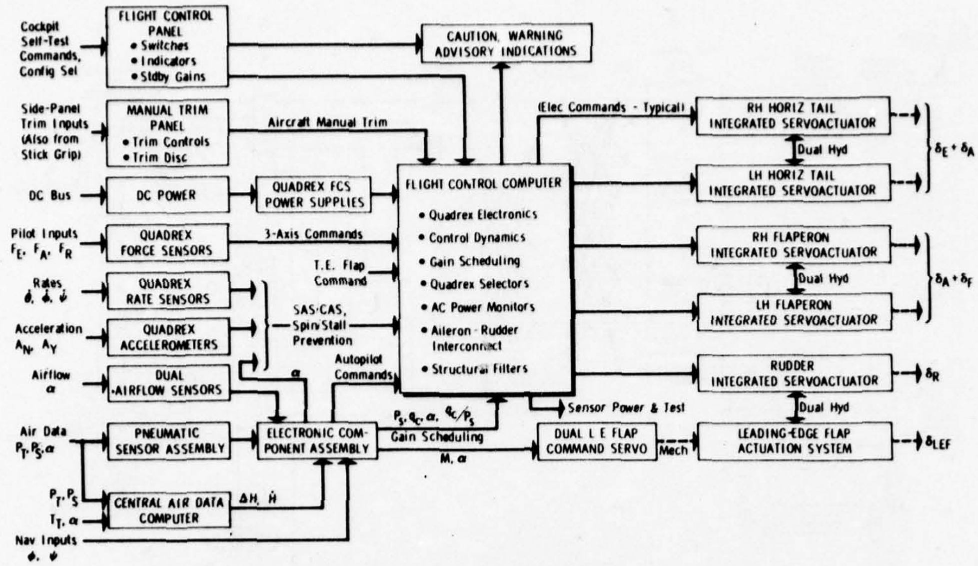

Figure 4 Flight Control System Functional Schematic

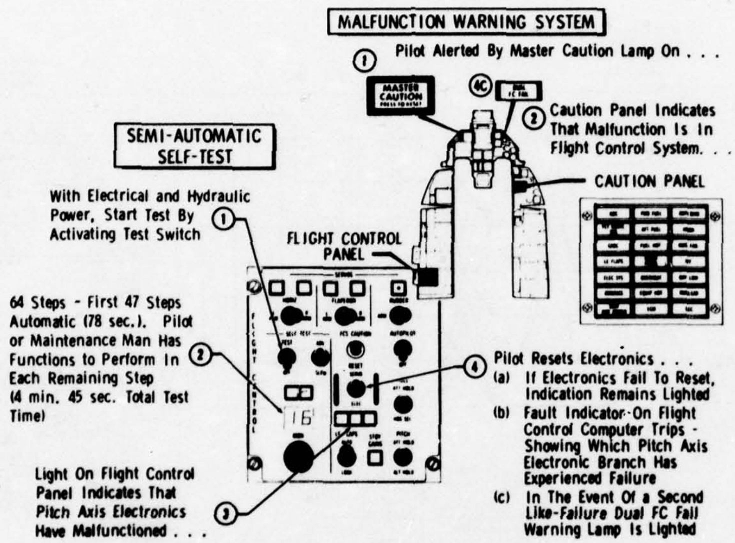

Figure 5 Flight Control Warning and Semi-Automatic, Self-Test System

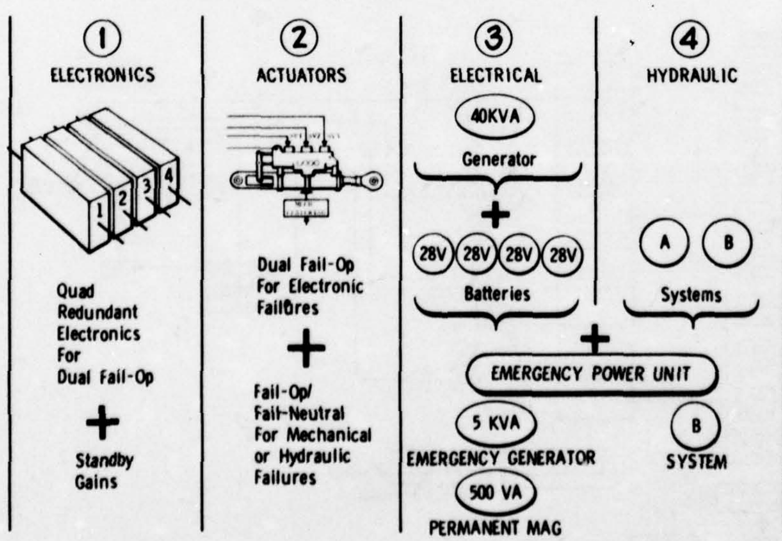

Figure 6 Flight Control System Redundancy Concept

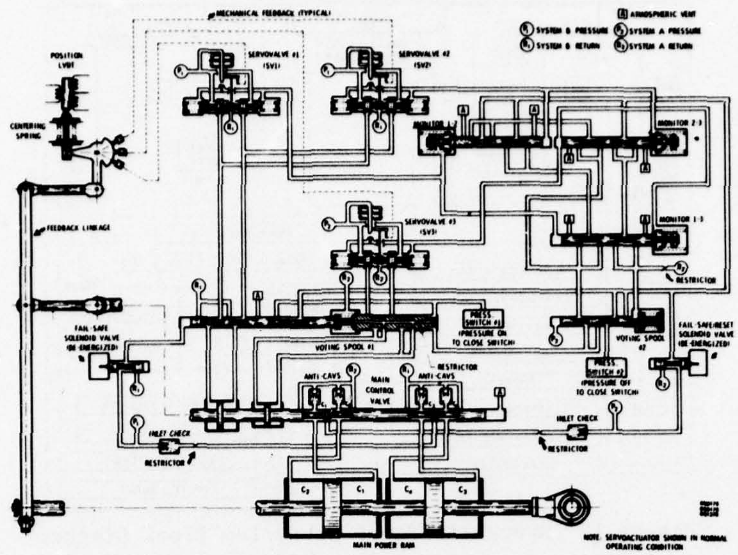

Figure 10 Integrated Servoactuator Schematic

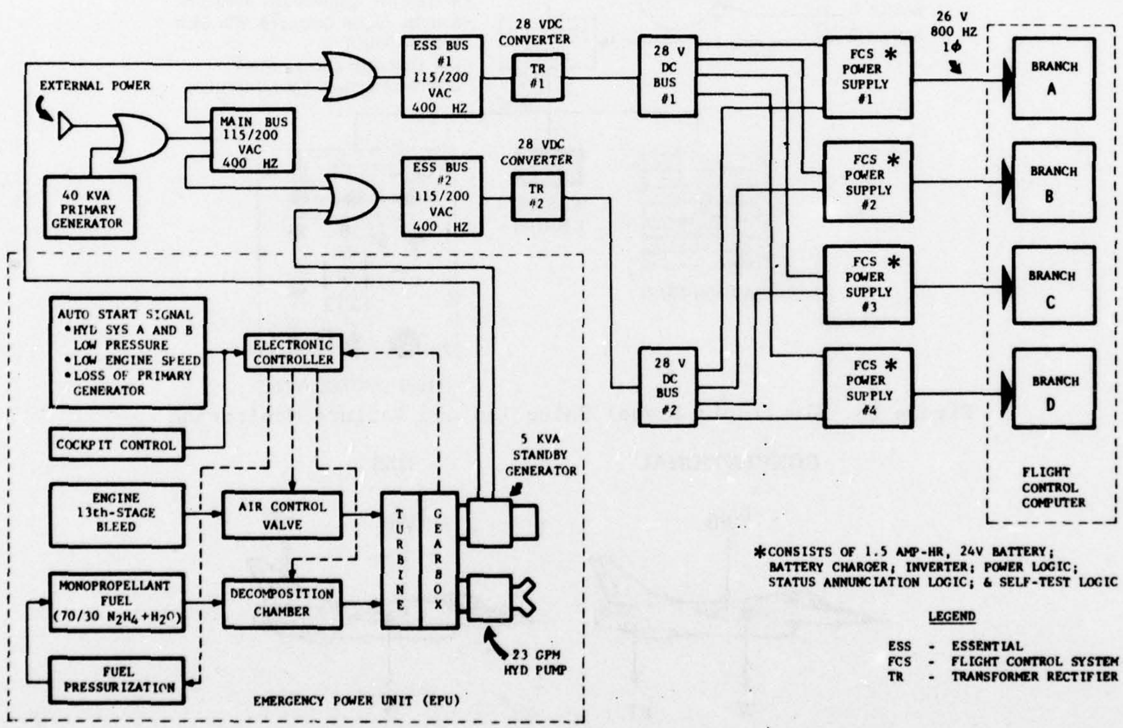

Figure 11 Flight Control System-Electrical Power Schematic Diagram

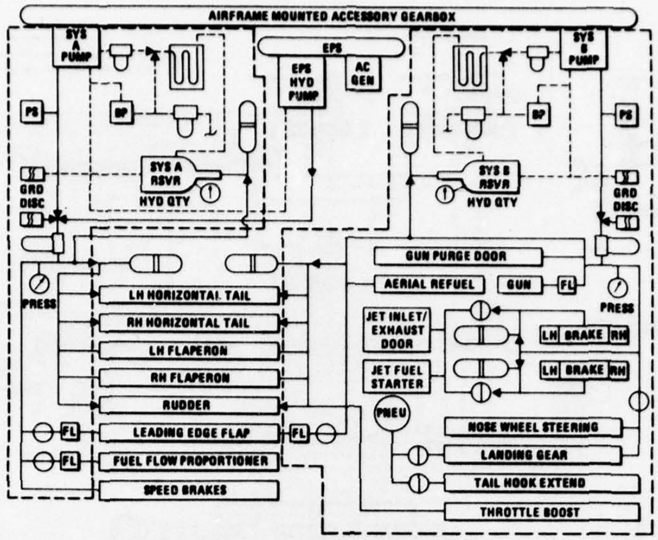

Figure 12 Hydraulic Power Subsystem Block Diagram

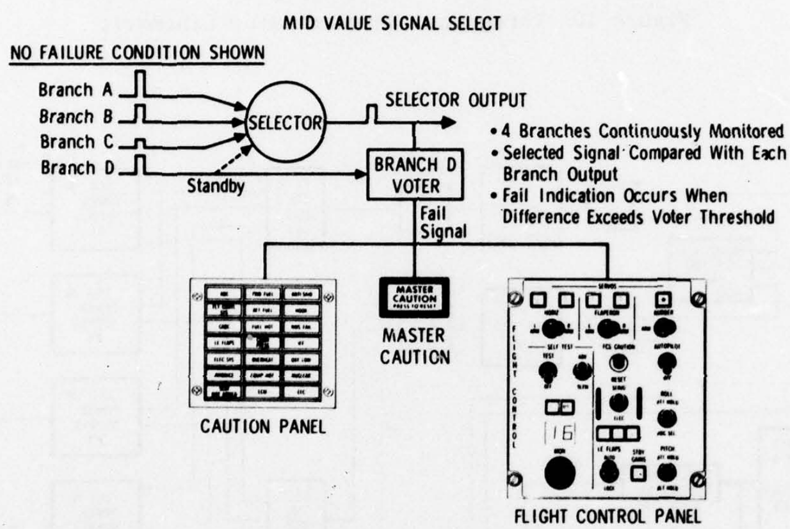

Figure 13 Electronic Signal Selection and Failure Monitoring

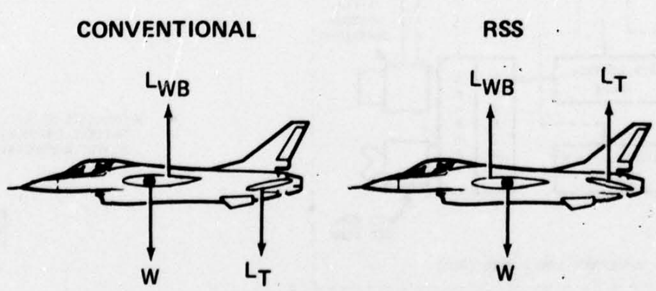

Figure 14 Subsonic Balance Comparison

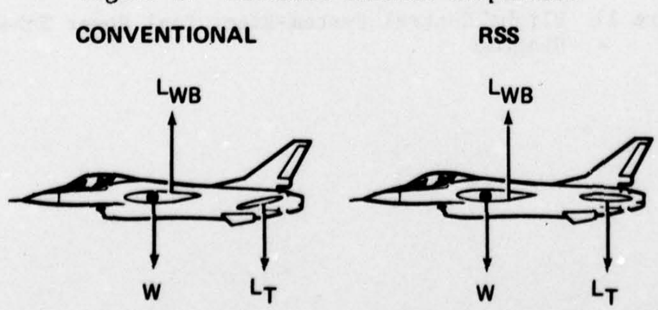

Figure 15 Supersonic Balance Comparison

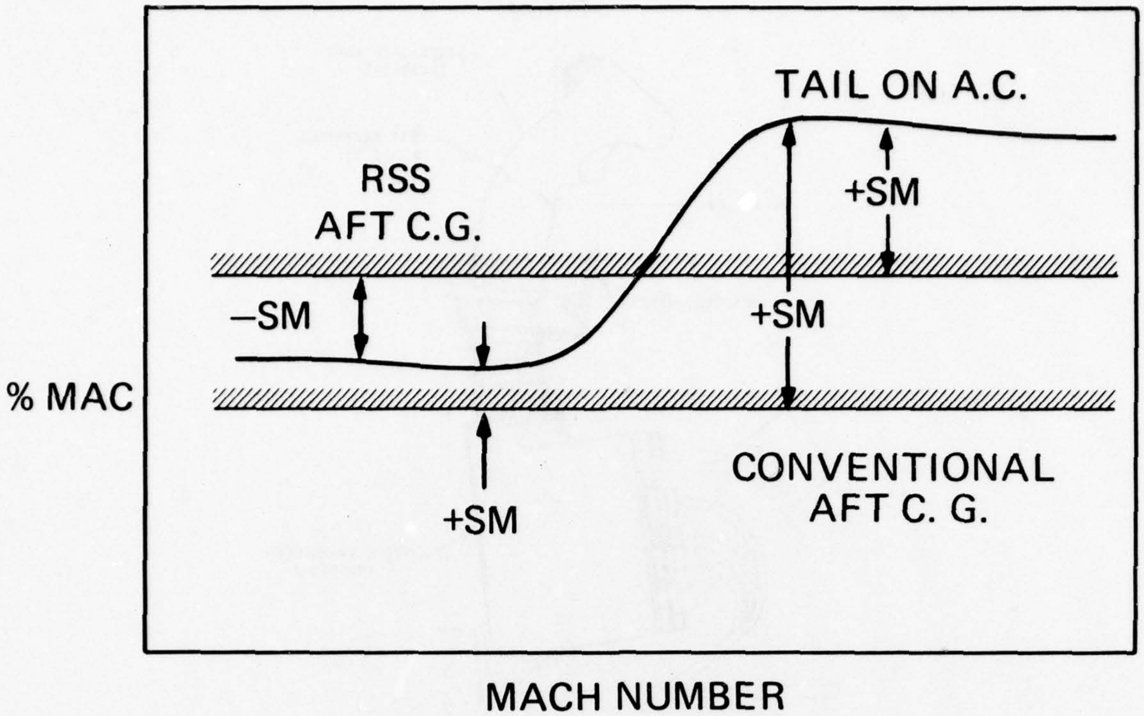

Figure 16 Aerodynamic Center Variation

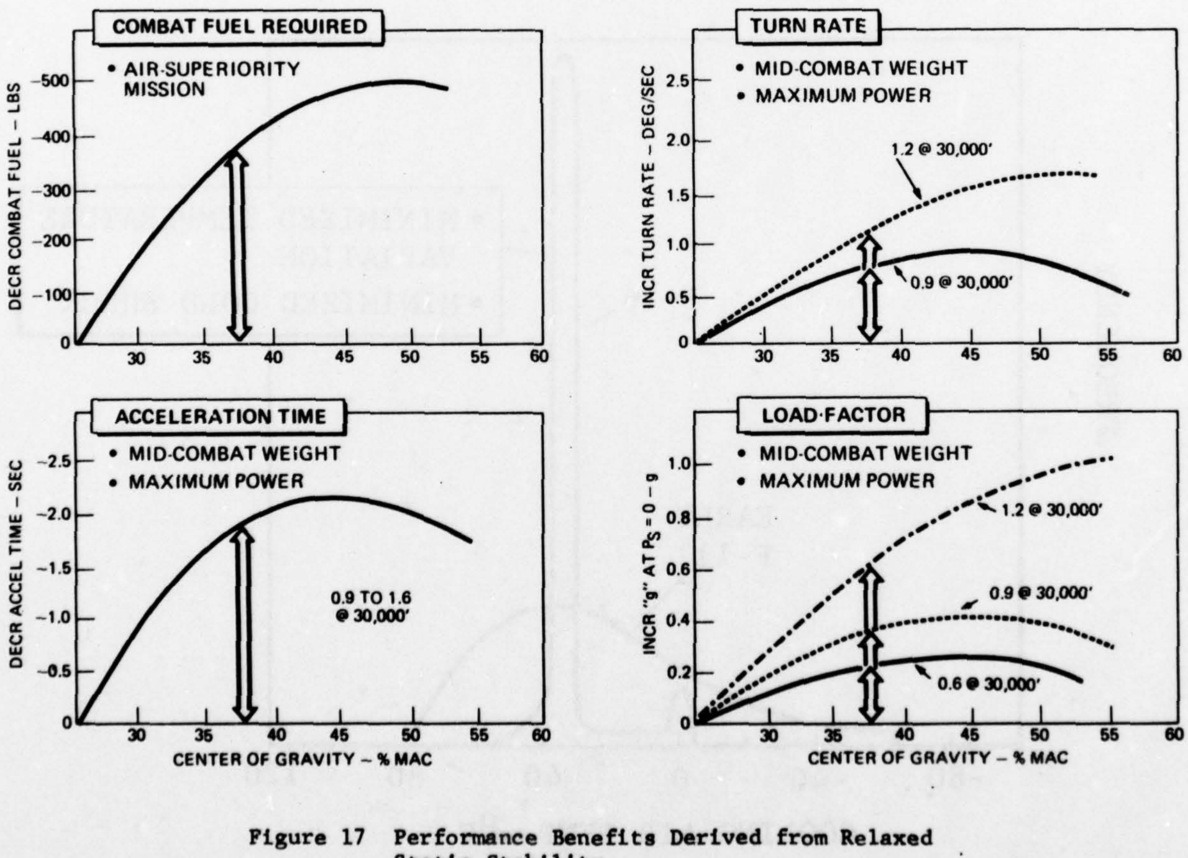

Figure 17 Performance Benefits Derived from Relaxed Static Stability

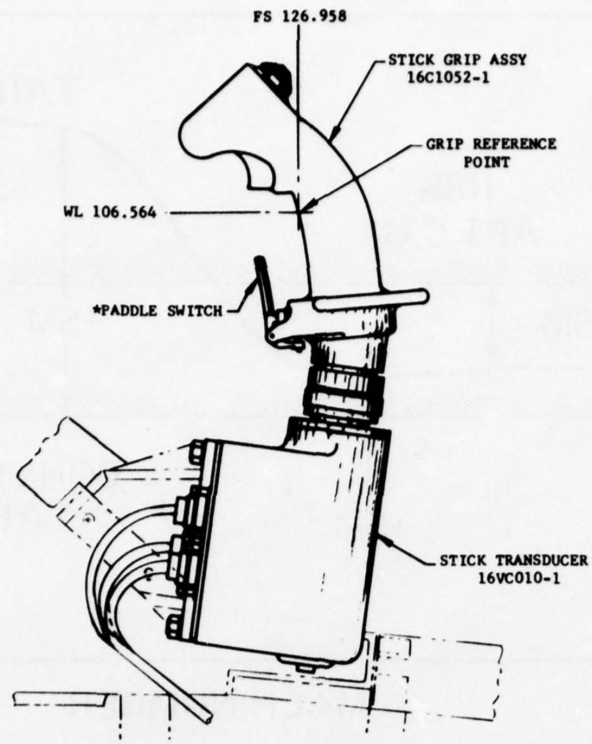

Figure 18 Pilot's Side-Stick Controller

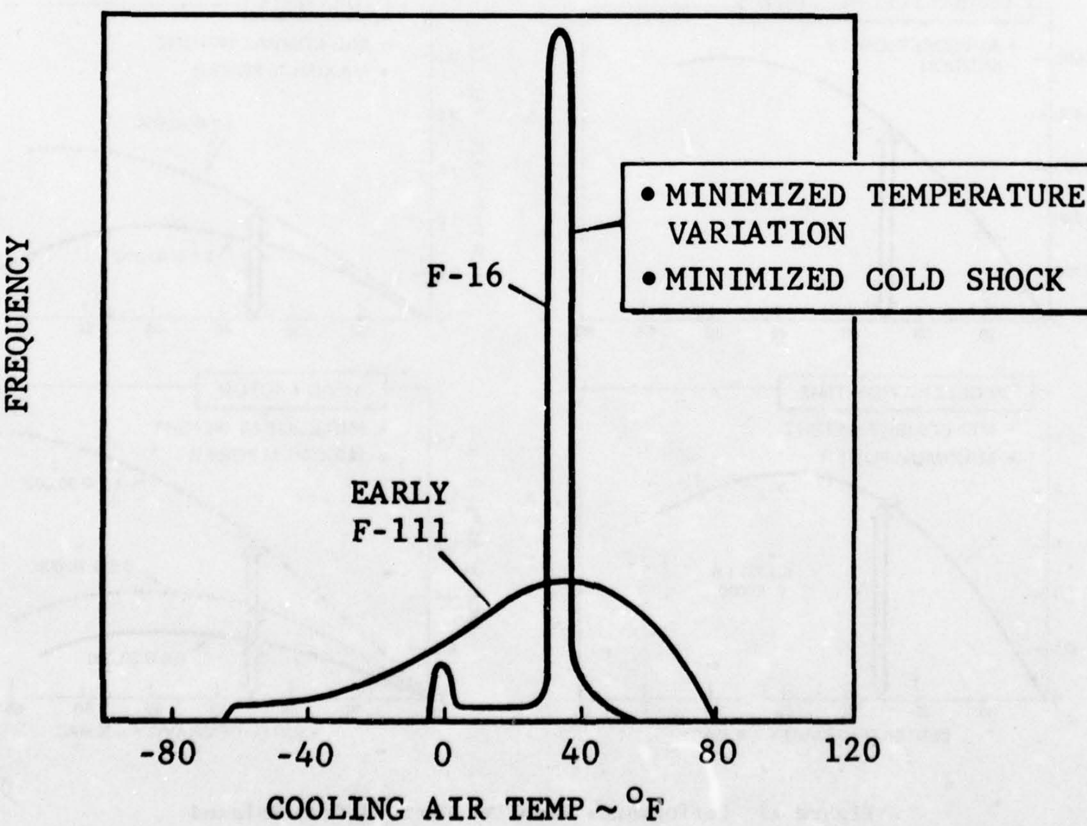

Figure 19 F-16 In-Flight Cooling Air Supply Temperature Characteristics

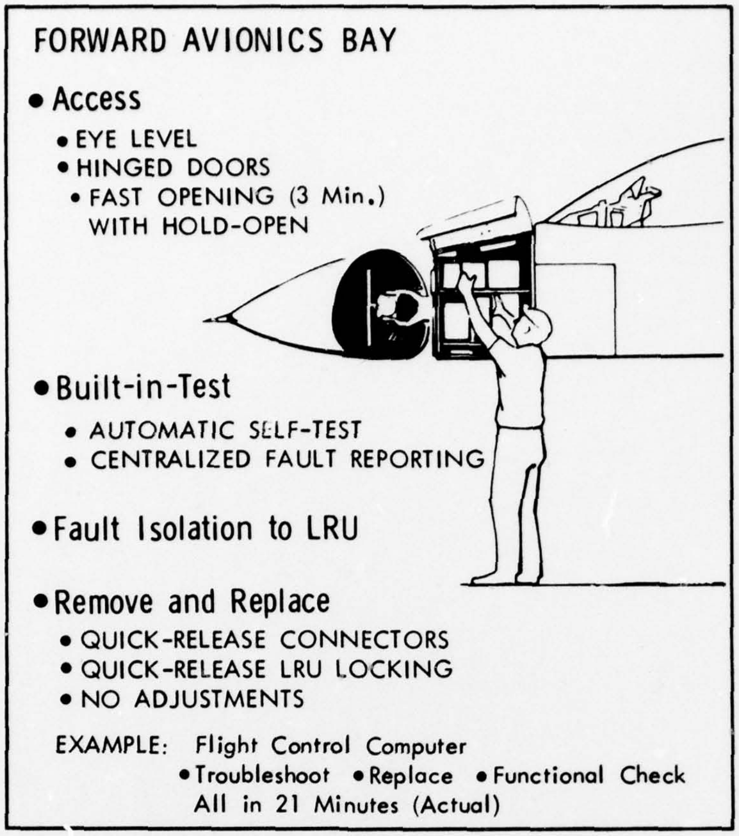

Figure 20 Avionic Equipment Accessibility

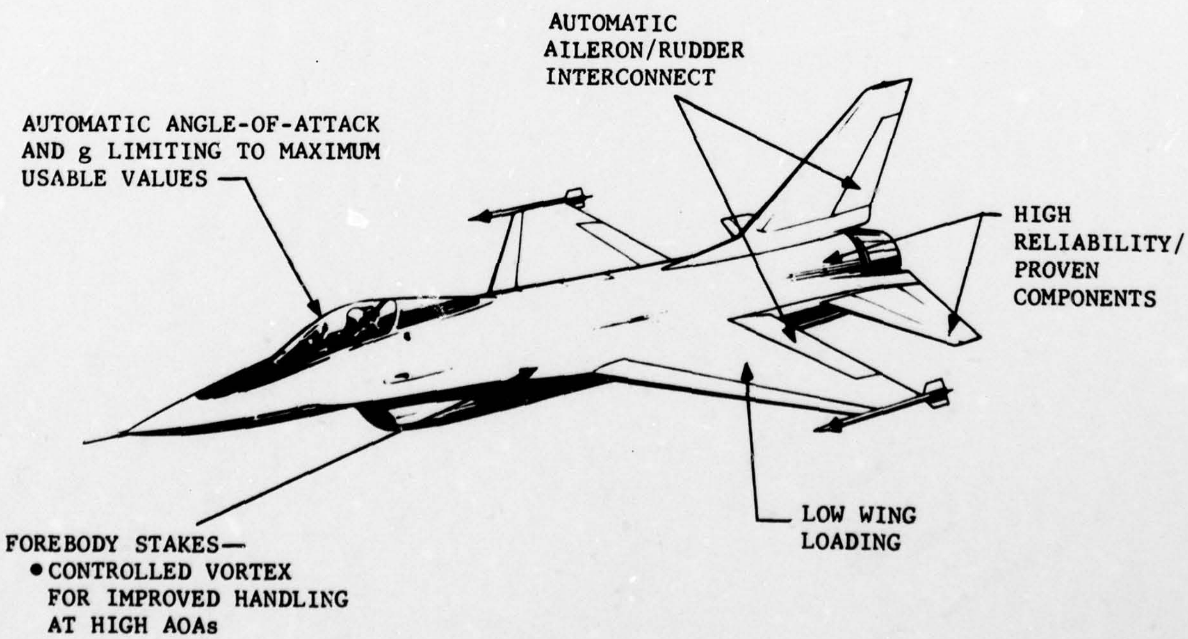

Figure 21 Flight Control Features that Improve Safety

F-8 ACTIVE CONTROL

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SUMMARY

NASA is conducting an advanced flight control research program with a modified F-8C aircraft. The primary objective of this program is to provide flight experience and a design data base for future practical flight-critical control systems. Key technologies being investigated include system redundancy management and active control laws. Two control law packages have been proposed for flight test. The first is the Control Configured Vehicle (CCV) package which incorporates command augmentation, boundary control, ride smoothing and maneuver flap functions. The second package is an adaptive control law based on a parallel channel maximum likelihood estimation algorithm. This paper describes the design, implementation and flight test experience with both sets of control laws.

LIST OF SYMBOLS

c_i	-	parameter used in F-8 parameterization
C^*	-	response variable $n_z + V_{co}q$
D	-	measurement matrix
F	-	plant matrix
g	-	acceleration due to gravity
G_1, G_2, G_3	-	input matrix due to control, gusts, trim
H	-	measurement matrix
K_i	-	control system gains
L_i	-	likelihood function for i-th filter
n_z	-	normal acceleration
n_y	-	lateral acceleration
n_{z_p}	-	normal acceleration at pilot
p	-	roll rate
P_s	-	specific excess power
q	-	pitch rate
$\bar{q}$	-	dynamic pressure
r	-	yaw rate
s	-	LaPlace operator
T	-	sampling interval
u	-	control input
V	-	true airspeed
w	-	wing disturbance
x	-	state vector
y	-	measurement
Z^{-1}	-	delay operator
RS	-	ride smoothing
MF	-	maneuver flap
CCV	-	Control Configured Vehicle
DFBW	-	Digital Fly-by-Wire
PCMLE	-	Parallel Channel Maximum Likelihood Estimation
CAS	-	Command Augmentation System
RAV	-	Remotely Augmented Vehicle
MLE	-	Maximum Likelihood Estimation
SAS	-	Stability Augmentation System
KIAS	-	Knots Indicated Airspeed
SIGSQ	-	residual parameter in PCMLE
PSS	-	Positive Speed Stability
α	-	angle-of-attack
β	-	angle-of-sideslip
$\delta e, \delta a, \delta r$	-	elevator, aileron, rudder surfaces
$\delta e_p, \delta a_p \delta r_p$	-	surface command
ϕ	-	roll attitude
θ	-	pitch attitude
ψ	-	yaw attitude
$\frac{\Delta\beta}{K}$	-	turn coordination parameter
ν	-	Kalman filter residual
∇	-	gradient operator
∇^2	-	second partial derivative matrix
M_α, M_δ	-	pitch moment due to indicated variable
V_{co}	-	cross-over velocity in C^*
Z_α, Z_δ	-	normal force due to indicated variable

1.0 INTRODUCTION

The National Aeronautics and Space Administration, in cooperation with industry and various academic institutions, is conducting research in the area of advanced flight control systems. As part of this research program, the Hugh L. Dryden Flight Research Center is currently flight testing a full authority fault-tolerant digital fly-by-wire (DFBW) control system[1-2] using an F-8C testbed aircraft (Figure 1). The primary objective of this program is to provide flight experience and a design data base for future practical flight-critical control systems. The key technology areas being investigated are systems redundancy management, and advanced control law concepts. This paper addresses primarily the control law aspects of the program.

The F-8C DFBW program has two major control law research objectives. The first is to examine the design, mechanization, and performance of an integrated set of control laws which would be typical of those for projected aircraft employing full-time active controls. The potential benefits of active controls in an all-new aircraft design are a matter of continuing study and experimentation[3-4]. The actual mechanization of active control laws in the fault-tolerant full authority F-8C DFBW control system was intended to provide insight into the control law design and implementation techniques applied to a realistic advanced control system.

The second objective is to exploit the computational advantages of a digital computer in the implementation of advanced control techniques. It is no longer unreasonable to consider control law structures which require the real time solution of modern estimation or filtering algorithms. A control law approach was selected which would not only tax the digital computer's capability, but also would provide benefits not attainable otherwise.

These objectives led to two control law packages for the test aircraft. The first is a "Control Configured Vehicle (CCV) Package,"[5] which incorporates command augmentation, boundary control, ride smoothing and maneuver flap functions, all designed for simultaneous operation. The second is an "Adaptive Control Law Package,"[6-7] utilizing a real-time parallel channel maximum likelihood estimation (PCMLE) algorithm to identify key aircraft parameters and continuously adjust control system gains as a function of aircraft flight condition and configuration without the use of air data parameters.

The F-8C aircraft was flown with the DFBW system for the first time in August 1976, utilizing the CCV control laws. The adaptive control law software has been developed and is undergoing final testing prior to flight test. This paper describes the design implementation and test experience with both sets of control laws.

Section 2 describes the design criteria and ground rules for the CCV control laws. Section 3 presents the design procedure and Section 4 the implementation of the CCV package. The CCV flight test program is summarized in Section 5. Section 6 presents the adaptive control structure. The parallel channel maximum likelihood estimation (PCMLE) algorithm is presented in Section 7. Its implementation for flight test is described in Section 8. The PCMLE performance on a simulator and with flight data is described in Section 9. Section 10 presents Conclusions.

2.0 DESIGN CRITERIA AND GROUND RULES FOR THE CCV CONTROL LAWS

The CCV control laws were to emphasize control configured vehicle benefits for fighter aircraft. Specific pitch axis objectives were improved handling qualities, angle-of-attack limiting, gust alleviation, drag reduction in steady and maneuvering flight, and a capability to fly with reduced static stability. The lateral-directional design objectives were improved Dutch roll damping and turn coordination over a wide range in angle-of-attack. An overall program objective was to explore the use of modern control design methodology to achieve these specific CCV benefits. Other objectives and ground rules are summarized below.

2.1 Control Surfaces and Actuators

The control laws were constrained to be compatible with the existing airframe without structural modification. Hence, the control laws use only existing elevator, rudder, and aileron as control effectors, each powered by existing actuators. Altered control surfaces or new force producers were not considered. However, the ailerons are commanded symmetrically to provide an additional control input in the pitch axis.

2.2 Sensors

The control laws were also constrained to use available sensors. These include pitch, roll, and yaw rate gyros, vertical and directional gyros and normal and lateral accelerometers. Air data measurements included angle-of-attack, Mach number, and pressure altitude.

2.3 Computer Capacity and Sample Rate

Because control law calculations typically consume only a small fraction of the total computational load in a flight computer (the majority is I/O, self test, mode and redundancy control), a general design guideline was that the control laws should be structured so as to use no more than 25 percent of the total available time. For the prespecified sample rate of 50 per second, this resulted in about 5 milliseconds of available frametime. The sample rate is also high enough to produce no substantial differences between direct digital design (discrete-time control laws designed for discrete-time models) and continuous time design with after-the-fact discretization.

3.0 THE CCV DESIGN PROCEDURE

The major steps in the design process involved linear modeling, control law synthesis with optimal control theory, digital controller analysis, and design verification on a nonlinear six-degree-of-freedom simulation. The study utilized existing software design tools in the first three areas. The last task was conducted using the DFBW F-8C simulators at NASA Langley and Dryden Research Centers.

3.1 Modeling

Nonlinear aerodynamic data for the F-8C as a function of Mach, altitude, angle-of-attack, and surface position was used to obtain linear models. The aircraft equations of motion were trimmed numerically at selected operating conditions. Numerical differentiation, based on small perturbations in each of the states and controls, was used to compute the system matrices. The actuator models and gust filters were then appended to complete the modeling process.

Linear models were obtained at 25 flight conditions defined by the nominal altitude, Mach, weight, geometric configurations and load factor. These models have the form:

$$\dot{x} = F(\underline{c}_i)x + G_1(\underline{c}_i)\delta + G_2(\underline{c}_i)w + G_3(\underline{c}_i)$$

$$y = H(\underline{c}_i)x + D(\underline{c}_i)\delta, \qquad i = 1, 2, \ldots, 25$$

where x denotes 12 aircraft states, δ denotes three control surface deflections, w denotes wind gust disturbances, and y's are assorted outputs to be measured or controlled. The matrices F, G_1, G_2, G_3, H, and D can be treated as continuous functions of the slowly changing parameters (c_i).

3.2 Control Laws

The longitudinal and lateral control laws were separately designed with modern linear quadratic optimization techniques to satisfy conventional design requirements. A quadratic optimization problem was solved at each flight condition. The resulting control laws were simplified and gains were approximated as functions of air data measurements to obtain gain schedules.[5]

3.2.1 The Longitudinal Axis---

3.2.1.1 Normal Elevator Control Mode--This mode was designed to satisfy the following performance specifications:

1) Command Augmentation Functions

   a) C* step response
   b) Stick gradient to meet MIL-F-8785 [8]
   c) Neutral speed stability

2) Regulator Function

   Reasonably damped gust responses (short period damping ratio > 0.3)

3) Tolerance for Uncertainties

   a) Classical loop gain margin greater than 6db and phase margin greater than 30 degrees
   b) Loop attenuation at high frequencies (< -20 db at 8 Hz)

These requirements are obviously not in the quadratic cost function form required by linear-quadratic optimization theory. Hence, an important step in the design process was to re-interpret them in more suitable terms. The reinterpretation rationale is discussed briefly below.

Consider the command augmentation requirements first. Item (1a) in this group is a dynamic response specification on C*, a linear combination of pitch rate and normal acceleration;

$$C^* = n_z + V_{co}q, \qquad V_{co} = 98.8 \text{ m/s}$$

This criterion was chosen because it permits the designer to control one response with one forcing function (the elevator). At high dynamic pressures, the elevator produces primarily normal acceleration, and at low dynamic pressures, it produces primarily pitch rate; hence, the sum of these responses produces a composite variable that has significance at all flight conditions.

In response to step commands from the pilot's control stick, the C* variable must stay within a specified response envelope whose center is closely approximated by the step response of a second-order linear model with $\omega \simeq 7$ rad/sec, $\zeta = 0.9$. This requirement was incorporated in the optimization framework by the common technique of appending an explicit "command model" and penalizing errors between it and the aircraft.[10, 11] Specification Item (1b) was then used to set the DC gain (acceleration per unit stick force) of the model, and Item (1c) was satisfied by penalizing an appended integral of the model-following error as well as the error itself. The latter approach is motivated by the fact that "neutral speed stability" is actually imposed in order to provide command insensitivity to trim changes.

A low pass filter was applied to the normal acceleration signal used to construct C*. This filter provides high frequency attenuation, Item (3b), which is required to assure adequate stability margins in the presence of uncertain servo characteristics and unmodelled flexure dynamics of the airframe. Loop transfer functions of trial designs without such filters tended to drop off too slowly to meet the specification.

Based on the above rationale, the criterion function was taken as a weighted sum of four quadratic terms--model-following error, $(C^* - C^*_c)$, integrated model-following error, $\int(C^* - C^*_c)dt$, elevator rate, $d/dt\,\delta_e$, and elevator command, δ_c. Quadratic weights were then selected iteratively until the resulting control law satisfied all performance specifications.

The final C* controller can accommodate reduced static stability caused by shifting the center of gravity aft. Variations in the center of gravity from 29 percent mean-aerodynamic-chord (nominal static stability margin) to 48 percent (very unstable) were studied. Such dramatic variations in the center of gravity are not experienced under normal loading conditions. Instead, they are meant to represent different airframe designs. The mechanization can accommodate a wide range of static margins without modifying the basic C* control loop.

3.2.1.2 Angle of Attack Limiting Mode--A separate elevator controller was designed to hold a reference angle-of-attack. This boundary controller, also designed with linear-quadratic optimization theory, provides proportional plus integral action on the angle-of-attack error and uses pitch rate feedback to damp the short-period dynamics sufficiently to prevent boundary overshoot. The criterion function was a weighted sum of squared errors in angle-of-attack, integrated angle-of-attack, elevator rate, and pitch rate.

3.2.1.3 Symmetric Aileron Control Mode--Two sub-modes were provided in the CCV control laws for commanding symmetric aileron deflection. They are the maneuver flap (MF) and ride smoothing (RS) mode. The first mode provides a steady state minimum drag schedule for the flaps. The second combines elevator and symmetric ailerons dynamically to provide direct lift in addition to drag reduction.

3.2.1.3.1 Maneuver Flap Mode--The MF mode was designed to position the symmetric ailerons for minimum drag during maneuvers. The schedule was determined from the nonlinear F-8C simulation by computing the drag with various symmetric aileron settings. The schedule was implemented as a function of steady-state pitch rate since at zero pitch rate no deflection is desired. A further advantage is that this schedule provides minor gust alleviation and thus is compatible with the RS mode. The schedule is implemented with a two second lag filter on pitch rate to effectively decouple the maneuver flaps from the short period dynamics.

3.2.1.3.2 Ride Smoothing Mode--The RS mode was designed to reduce rms acceleration due to gusts and to enhance the pitch response through a combination of symmetric flap and elevator deflection. A significant reduction in the rigid body acceleration due to gusts can be realized if lift can be produced "directly" rather than by changing the angle-of-attack with the elevator. Utilization of direct lift also provides "command augmentation" whereby the amount of pitch rate overshoot required to obtain a fast n_z response is reduced.

Implementation of direct lift required a flap-to-elevator crossfeed to compensate for the pitching moment of the symmetric ailerons. This crossfeed is easily computed by comparing the relative magnitudes of the pitching moment and normal force of the elevator and flaps. Fortunately this ratio is nearly constant for all flight conditions and a fixed crossfeed was adequate. To reduce the gust-induced normal acceleration, a controller structure that commands direct lift as a function of normal acceleration was selected. The gain on n_z was scheduled to keep the loop gain near unity. This gain value will provide about a 50 percent reduction in gust-induced accelerations and does not present any stability problems.

3.2.2 The Lateral-Directional Axes--The objectives of the lateral-directional control laws were 1) good roll rate response (first order with a 0.2 second time constant), 2) Dutch roll damping > 0.19, and 3) good turn coordination (Level I, MIL-F-8785B).

A quadratic optimal controller was designed to follow an explicit roll rate model and to minimize sideslip and lateral acceleration for roll stick commands.

The criterion function was a weighted sum of roll rate model-following error, integral of lagged lateral acceleration, kinematic sideslip rate ($r - p\alpha - g/V\ \phi$), and control effort (aileron and rudder commands).

The resulting design produced good Dutch roll damping and turn coordination over the flight envelope but requires a full complement of sensors (p, r, ϕ, α, V). This structure is often called "inertial turn" coordination since the controller commands the stability axis yaw rate needed to balance centrifugal and gravity forces along the body Y axis. This balanced condition corresponds to a coordinated turn. However, this structure requires very precise attitude and true air speed measurements for implementation. Since these sensors are not available on the test aircraft, the design was simplified to use only roll rate, yaw rate, and lateral acceleration feedbacks, plus an aileron to rudder crossfeed. This simplification was performed by means of fixed form solutions of the original optimization problem.[5]

4.0 CCV CONTROL LAW IMPLEMENTATION

4.1 Functional Block Diagrams

The CCV control design process provided three independent control modes for the pitch axis, one basic mode for the lateral-directional axes, and several outer loops. These were integrated into a simple package, as described briefly below.

The integrated pitch CAS consists of the basic C* elevator controller, plus angle-of-attack limiting plus two additional modes that command symmetric aileron deflection. A block diagram is shown in Figure 2. Specific details are contained in reference 5.

The C* and alpha limiter modes were integrated with a switching strategy which provides smooth transition from normal control to boundary control whenever the pilot commands an angle-of-attack higher than a preset reference limit. The mode transition also protects against unaccelerated stalls. The switching strategy is based on commanded elevator rates. This ensures smooth transitions and eliminates elevator trim problems.

The direct lift function is integrated with the C* mode such that the pitch stick continues to command C*. However, with the RS mode engaged, the improved ride smoothing and N_z responses offered by direct lift are both realized. Stick gains for the RS mode were selected to produce the same steady state flap deflections as the MF mode.

Conventional outer loop modes including altitude hold, attitude hold, and Mach hold were added to the pitch CAS.

A function block diagram for integrated lateral-directional SAS is shown in Figure 3. The roll axis commands aileron position as a function of the difference between the output of a first order model and measured roll rate. The yaw axis provides Dutch roll damping through conventional high-passed yaw rate and turn coordination via the aileron crossfeed and lateral acceleration feedback. Additional details are contained in reference 5.

The lateral axis includes a roll-attitude-hold and a heading-hold outer loop. These loops were designed with classical frequency domain methods.

4.2 CCV Control Law Mechanization

The quadratic optimal control law design process produced continuous domain block diagrams applicable for a single channel system. Two steps were required before the CCV control laws could be implemented in the triplex digital flight control system. The first was the transformation of continuous elements to discrete equations which could be programmed. The second was the integration of the control laws with the other flight software.

4.2.1 Control Law Discretization--Based on experience in the first phase of the F-8C DFBW program[12] it was known that sufficient gain and phase margin could be achieved with a sample period of 30 milliseconds. In fact, the first all-digital simulations of the CCV control law were executed in a 32 millisecond sample period in order to be compatible with the real time simulation executive software. The inner loop sample period for the flight system was chosen to be 20 milliseconds to provide some design margin, especially for the high bandwidth ride smoothing mode, and also to forego potential sample rate quantization effects in the roll axis where high command rates are common.

The outer loop autopilot and gain scheduling functions were executed every four minor cycles, or 80 milliseconds.

The continuous-domain filters were discretized using the bilinear, matched z-transform method of reference 13. This method transforms filters of various forms as shown in Table 1. For a sample period of 20 milliseconds, and for the filters used in the CCV control laws, both this method and the Tustin transform method yield discrete filter coefficients that agree within 1 percent.

Integrators appearing in closed-loop sections of the control law were implemented as

$$\left(\frac{T}{2}\right)\left(\frac{1 + z^{-1}}{1 - z^{-1}}\right)$$

whereas integrators used for trim and other quasi-open loop functions were mechanized as

$$T\left(\frac{1}{1 - z^{-1}}\right)$$

All control law functions are computed in single precision, floating point arithmetic, which is a hardware feature of the airborne computer. The floating point format employs a seven-bit exponent and a signed 24 bit mantissa. Assembly language programming was used for all routines.

With the half-sample frequency of 25 Hz being an order of magnitude higher than the highest bandwidth control loop, no significant sample rate effects were expected or seen. A comparison of the pitch CAS response predicted for a continuous and discrete control system implementation is shown in Figure 4, and illustrates the minor effects of the discrete implementation.

4.2.2 Software Integration--The approach taken in the F-8C DFBW mechanization was to isolate the control laws from the system redundancy management software. The control laws operate as if they were being executed in a single channel system.

The software execution sequence for a 20 millisecond minor cycle is shown in Figure 5. Executive functions, consisting of computer synchronization and computer redundancy management are performed first. Then, all sensor and discrete information is read. The three computers exit the synchronization routine within 50 micro-seconds of each other. This represents the maximum skew between redundant sensor data read. Each computer receives all redundant sensor data. The sensor signal selection algorithm chooses one set of data for the control laws to use. The selected signal is either a mid value or average value for an unfailed triplex or duplex sensor set, respectively.

The control law calculations are divided into two sections. The first part contains only the computations necessary to produce the surface command and display outputs. The second part of the control law software contains filter updates, gain scheduling, and other functions that can be performed after the output commands are computed. The three computers provide bit-identical output commands. Tracking of the analog commands to the actuator electronics is within 1 percent.

The detection and isolation of faulty sensors is accomplished immediately after the control law calculations. Faults are either logged for ground maintenance or annunciated to the pilot, depending on the sensor and level of failure.

The digital data telemetry routine transmits 20 32-bit words to an onboard recorder for post-flight analysis. In the remaining 4 milliseconds, the in-flight computer self-test program performs central processor tests, memory sums, and other checks on computer hardware operation.

Twelve bit analog-to-digital and digital-to-analog converters are used in the F-8C DFBW input-output hardware. This yields the sensor and surface command resolution values shown in Table 2.

One additional feature of the flight software is its restart protection. In the event of a transient interruption in processing in one machine due to a power loss or other cause, the control laws are reinitialized in order to permit continued operation in that channel. Should a channel request a restart, all machines execute a special data exchange routine which provides the interrupted computer with the information required to be initialized. This includes the current surface commands and certain node values, the present control flag states, and mode information. Only the computer being restarted reinitializes itself. From the time a restart request is observed, approximately 10 milliseconds is required to complete the reinitialization.

This restart feature is an important element of full authority, full time control laws which must operate through unforeseen transient problems. This capability was demonstrated on one flight during which a serious I/O transient fault led to a restart. The channel was automatically reinitialized and continued normal operation without any adverse effect.

5.0 CCV FLIGHT TEST PROGRAM SUMMARY

Twenty-two flights have been made by two pilots as of October 1977. The flight software and two triplex systems have accumulated approximately 1500 hours in ground and flight testing since flight qualification. The CCV control laws have been examined at various points in the flight envelope, as follows:

Mode	Flight Condition Extremes
Basic CAS and lateral-directional SAS	160 - 500 KIAS 3000 - 12000 meters
Maneuver Flap	400 KIAS; 5.6g
Ride Smoothing	400 KIAS; 2.0g

No rigid body or structural mode stability problems were predicted or observed during the envelope expansion testing. The control system has been evaluated in instrument flight maneuvers, formation flight, and during mild aerobatic maneuvers.

Because the tracking of the three digital channels is within 1 percent, the actuator and airplane responses are indistinguishable from those which would result from a single channel system.

Although refinements have been and will continue to be made to the control laws, it is notable that no problems were encountered which have required coding changes to be made to the control laws.

5.1 Basic CAS and SAS Mode Flight Experience

The basic inner loop control laws, without flap or autopilot modes engaged, provide good closed-loop response. The pitch CAS step response for two typical flight conditions is shown in Figure 6. The short period response is well damped, and the forward loop integration is apparent in the stabilizer response. The normal acceleration response is nearly identical for the two flight conditions whose dynamic pressures differ by approximately a factor of two.

Pilot comments on the CAS mode in a variety of tasks have been very favorable with few exceptions, the longitudinal control is rated better than a three on the Cooper-Harper scale, as shown in Table 3. Refinements are currently being made to the stick shaping to improve formation flight characteristics.

The lateral-directional responses are very close to those predicted during synthesis. Figure 7 shows the response of the airplane to a pilot-commanded roll stick step input at Mach 0.8 and 12,200 meters. This response is typical of those observed in the flight test program. The Dutch roll is critically damped and the roll time constant is approximately 0.4 seconds. Sideslip generation is small, but the lateral acceleration response at the pilot's station is moderate and has a fairly rapid onset. Pilots observed this as a jerkiness in directional response. This abrupt response is caused by the relatively sharp-edged rudder motion resulting from the aileron-to-rudder interconnect. A first order filter with an 8 rad/sec bandwidth was activated in the interconnect path, and this has alleviated the problem.

Table 4 summarizes measurements taken from the lateral-directional SAS responses. As is evident, the roll response and Dutch roll damping are good everywhere. At some low angle-of-attack conditions, the interconnect gain appears high, producing excessive favorable yaw with the resultant large $\Delta\beta/k$. The interconnect schedule is being modified for low angle-of-attack conditions.

5.2 Active Flap Mode Flight Experience

The active flap modes have been tested to 400 KIAS at 6100 meters. No stability or structural resonance problems have been encountered. Handling quality evaluations have been performed at 300 KIAS and 350 KIAS.

5.2.1 Ride Smoothing Flap--The RS mode flight test results show very good dynamic characteristics, as predicted. The pitch pulse responses of the CAS mode with and without RS are compared in Figure 8. Both systems show adequate damping, acceptable rise times, and little overshoot. The slightly greater initial elevator response with the RS active shows the effect of the flap-to-elevator interconnect.

As yet, the RS system has not been used in turbulence, so direct evaluations of the system's performance in flight are unavailabe.

5.2.2 Maneuver Flap--Flight test results from windup turns with and without MF show that the system operates as designed. Figure 9 shows windup turn time histories for the CAS mode with and without MF. For a 1g load factor change, the angle of attack increases 3.2° without MF and only 0.2° with MF; thus, nearly all of the load factor change in the MF mode is due to flap deflection.

In order to assess the effects of the maneuver flap system on turning performance, measurements of the specific excess power, p_s, were made at various accelerated flight conditions. Figure 10 shows the performance for a typical maneuvering condition, 350 KIAS at 6000 meters. The maneuver flap provides approximately 0.5g increase in the maximum sustained g-level ($p_s = 0$), and at 4.5g shows an increase of 22 meters/second in p_s.

Investigation of wing bending and shear loads during early flight tests showed that although the lateral center of pressure on the wing shifted inboard, wing bending loads and shear loads increased about 15 percent for the same load factor (Figure 11). Analysis indicates that this is due to the substantial down force on the horizontal stabilizer that is required to provide a balancing moment for the negative pitching moment of the flaps. The wing must then produce more lift to maintain a constant load factor; this results in the higher wing bending and shear loads.

5.2.3 Flap Mode Handling Qualities--The handling qualities of the CAS and CAS with the active flap submodes were evaluated by using five precision maneuvering tasks (Table 5). The individual effects of the flap submodes were determined by testing, in turn, CAS, CAS with RS, CAS with MF, and CAS with RS and MF. The CAS handling qualities were generally good to excellent, with only one Cooper-Harper pilot rating[6] poorer than 3 (Figure 12). The RS and MF submodes caused only minor changes in handling qualities, and those changes that were noted were usually improvements. No Cooper-Harper pilot ratings poorer than 3 were recorded for the active flap modes.

5.3 Boundary Controller Flight Performance

The reference angle-of-attack used in the boundary controller was set to values lower than the operational limit for flight test purposes. Performance of the boundary controller has been very good. Figure 13 shows a full aft-stick maneuver of 350 KIAS with the alpha limit set to 12 deg. The angle-of-attack is seen to be hard-limited at 12 deg for this slow approach to the boundary of approximately 0.2 deg/second. The normal acceleration, displaying the effects of buffet, limits at -3.6g (positive pilot g's). Figure 14 illustrates a more rapid approach (1 deg/sec) to the alpha limit, for a limit value of 6 deg. Because of the rapid approach to the boundary, the limit is not quite reached (5.85 deg maximum). There is no overshoot in this maneuver.

6.0 SELF-ADAPTIVE DESIGNS

6.1 Ground Rules

The aim of the adaptive design effort was not to develop new theoretical procedures or algorithms but to turn existing concepts into flight-worthy control laws for the specific test aircraft. Of course, the concepts and design processes should be general enough to apply to other aircraft as well. This is important because the F-8C is not difficult to control with non-adaptive techniques as demonstrated with the CCV package.

The adaptive designs were constrained to operate without air data. This was motivated by the belief that removal of air data measurements from flight control is one of the tangible benefits which adaptive controls can offer for aircraft (like the F-8C) whose basic performance requirements can be satisfied with air-data-scheduled control laws.

The other ground rule concerns test signals. The adaptive system must operate in the presence of normal pilot inputs and also when such inputs are absent. Any test signals required for the latter case must be small enough not to interfere with the aircraft's mission. This generally means that test input normal accelerations, as sensed at the pilot station, should be below 0.02 - 0.03g's rms, and lateral acceleration should be even lower in the range of 0.01g's. This groundrule establishes a crucial distinction between identifiers designed for adaptive control and those designed for post-flight-data parameter estimation.[14][15] In the latter case, test inputs are deliberately large and often optimized for identification accuracy. For adaptive control, these inputs must be small and, hopefully, not noticeable.

6.2 Candidate Concepts

Three different adaptive designs, each based on a different technique were considered. An over-all functional diagram for all three adaptive concepts is shown in Figure 15. Each concept uses the CCV control laws but adjusts gains with a different adaptive algorithm. The individual adaptive algorithms are discussed in Reference 6.

The preceding designs were compared on the basis of performance, growth potential, and computer requirements. An overview of the comparison is given in Table 6. The MLE design was selected primarily on the basis of growth potential and is discussed in more detail in the next section.

7.0 THE PCMLE ALGORITHM

The PCMLE algorithm is based on standard maximum likelihood estimation theory as applied to longitudinal short-period F-8C dynamics. Instead of using the usual iterative calculations to maximize likelihood functions, however, it uses the parallel channel implementation shown in Figure 16. Several Kalman filter channels operate at fixed locations in parameter space. Likelihood functions are computed for each. Sensitivity equations are then solved only for the maximum likelihood channel and used to interpolate from there to the final parameter estimate with a single Newton/Raphson or a Kalman filter parameter correction.

The Kalman filter correction is based on estimating time-varying parameters to improve parameter tracking.[16] Finally, two level gust estimation is provided by comparing the likelihood functions of two Kalman filters with identical parameters but different gust statistics. The gains of the filter with the best fit of gust statistics are used in the maximum likelihood channel (best parameter fit). Highpass filters on the measurements remove the effects of trim and highpass filters on the likelihood functions deweight old data.

7.1 PCMLE Structure for F-8C Application

Theoretical identifiability results were used to determine the number of parameters that could be identified with small test inputs. This accuracy analysis also provides insight into the number and location of the filter channels. Five parallel channels are used to represent the F-8C aircraft over its entire operational flight envelope. The locations of these channels in $M_{\delta e}$ - M_{α} parameter space are shown in Figure 17. Up to four parameters--surface effectiveness ($M_{\delta e}$), pitching moment due to angle-of-attack (M_{α}), airspeed (V) and normal force due to angle-of-attack ($Z_{\alpha}V$) can be estimated. Estimation accuracy depends strongly on the signal levels in the control loop. For the small test signals producing less than 0.05g rms of normal acceleration, errors of 10 to 20 percent in M_{δ_e} and 20 to 30 percent in M_{α} and V are typical in six-degree-of-freedom simulation runs.

Theoretical accuracy analyses confirm these error levels.

7.2 Parameterization

One of the features that allows the PCMLE algorithm to work well while estimating a small number of parameters is its method of parameterization. All the pitch axis parameters are fit to a function of M_{δ_e} to reduce initial parameter uncertainty. The model coefficients are computed from one dominant parameter c_5 (or M_{δ_e}) plus other small perturbation parameters (c_1-c_4, c_6) as shown in Table 7. The PCMLE algorithm estimates c_5 and c_2 with an option of estimating c_3 and c_4. The small perturbations on M_q and $Z_{\delta}V$ (c_1 and c_6) are not estimated on-line since they have little effect on the other parameters.

8.0 PCMLE IMPLEMENTATION

The PCMLE adaptive control design will be flight tested using the F-8C DFBW research aircraft with a remote digital augmentation technique originally developed for flight testing remotely piloted research vehicles.[13] The remotely augmented vehicle (RAV) concept, as developed for the F-8C DFBW aircraft, allows complete closed-loop control of the aircraft through a ground-based digital computer and telemetry links.

A modified RAV technique will be utilized in the PCMLE adaptive control flight test experiment, as shown in Figure 18. The measurements required by the PCMLE algorithm--pitch rate, normal acceleration, and horizontal stabilator position--are received by the ground-based digital computer via the telemetry downlink. The PCMLE algorithm estimates the aircraft parameters. A dynamic pressure estimate is derived as a linear function of M_{δ_e}. This dynamic pressure estimate is transmitted to the aircraft through the telemetry uplink and transferred to the triplex on-board digital flight control computers. The pilot may then select either the derived dynamic pressure or the on-board value computed from air data. The selected dynamic pressure is then used for the scheduled gains in the pitch axis control laws.

The PCMLE concept was originally developed for on-board implementation. However, for the adaptive control flight test a ground-based digital computer will be dedicated to the PCMLE software. This software contains a number of options for experimental purposes which would not be included in an on-board system. As such, the time and core required are larger than would be expected for on-board applications. The PCMLE software has a multi-rate structure with the highest rate, 53 Hz, used for the Kalman filter and likelihood computations. Other more time-consuming calculations are done at approximately 8 Hz.

The ground-based digital computer, which contains the FORTRAN-coded PCMLE adaptive control algorithm, is the key element in providing the versatility and flexibility necessary to thoroughly investigate the PCMLE adaptive concept in a flight test experiment. The ground computer is a general purpose minicomputer with a 16 bit word length and floating-point hardware. It has 32K words of memory, a 330 nanosecond memory cycle time and a floating-point add-time of 5.1 microseconds. The ground computer program is controlled by an external interrupt slaved to the telemetry uplink and operates on an 18.75 millisecond frame rate. The computer input (or downlink) operates asynchronously on a 5 millisecond frame rate. Direct digital transfers are used for all input-output activity on the ground. Because the adaptive algorithm is coded in FORTRAN, and physically separated from the flight-critical, on-board digital fly-by-wire control system, program modifications for experimental purposes may be made easily and safely. This yields significant benefits in reduced software verification and qualification effort.

9.0 PCMLE PERFORMANCE

9.1 Simulation Results

The PCMLE algorithm was tested on the F-8C Iron Bird at NASA/DFRC. In addition to convergence tests at fixed flight conditions, an acceleration maneuver was used to evaluate the performance of the various options. Figure 19 shows an example of the time histories for this maneuver using the Newton/Raphson parameter correction.

The maneuver shown is a constant altitude acceleration from Mach = 0.4 to Mach = 1.1 at 6,000 meters. The excitation consisted of the small random test signal (produces 0.03g rms normal acceleration) plus small pilot inputs used to maintain trim. The top four traces in Figure 4 show the response of the aircraft during the maneuver. The lower 5 traces illustrate PCMLE performance. The channels switch from 2 to 3 to 4 to 5 as the aircraft accelerates (M_{δ_e} and M_α become more negative) and goes supersonic. The switch to channel 5 corresponds closely to the transition to supersonic flight. The time histories of $\hat{M}_{\delta_e}$ and $\hat{M}_\alpha$ are shown in addition to M_{δ_e} errors and $\hat{q}$ errors. The $\hat{M}_{\delta_e}$ error was computed by comparing the $\hat{M}_{\delta_e}$ estimate with the slope of the $C_{M_{\delta_e}}$ function in the simulation. The $\hat{M}_{\delta e}$ estimate is within 12 percent of the true value. An estimate of dynamic pressure was computed as a linear function of $\hat{M}_{\delta_e}$. A $\hat{q}$ error was computed by comparing the dynamic pressure estimate with the simulation value. This error trace (shown in the bottom of Figure 19) is similar to the $\hat{M}_{\delta_e}$ error and shows the error is less than 15 percent of the true value.

9.2 Flight Data Processing

The PCMLE algorithm was tested using recorded sensor outputs from F-8C flight tests. The flight test recordings do not contain any test signal. Two different types of maneuvers were used. First, as a measure of identification accuracy, abrupt pilot pulses and doublets were used to provide the best possible conditions for identification. Secondly, an acceleration maneuver was used to illustrate the parameter tracking ability of the algorithm.

9.2.1 Parameter Estimation for Abrupt Maneuvers--As examples, five conditions for which data was processed identified in the figure are discussed in this section. Each test point corresponds to about 10 seconds of data.

9.2.1.1 PCMLE Performance--Table 8 summarizes the performance of the PCMLE algorithm. $\hat{M}_{\delta_e}$ is the surface effectiveness estimate. The $\hat{M}_\alpha$ estimate is computed from the M_{δ_e} and $\hat{c}_2$ estimates via the parameterization of Table 7. The $\hat{V}$ estimate is based only on $\hat{M}_{\delta_e}$ since c_3 was not estimated. The sigmas shown in the table are PCMLE estimates of accuracy and tend to be optimistic due to the effects of neglected parameters. The channel number and gust level indicate which of the parallel channels and which gust level was selected. The SIGSO parameter is a ratio of the actual residual rms level to the design value. Ideally, it should be unity when a good parameter fit is obtained. A sample time history of the aircraft response and PCMLE outputs is shown in Figure 20.

9.2.1.2 Offline MLE Results--Two different algorithms were used to estimate parameters for the same five flight conditions. A general purpose identification program (GPMLE) was used at Honeywell to estimate all the pitch axis parameters shown in Table 7. It highpassed the measurements as PCMLE does and then iteratively updated the parameters in the Kalman filter (similar to a single channel of PCMLE) until the likelihood function converged.

Another independent check was obtained by the NASA Dryden Flight Research Center using a modified maximum likelihood estimation (MMLE) program. The MMLE batch processing program (Reference 14) was used to estimate the aircraft stability and control derivatives from the flight maneuvers. It used a three state model (q, α, V) without highpassing the measurements.

9.2.1.3 Comparison of Parameter Estimates--The MMLE estimates of the aerodynamic coefficients were used for comparisons with the estimates obtained from the PCMLE and GPMLE algorithm. A comparison is given in Table 9 for $\hat{M}_{\delta_e}$ and $\hat{M}_\alpha$. Corresponding wind tunnel values (WT) are also shown. A plot of $\hat{M}_\alpha$ vs. $\hat{M}_{\delta_e}$ is given in Figure 21. The solid line illustrates the parameterization used by PCMLE. The dashed lines indicate the range of c_2 based on linear models derived from the wind tunnel data. The flight data estimates from the three identifiers are shown as discrete points. The plot confirms the consistency of the parameter estimates. The two offline methods compare closely. The PCMLE estimates are within 12-15 percent of the off-line values. This establishes the validity of the parameterization. Similar plots for other aerodynamic derivatives are given in Reference 17.

9.2.2 Maneuvering Flight--The performance of PCMLE during a maneuver is shown in Figure 22. The top five traces show the response of the aircraft. The maneuver, lasting about 135 seconds, is an acceleration from Mach ≈ 0.85 to Mach = 1.15 during which the altitude decreases from 41,000 feet to 23,000 feet. This is immediately followed by a deceleration back to Mach = 0.89. The F-8C is supersonic for about 57 seconds during this maneuver. The next two traces show the $\hat{M}_{\delta_e}$ and $\hat{M}_\alpha$ estimates from PCMLE. Note how $\hat{M}_\alpha$ goes sharply more negative (as it should) as the aircraft goes supersonic. The $\hat{M}_{\delta_e}$ estimate was used to produce an estimated dynamic pressure $\bar{q}$. In the bottom trace of Figure 22, this estimate is compared to a dynamic pressure ($\bar{q}_a$) computed from the measured altitude and mach number. The $\bar{q}$ error is large for about 10 seconds because there is no pilot activity. (This maneuver does not contain any test signal.) During the remainder of the maneuver the rms error is about 20 percent.

10.0 CONCLUSIONS

This paper has described design methodologies, performance predictions, and flight test results for several advanced control laws flown as part of NASA's experimental F-8C DFBW program. The outstanding feature of these advanced control law experiments has been the uniform agreement obtained between test results and design/analysis/simulation predictions. The CCV control modes achieved in flight are the same fighter performance benefits predicted during design. Although modest for aircraft like the F-8C, these benefits are dramatic in more modern aircraft. The F-8C DFBW experience suggests that developments of CCV fighter flight controls can proceed with confidence. It is also important to note that the various advanced control modes were readily implemented as part of an integrated digital flight control software package. The success of this system adds needed credibility to the world's still limited flight experience with complex digital flight control systems.

With respect to design methodologies, the program establishes modern linear-quadratic optimization methods as viable tools for serious control system design. A proviso which needs to be emphasized here, however, is that a reasonable formulation of the optimization problem with due consideration of classical constraints is critical. The development of such a formulation still seems to fall into the realm of "designer's art."

Finally, the results available to date on the adaptive control laws suggest the same close agreement between predicted and actual performance. If this level of agreement holds through closed loop RAV flight tests, the adaptive experiments will signify renewed vitality of adaptive techniques as alternatives in modern digital flight control implementations.

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ACKNOWLEDGEMENTS

The authors wish to gratefully acknowledge the assistance of Mr. Al Winton in preparing the artwork and of Ms. Elin Zehren in typing the manuscript.

TABLE 1. DISCRETE TRANSFORMATION FORMS
T = SAMPLE PERIOD, SEC.

Continuous Form	Discrete Form	K	A	B
$\frac{1}{s/a + 1}$	$\frac{K(1 + Z^{-1})}{1 + AZ^{-1}}$	$\frac{1 - A}{2}$	$-e^{-aT}$	-
$\frac{a^2 + b^2}{(s+a)^2 + b^2}$	$\frac{K(1 + 2Z^{-1} + Z^{-2})}{1 + AZ^{-1} + BZ^{-2}}$	$\frac{1 + A + B}{4}$	$-2e^{-aT}\cos bT$	e^{-2aT}

TABLE 2. QUANTIZATION LEVELS

Stick quantization, cm/bit-	
Pitch	0.0080
Roll	0.0058
Rudder pedals	0.0040
Surface command quantization, deg/bit-	
Horizontal stabilizer	0.0105
Aileron (each)	0.0220
Rudder	0.0103
Flap	0.0220
Effective trim quantization, deg-	
Pitch	0.0250
Roll	0.0200
Yaw	0.0100

TABLE 3. BASIC PITCH CAS HANDLING QUALITIES

Task	Pilot Ratings	
	Pilot A	Pilot B
Instrument maneuvers	1 - 3 4 (one point)	1 - 2
Large speed, altitude, attitude changes	1 - 2	2
Formation	4	-
Mild Aerobatics	2	-

TABLE 4. LATERAL-DIRECTIONAL RESPONSE MEASUREMENTS

Flight Condition	Roll Time Constant		Dutch Roll Damping		$\Delta\beta/K$	
	SAS	Unaugmented	SAS	Unaugmented	SAS	Unaugmented
250 KIAS, 6100 M	0.4	0.5	0.7	0.3	1.6	4.5
350 KIAS, 6100 M	0.3	0.5	0.7	-	0.6	1.4
400 KIAS, 6100 M	0.3	0.35	0.7	0.15	2.5	1.0
500 KIAS, 6100 M	0.4	0.6	0.5	0.25	1.1	1.4
242 KIAS, 12,200 M	0.3	0.6	0.4	0.3	0.6	1.6

TABLE 5. HANDLING QUALITY EVALUATION TASKS

Task	
A	Trimmed straight and level flight followed by small heading changes
B	Altitude changes of ±1000 feet
C	Airspeed changes of ±50 KIAS at constant altitude
D	2g to 2.5g windup turns
E	Climb to 5000 feet at $\theta = 20^{\circ}$. Level quickly, return to trim, increase airspeed 50 KIAS and trim

TABLE 6. OVERALL COMPARISON OF CONCEPTS

Characteristic	Model tracker	High-gain limit cycle	Maximum likelihood estimation
Performance	Acceptable to good	Good	Good to excellent
Growth potential	Low--limited to single variable explicit gain schedule	Low--limited to single variable implicit adaptation	High--multiple parameter gain adjustment possible
Computer requirements*	Minimal	Low	Medium

*These requirements are relative to F-8C computer capacity.

TABLE 7. F-8 PARAMETERIZATION

M_q	=	$-0.23 + (.028 - .017\ C_2)\ C_5 + C_1$
M_{α}	=	$(0.61 + 0.92\ C_2)\ C_5$
V	=	$(200. + C_3)\sqrt{-C_5}$
$Z_{\alpha}V$	=	$(53. + C_4)\ C_5$
M_{δ}	=	C_5
$Z_{\delta}V$	=	$(7.7 + C_6)\ C_5$

TABLE 8. PERFORMANCE OF PCMLE ON FLIGHT DATA

Test Point	Measured			$\hat{M}_{\delta e}$	σ_{m}	$\hat{c}_2$	σ_{c_2}	$\hat{M}_{\alpha}$	$\hat{V}$ (ft/sec)	Channel	Est. Gust Level	SIGSQ
	H (m)	V (m/sec)	Mach									
1	7,400	165	0.44	-5.38	0.31	0.255	0.039	-4.55	464	2	Hi	0.883
2	8,200	420	1.12	-33.1	2.10	0.946	0.097	-49.0	1418	5	Lo	0.706
3	7,340	214	0.566	-8.704	0.638	0.2044	0.0244	-6.946	590	3	Lo	0.706
4	7,240	322	0.85	-28.12	1.787	0.0801	0.0266	-19.22	1132	4	Lo	0.6623
5	14,450	222	0.63	-7.776	0.455	0.0984	0.0537	-5.447	558	2	Hi	1.100

TABLE 9. COMPARISON OF PARAMETER ESTIMATES

Test Point	$\hat{M}_{\delta e}$				$\hat{M}_{\alpha}$			
	PCMLE	GPMLE	MMLE	Wind Tunnel	PCMLE	GPMLE	MMLE	Wind Tunnel
1	-5.38	-5.66	-5.75	-5.77	-4.55	-3.92	-3.86	-3.83
2	-33.1	-32.3	-31.98	-35.95	-49.0	-43.5	-43.76	-59.76
3	-8.7	-10.1	-10.01	-9.86	-6.95	-6.51	-6.47	-6.07
4	-28.1	-26.7	-26.33	-24.62	-19.22	-18.9	-18.64	-13.29
5	-7.78	-5.78	-5.60	-4.46	-5.45	-4.22	-4.16	-2.20

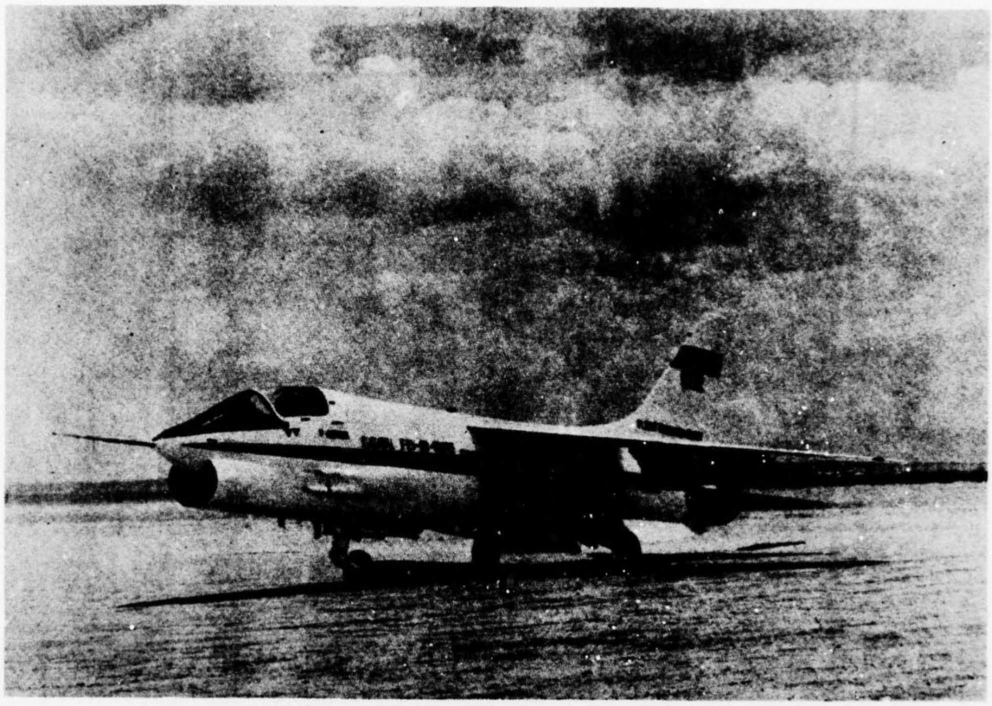

Figure 1. F-8 Ready for Flight

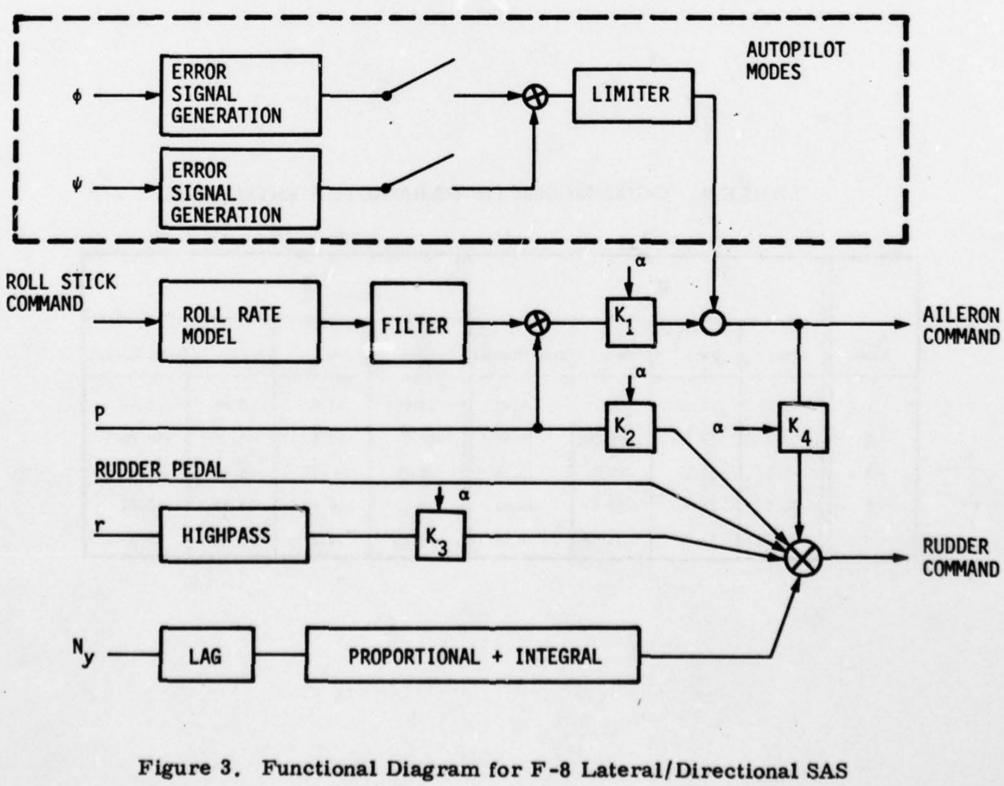

Figure 3. Functional Diagram for F-8 Lateral/Directional SAS

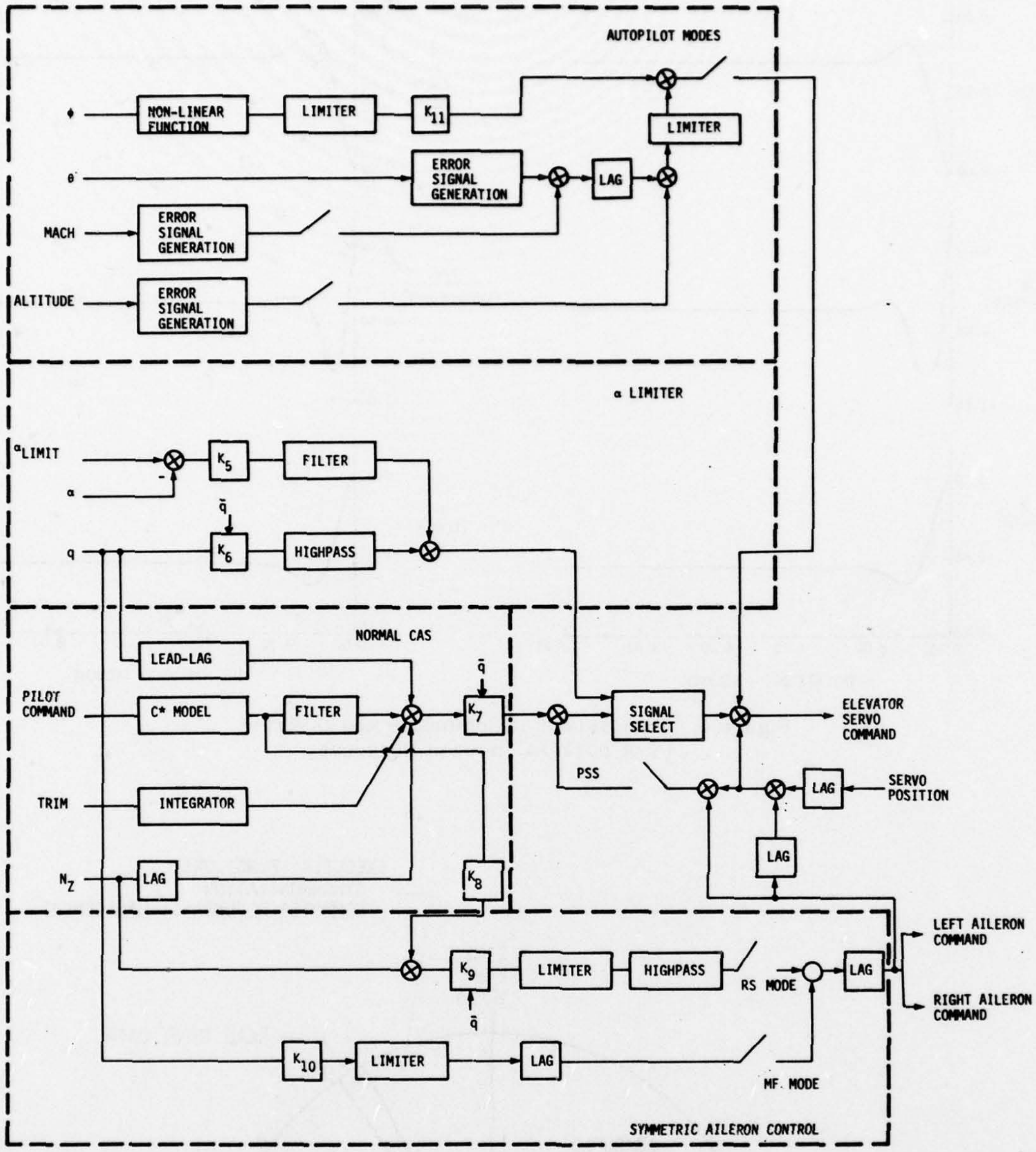

Figure 2. Functional Diagram for F-8 Longitudinal Control

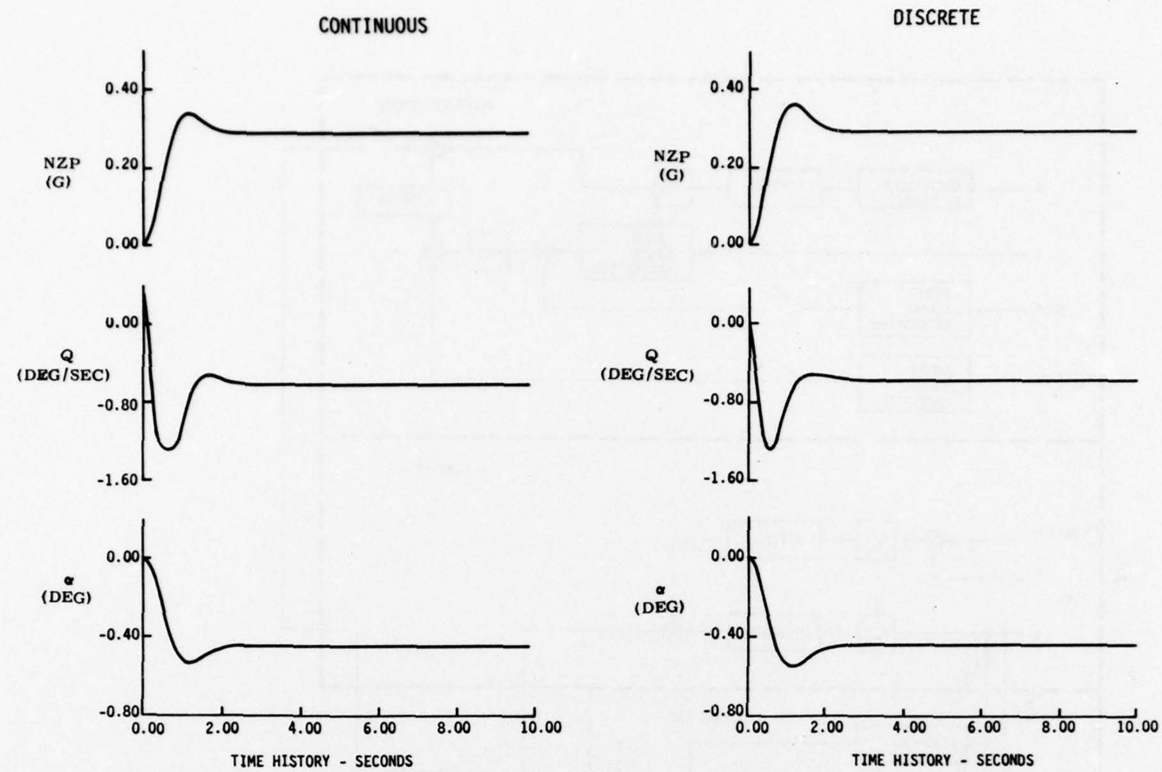

Figure 4. Comparison of Continuous and Discrete (T=0.03125) Command Response

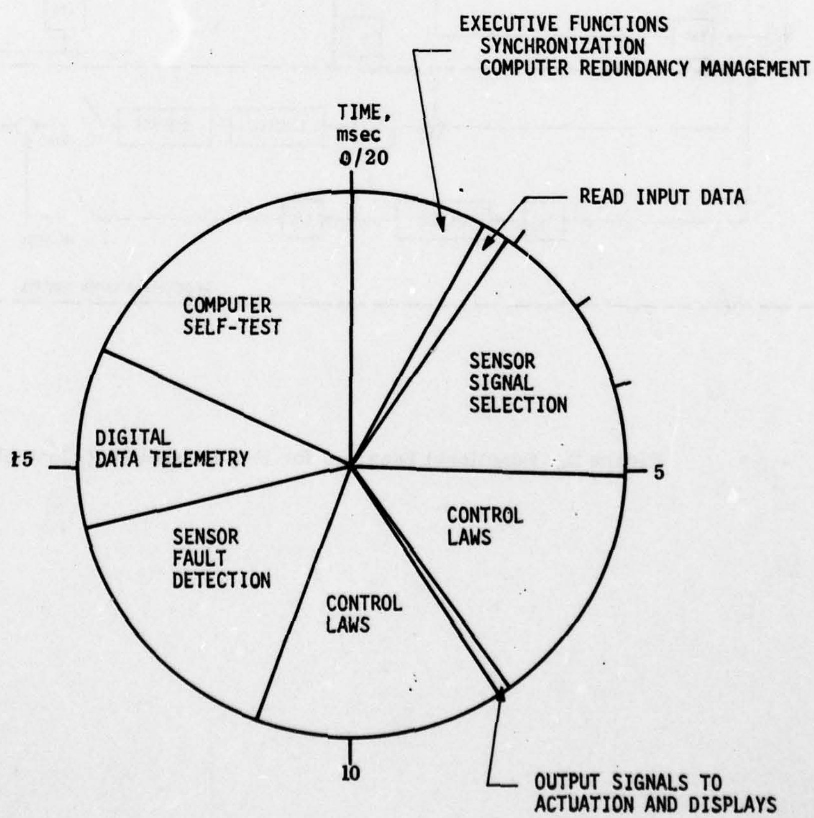

Figure 5. F-8 DFBW Software Timing Chart

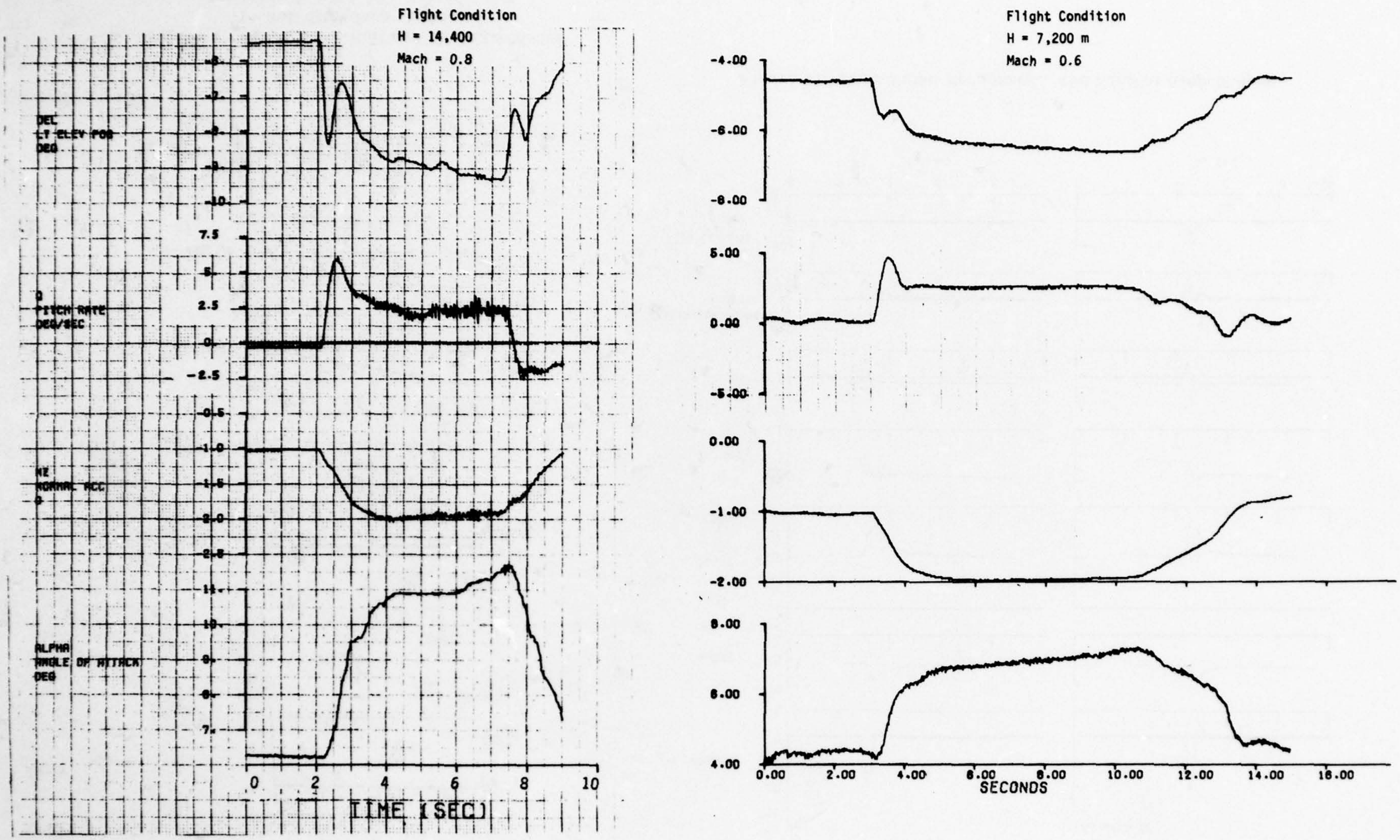

Figure 6. Pitch CAS Response at Two Flight Conditions

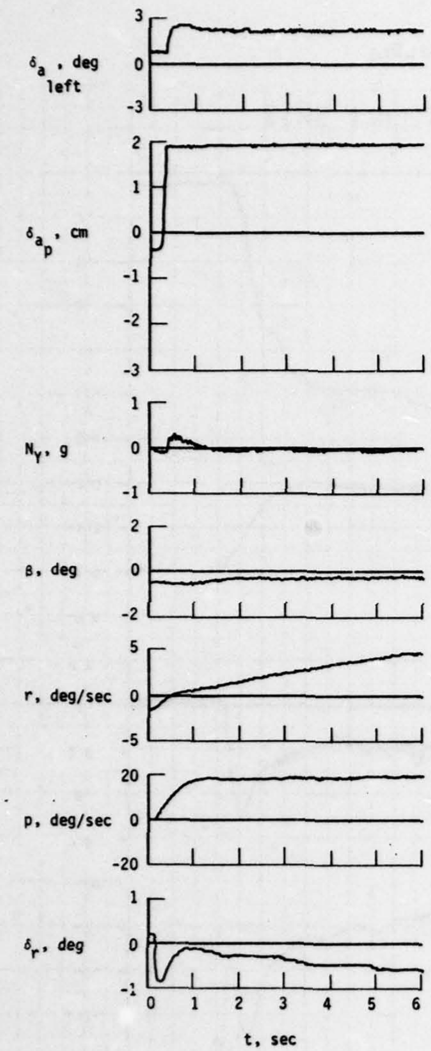

Figure 7. Response in Roll and Yaw SAS Modes to Roll Stick Step Command. Mach 0.8; 12,200 Meters

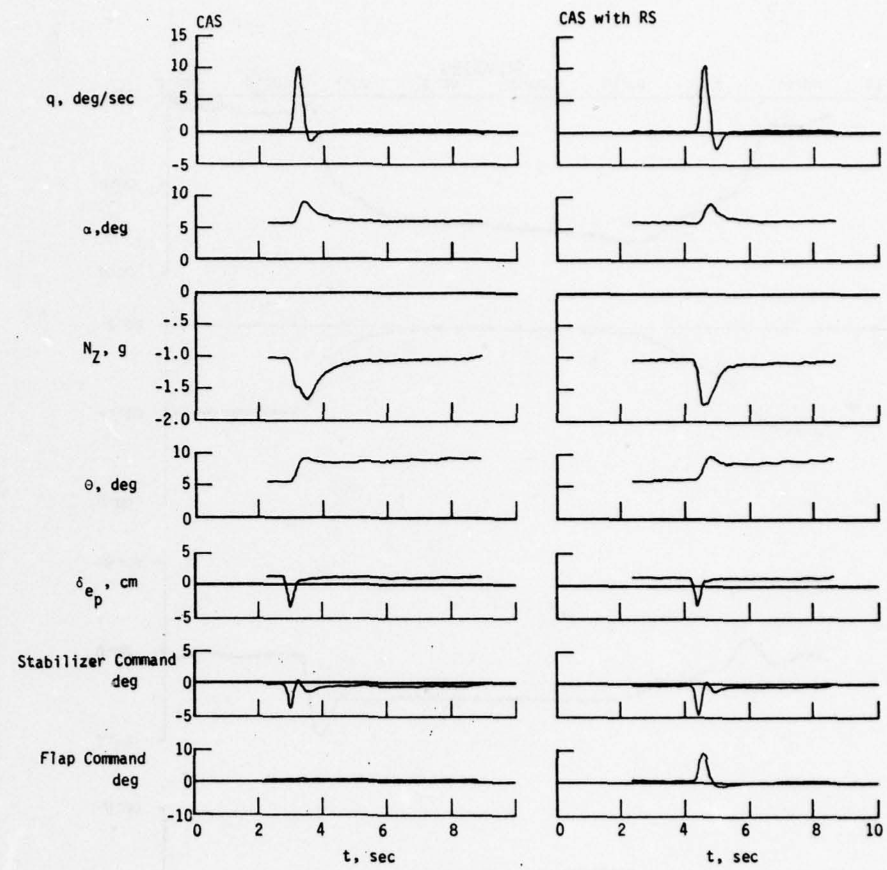

Figure 8. Pitch Pulse Responses. 350 KIAS at 6100 Meters.

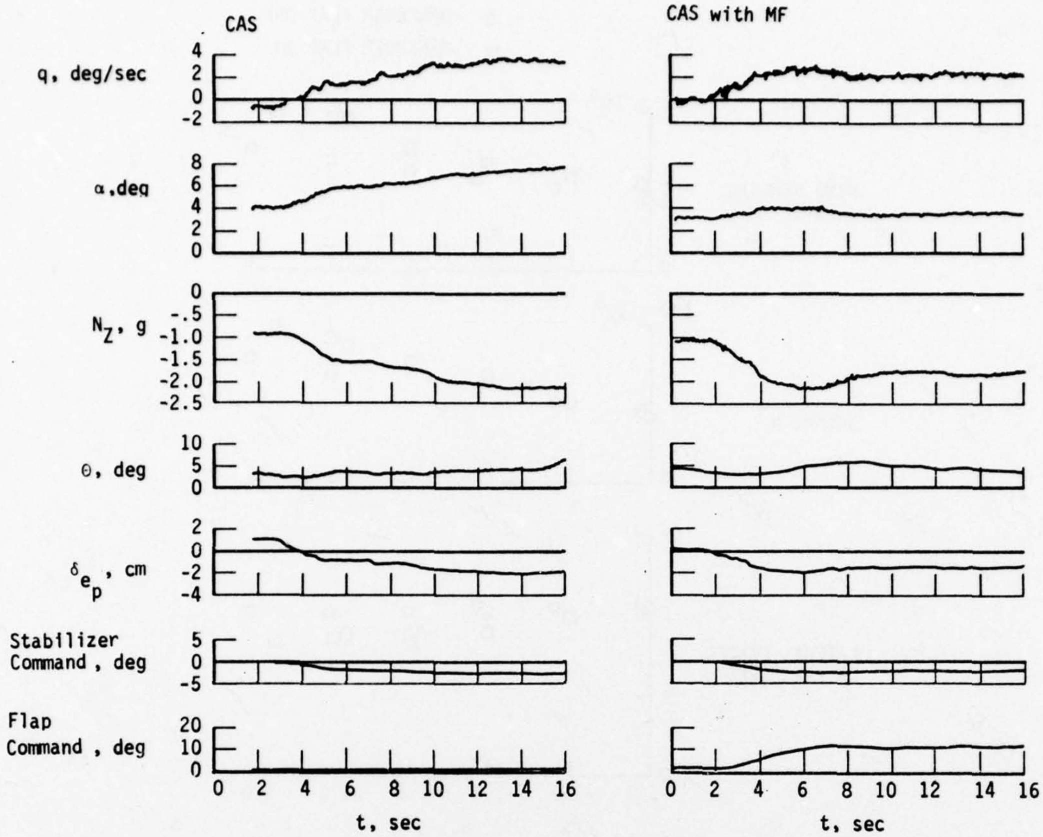

Figure 9. Wind-up Turn 350 KIAS at 6100 Meters

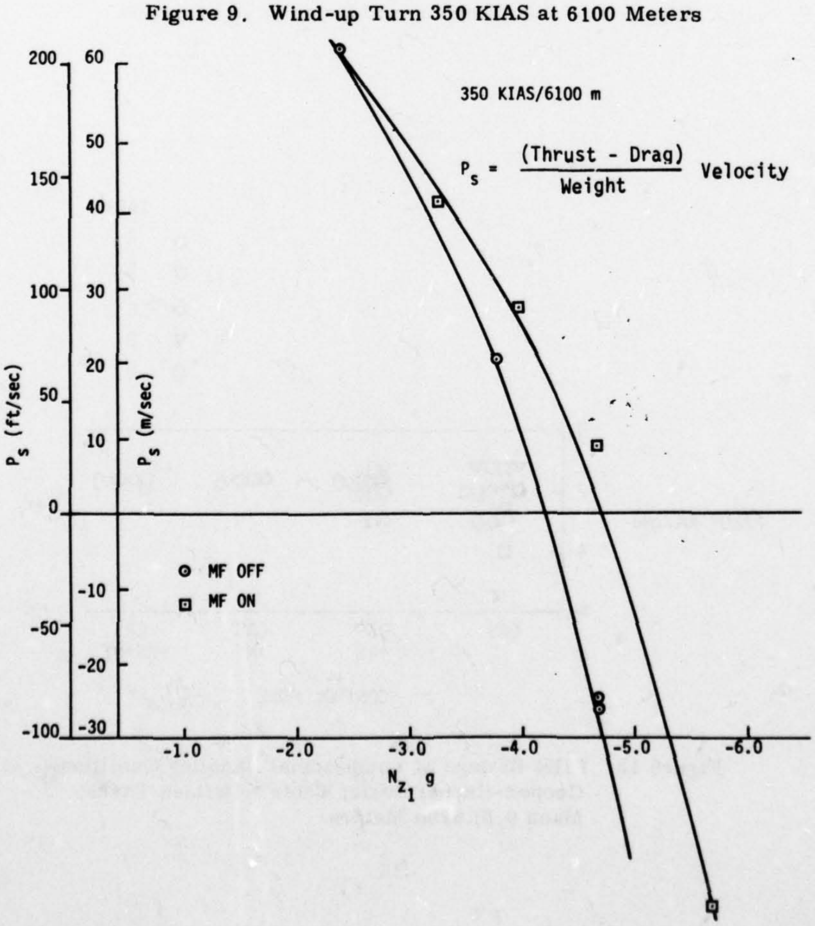

Figure 10. Flight Measurements of P_S

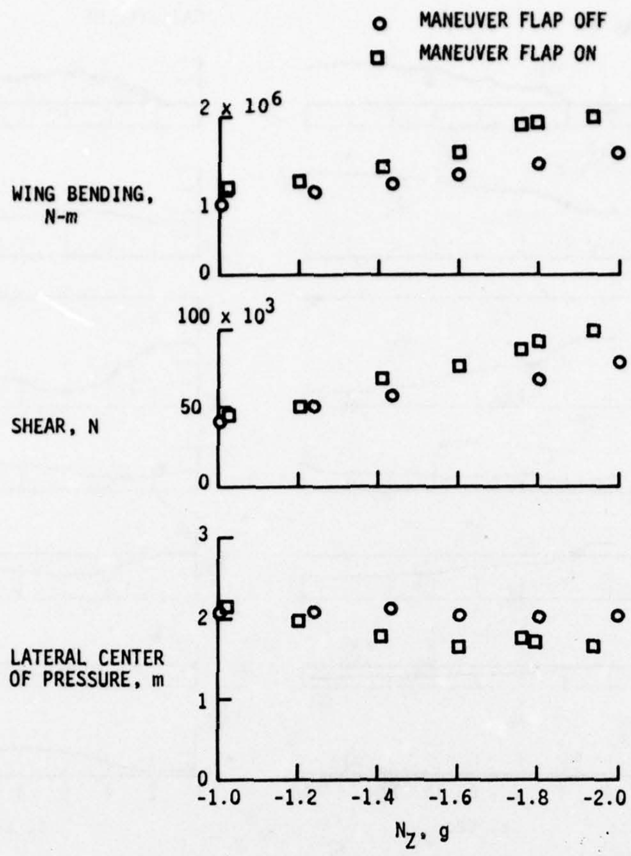

Figure 11. Wing Bending and Shear Load Variation

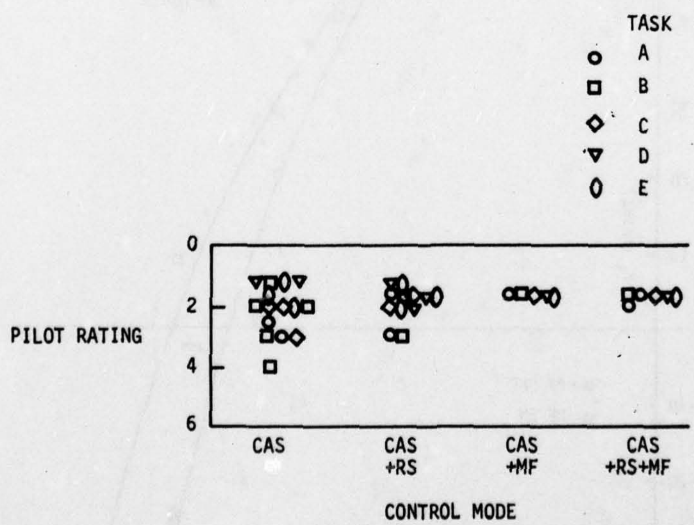

Figure 12. Pilot Ratings of Longitudinal Handling Qualities. Cooper-Harper Scale; Table 5 Defines Tasks; Mach 0.6; 6100 Meters

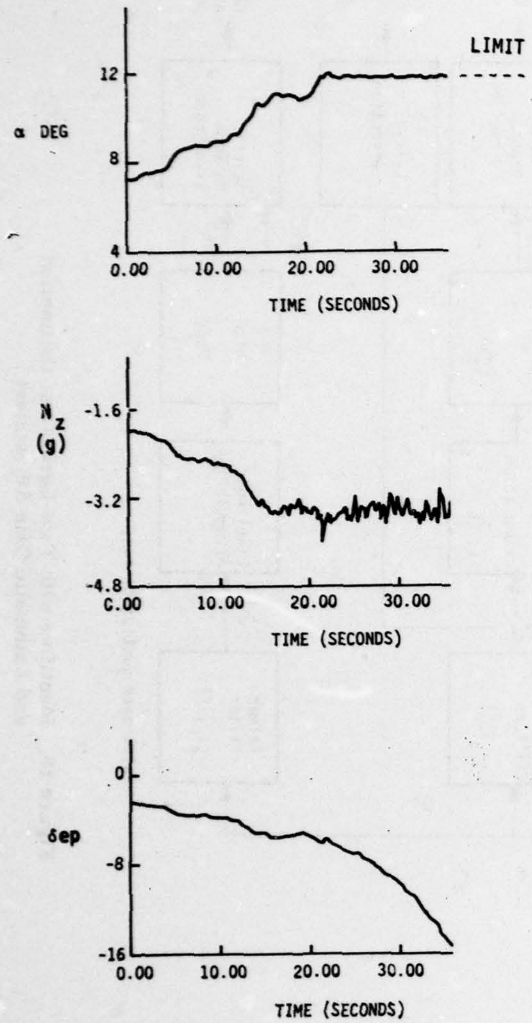

Figure 13. Alpha Limiter Test,
Slow Approach to Limit

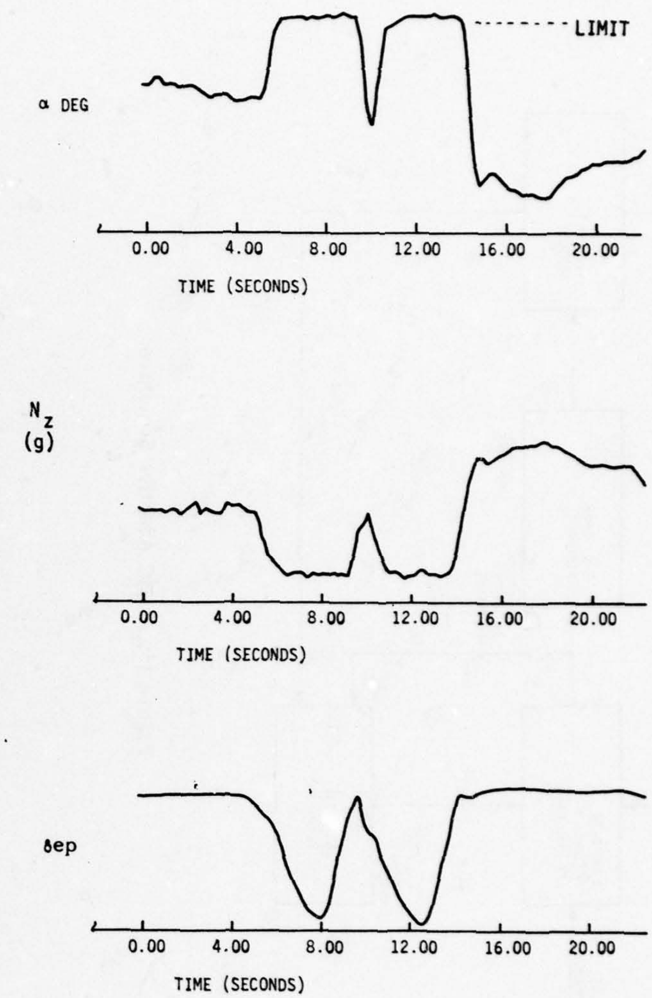

Figure 14. Alpha Limiter Test, Fast Approach to Limit

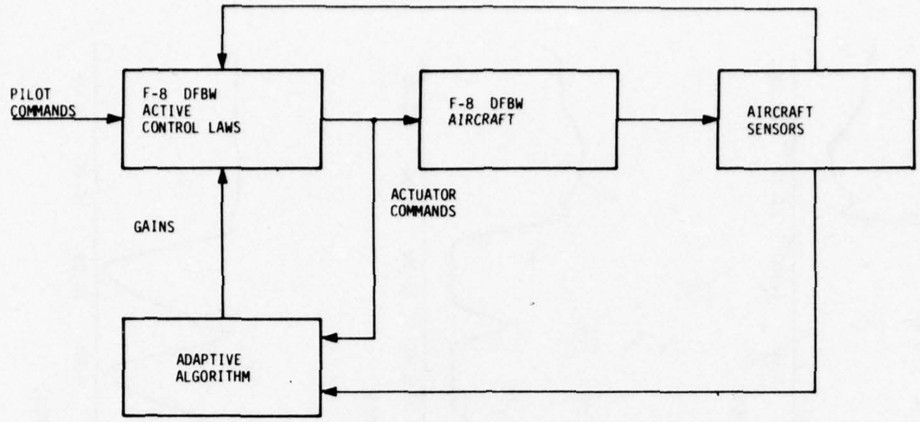

Figure 15. F-8C Adaptive Structure

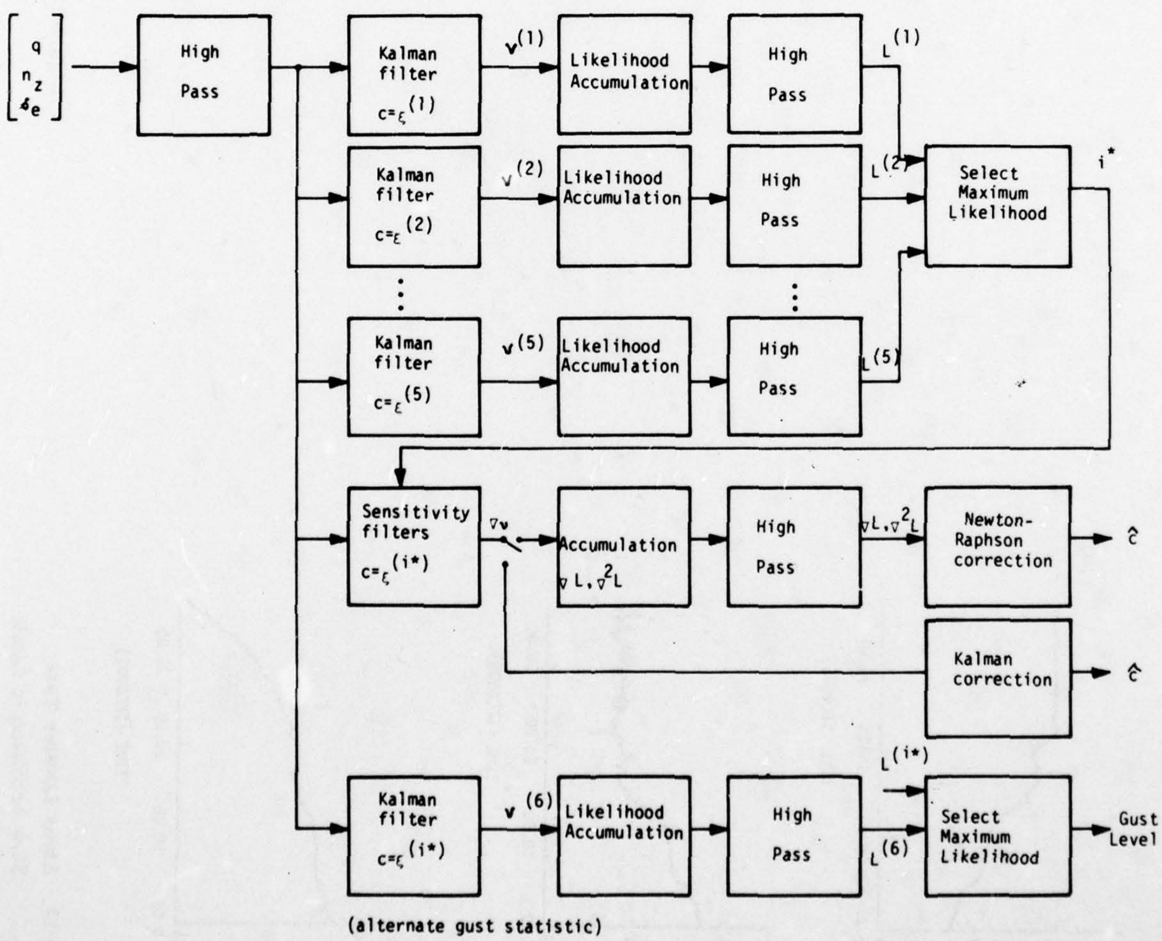

Figure 16. Identifier with Two-Level Gust Estimation and Automatic Gain Adjustment

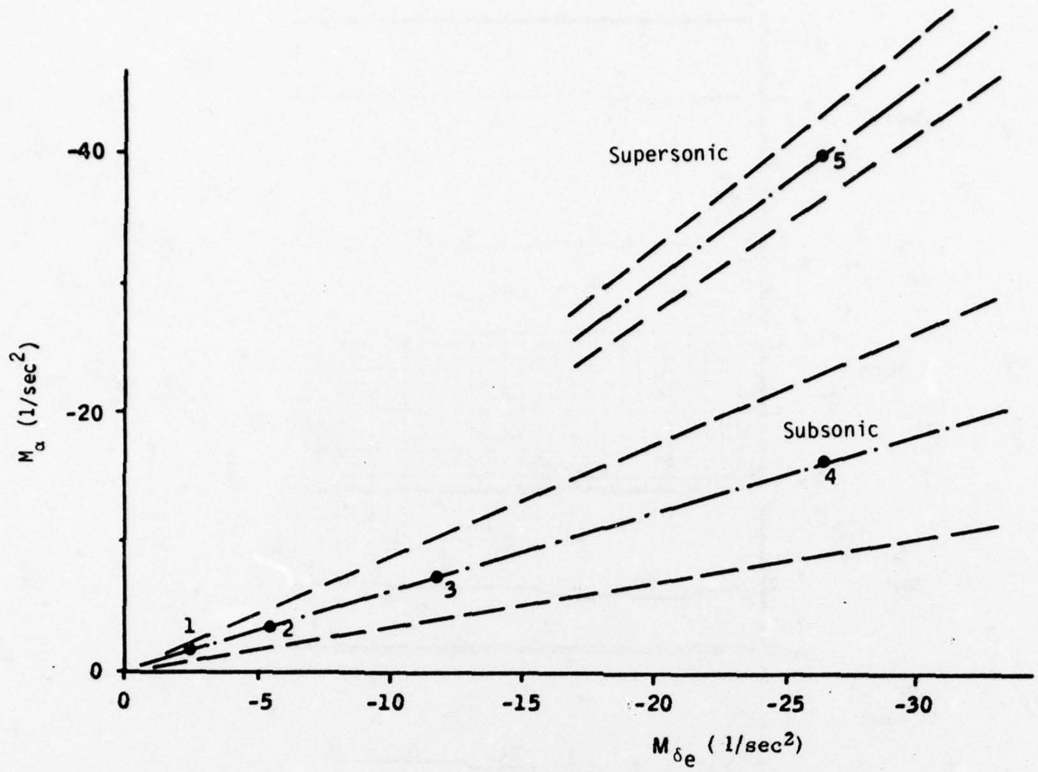

Figure 17. F-8C Identifier Channel Locations

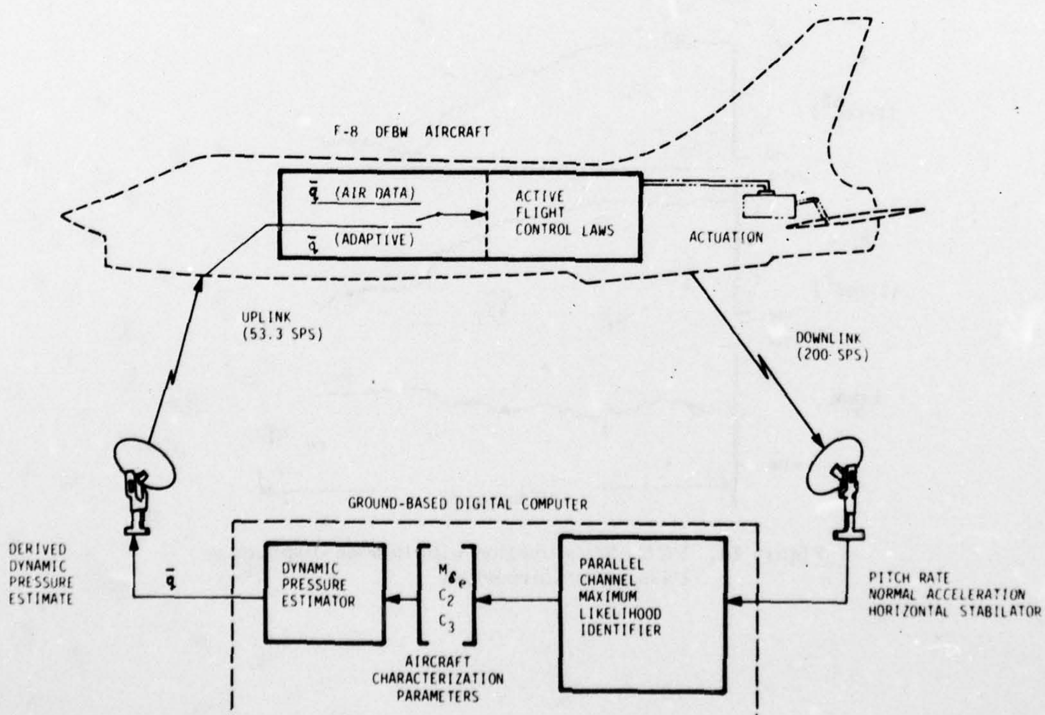

Figure 18. Adaptive Control Flight Test Approach

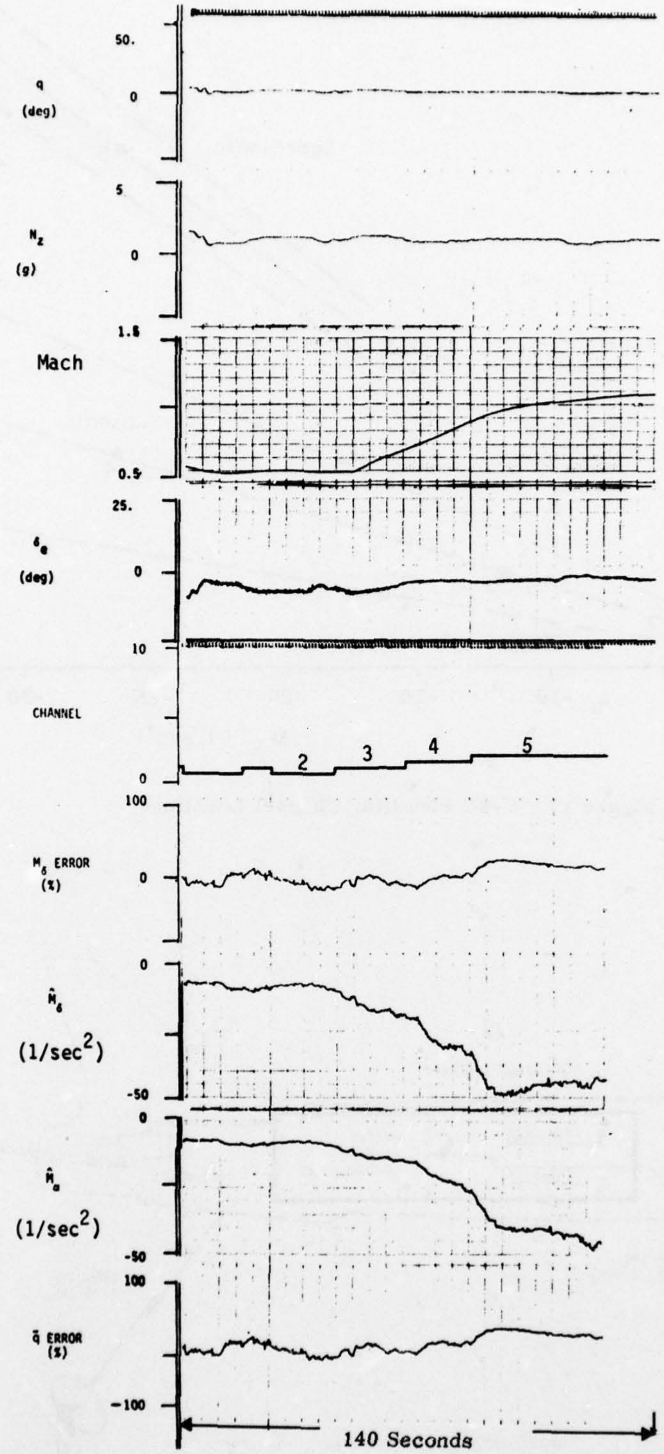

Figure 19. PCMLE Estimation with Newton-Raphson Parameter Correction

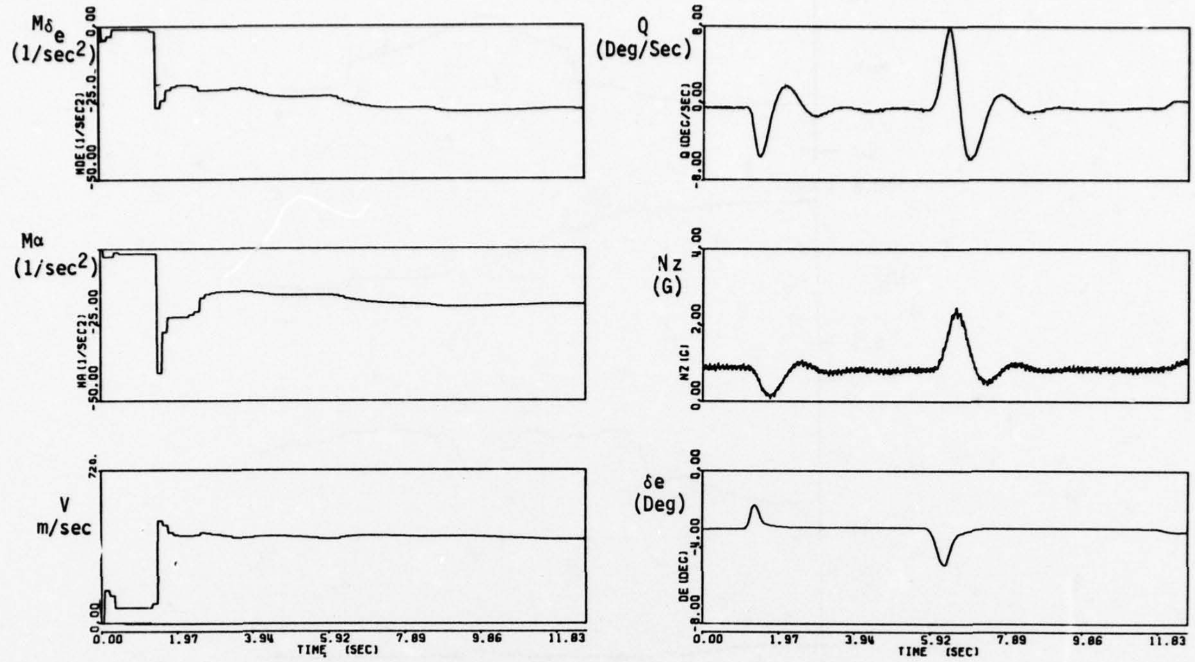

Figure 20. PCMLE Performance at Test Point 4

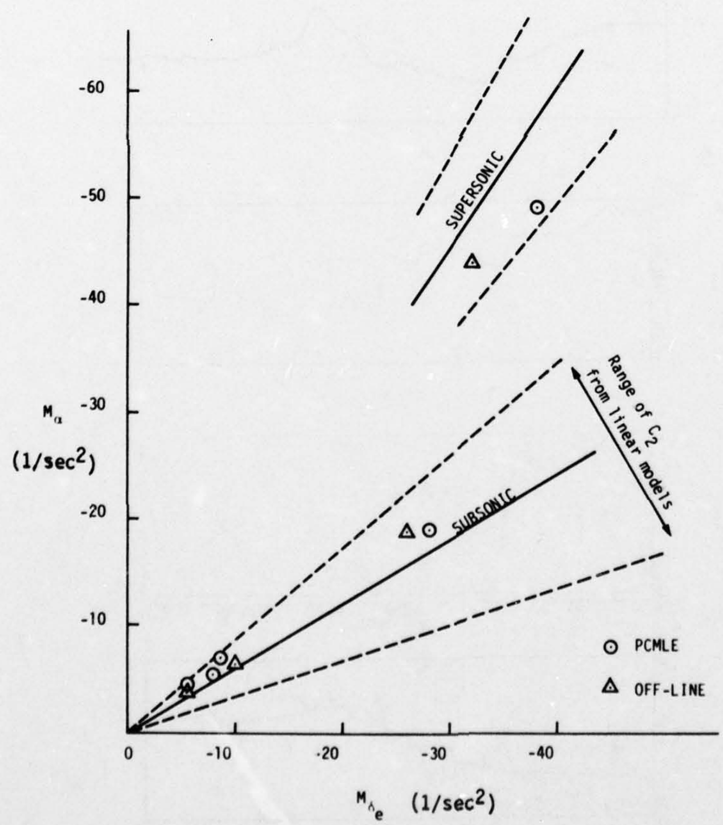

Figure 21. M_α versus M_{δ_e}

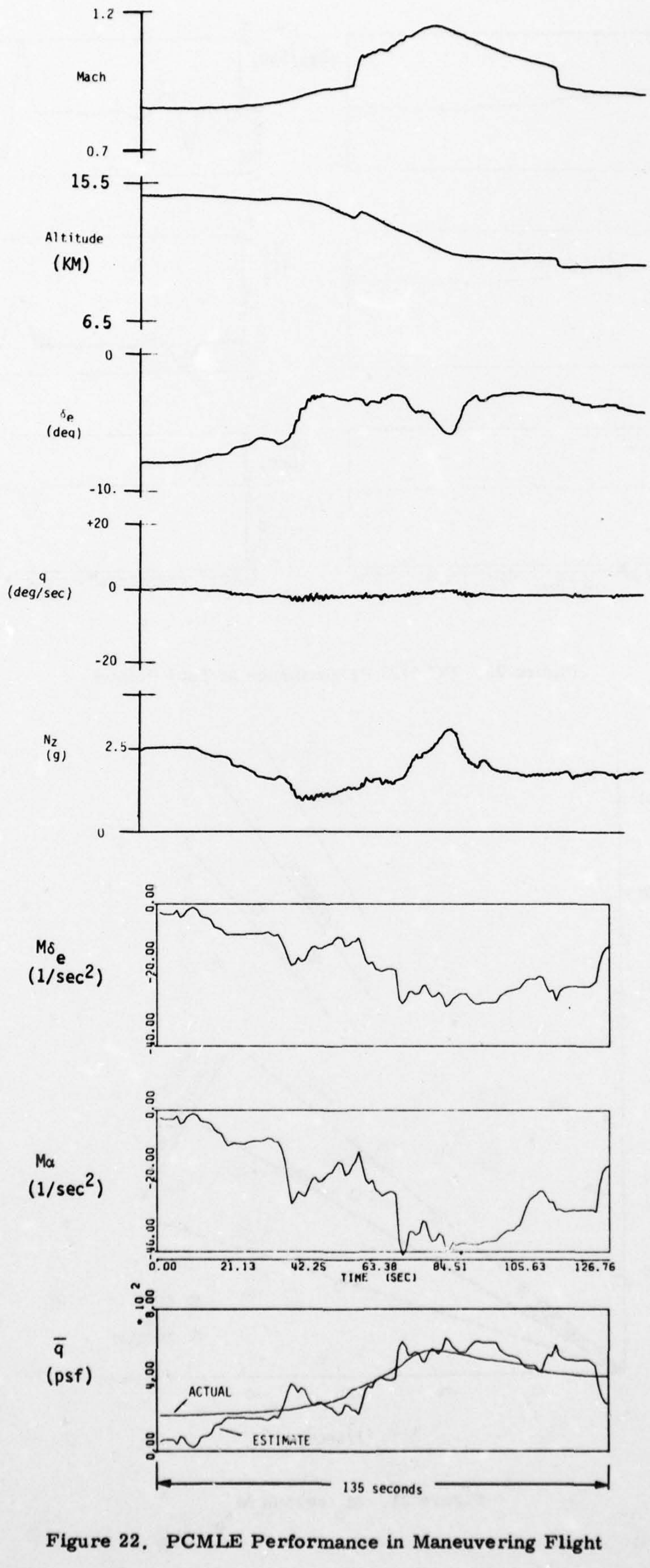

Figure 22. PCMLE Performance in Maneuvering Flight

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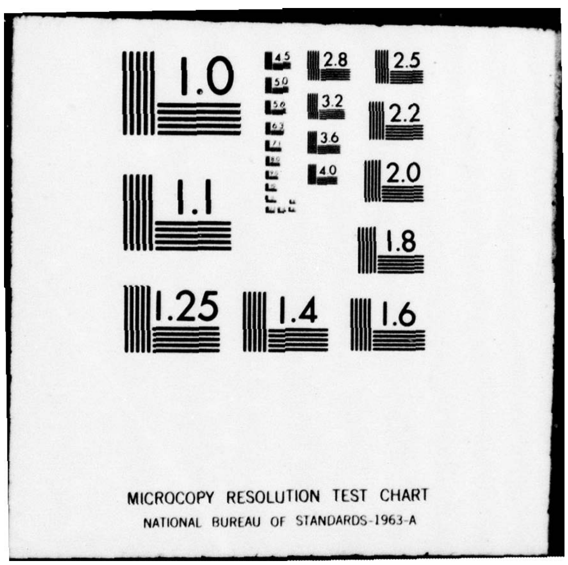
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MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

HIGHLY MANEUVERABLE AIRCRAFT TECHNOLOGY

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SUMMARY

A remotely piloted research vehicle (RPRV) with active controls has been designed to develop highly maneuverable aircraft technologies (HiMAT). The HiMAT RPRV is the central element in a new method to bring advanced aircraft technologies to a state of readiness. The RPRV is well into the construction phase, with flight test evaluations planned.

The closely coupled canard-wing vehicle includes relaxed static stability, direct force control, and a digital active control system. Nonlinearities in the aerodynamics led to unusual demands on the active control systems. For example, the longitudinal static margin is 10-percent negative at low angles of attack, but increases to 30-percent negative at high angles of attack and low Mach numbers.

This paper discusses the design procedure followed and experiences encountered as they relate to the active control features. Emphasis is placed on the aspects most likely to be encountered in the design of a full-scale operational vehicle. In addition, a brief overview of the flight control system features unique to the RPRV operation is presented.

NOMENCLATURE

C_L	lift coefficient
C_m	pitching moment coefficient
$C_{m_{C_L}}$	$\frac{\partial C_m}{\partial C_L}$, pitching moment with respect to lift coefficient
C_n	yawing moment coefficient
C_{n_β}	$\frac{\partial C_n}{\partial \beta}$, yawing moment with respect to angle of sideslip, rad^{-1}
$C_{n_{\delta_r}}$	$\frac{\partial C_n}{\partial \delta_r}$, yawing moment with respect to rudder deflection, rad^{-1}
cg	center of gravity
$\bar{c}_w$	mean geometric and aerodynamic chord of the wing, m
HiMAT	highly maneuverable aircraft technology
h	altitude, m
K	gain
$K(\cdot)$	gain programed as a function of $(\cdot)$
K_{n_Y}	lateral acceleration feedback gain to the rudder, deg/g
K_p	roll rate feedback to the antisymmetric elevons, sec
K_r	yaw rate feedback to the rudders, sec
LVDT	linear voltage differential transformer
M	Mach number
n_Y	lateral acceleration, g
n_Z	normal acceleration, g
PROM	programable read-only memory
p	angular velocity about the roll axis, deg/sec
q	angular velocity about the pitch axis, deg/sec
$\bar{q}$	dynamic pressure, N/m^2
RPRV	remotely piloted research vehicle
RSS	relaxed static stability
r	angular velocity about the yaw axis, deg/sec
s	Laplace variable
α	angle of attack, deg
α_ℓ	limit angle of attack, deg
β	angle of sideslip, deg
δ_a	aileron deflection, $\delta_{a_L} - \delta_{a_R}$, deg
δ_{a_p}	pilot's roll stick command, cm
δ_c	canard flap deflection, $\frac{\delta_{c_L} + \delta_{c_R}}{2}$, deg

$\delta_{c_t}(\cdot)$ canard flap trim input programed as a function of $(\cdot)$, deg

δ_e elevator deflection, $\frac{\delta_{e_L} + \delta_{e_R}}{2}$, deg

δ_{e_p} pilot's pitch stick command, cm

δ_f elevon surface deflection, deg

δ_{f_a} antisymmetrical elevon deflection, $\delta_{f_L} - \delta_{f_R}$, deg

δ_{f_s} symmetrical elevon deflection, $\frac{\delta_{f_L} + \delta_{f_R}}{2}$, deg

δ_r rudder deflection, deg

δ_{r_p} pilot's rudder pedal command, cm

Subscripts:

com command

L left

R right

INTRODUCTION

A subscale remotely piloted research vehicle (RPRV) has been designed as part of the highly maneuverable aircraft technology (HiMAT) program. The HiMAT RPRV design is the central element in a new method for bringing advanced aircraft technologies to a state of readiness. The method begins with a paper design of an aircraft incorporating the new technologies of interest. A subscale RPRV is then designed and built to demonstrate the advanced technologies in a flight environment. The first application of the method is in the high maneuverability areas incorporated in the HiMAT vehicle. An overview of the HiMAT RPRV in terms of the various new technology areas to be demonstrated is given in reference 1.

One of the new technology areas included in the HiMAT program is active controls. Although several functions are included under the active controls banner, relaxed static stability (RSS) and direct force control are the only functions included in the HiMAT design, because they offer the greatest potential for improved performance. RSS is applied to both the longitudinal and directional axes, although the longitudinal axis is substantially more dependent on active controls. Several studies, such as those described in references 2 and 3, have concluded that RSS benefits are highly dependent on configuration. The HiMAT vehicle, with a closely coupled canard-wing planform, represents a fighter configuration significantly different from the conventional wing-tail fighter configuration. In addition, the HiMAT vehicle has aeroelastic tailoring which, when combined with a construction technique that yields a more flexible structure, results in highly nonlinear aerodynamics. Thus, the application of RSS to this unconventional configuration offers the potential for an important advancement in the active controls technology base for fighter aircraft.

The NASA-sponsored HiMAT program has received the guidance and assistance of the United States Air Force, both in Washington, D.C., and at Wright-Patterson Air Force Base. Rockwell International is under contract to the NASA Dryden Flight Research Center (DFRC) to design and manufacture two vehicles. The HiMAT RPRV is currently in the construction phase at the Rockwell International, Los Angeles Division, facility; delivery of the first vehicle is planned for March 1978. Flight test evaluations are to be performed at the NASA DFRC following ground checkout.

This paper discusses the design procedure followed and experiences encountered as they relate to the active control features. Emphasis is placed on the aspects most likely to be encountered in the design of a full-scale aircraft of similar planform. Details of the resulting primary control laws are presented. An overview of the backup control laws and the implementation of the primary and backup systems is given, since they are unique to the RPRV operation.

HiMAT RPRV CONCEPT

The HiMAT RPRV concept is to use RPRV's to speed the technology transition from wind tunnel to flight and to reduce the cost of aeronautical experiments exercising new technology. The concept involves two distinct steps: the first is a design study of a full-scale airplane; the second is the design, manufacture, and flight test of a subscale RPRV.

Full-Scale Fighter Design

Three contractors performed conceptual design studies of a full-scale fighter aircraft employing synergistic combinations of new technologies. The maneuverability goal for the full-scale fighter aircraft was the ability to sustain an 8g turn at Mach 0.9 at an altitude of 9140 meters. The studies also included an assessment of the problems associated with demonstrating the technologies on a subscale RPRV.

The Rockwell design stresses the use of aerodynamic and structural technologies to obtain the high maneuverability. The design closely couples canard and wing, bringing the two lifting surfaces close together to develop a favorable interaction in their flow fields by way of a tailored total airplane span load distribution. This requires low drag at lift coefficients greater than 1.0.

Subscale RPRV

The subscale HiMAT RPRV to be used to demonstrate the new technologies in flight is a 0.44-scale version of the full-scale fighter aircraft. The maneuverability goal for the RPRV was the ability to sustain an 8g turn at Mach 0.9. An altitude of 7620 meters was selected to effectively match the wing loading of the full-scale aircraft. Figure 1 is a three-view drawing of the HiMAT RPRV with control surfaces indicated. The canard flaps are used either symmetrically, for longitudinal control, or antisymmetrically, for direct side force control. The ailerons and elevators are used for roll and pitch control, respectively. The elevons may be commanded antisymmetrically

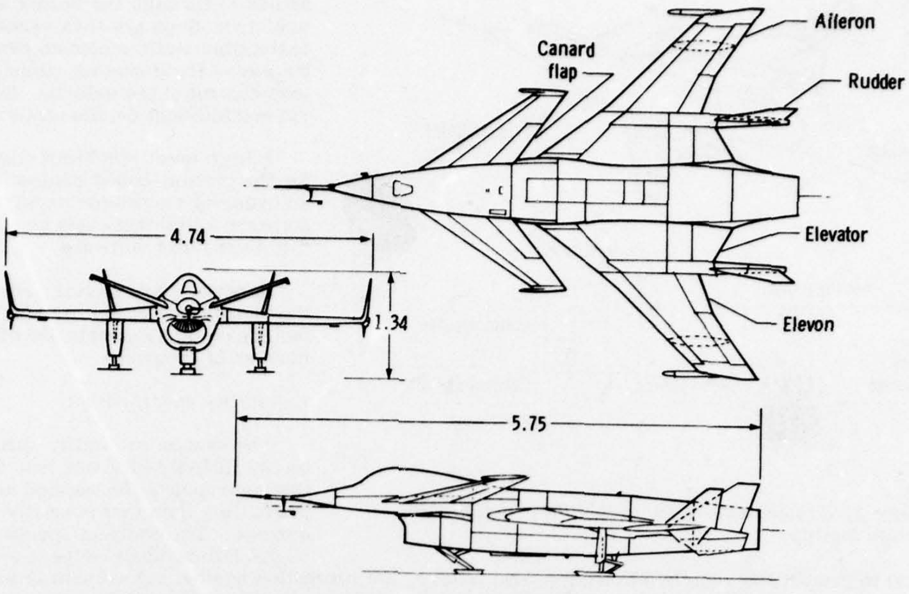


Figure 1. Three-view drawing of HiMAT RPRV. Dimensions are in meters.

for roll control or symmetrically for pitching moment control. The rudders may be commanded collectively for yawing moment control or differentially as a speed brake. The 1500-kilogram vehicle is to be air-launched from a B-52 airplane and will carry 270 kilograms of fuel for the J85-21 engine. The vehicle will be landed horizontally on a dry lakebed under primary control of a ground-based pilot using controls and instrument displays typical of those used in conventional fighter aircraft, as well as a television display generated from an onboard, forward-looking television camera.

Several features of the full-scale fighter design were not included in the RPRV. For example, a two-dimensional nozzle was not included because of cost constraints. Blended wing-body and canard strakes were not included because the two-dimensional nozzle would have been necessary to trim out the resulting high angle-of-attack pitching moment characteristics. A low cost approach was used whenever possible. In some cases, low cost could be achieved by using methods unique to an RPRV operation and without compromises in performance. In other cases, some compromises in performance were necessary. Some of the low cost methods and the effects on performance are detailed below.

Limited wind tunnel data base

Although computerized aerodynamic methods were preferred as a configuration development design tool, none of the methods completely accounted for the entire configuration (body, canard, wing, and winglet). Wind tunnel tests were consequently a necessary ingredient in the aerodynamic configuration development. However, in keeping with the low cost approach, the amount of testing was considerably less than that usually expended for a refined manned fighter aircraft. Only 800 hours of wind tunnel tests were run before initiating fabrication of the HiMAT vehicle. In comparison, approximately 2000 hours are required for a typical prototype manned airplane (for example, 1940 hours for the YF-16 airplane) and approximately 10,000 hours for a fully refined airplane (12,000 hours for the F-14 airplane).

Low cost elements

An important aspect in achieving low cost in the subscale RPRV was the modular design of the control surface actuators (canards, ailerons, elevons, elevators, and rudders). The combination of common components led to a variety of servoactuator implementations, namely dual tandem, single, and single tandem. This was cost effective in design and fabrication, minimized spare part requirements, and will facilitate future configuration changes.

While all performance requirements were met with the modular actuators, several compromises were made in the system's functional capability to accrue additional cost savings. Although an all-movable canard would have been necessary to obtain the incremental 1g design goal for direct lift, a less effective and less costly canard flap was used. Despite falling 50-percent short of the design goal, the direct lift capabilities of the closely coupled canard configuration can still be demonstrated.

In some instances, manual configuration changes must be made on the ground between flights of the RPRV in order to demonstrate features of the full-scale fighter aircraft. The wing and canard leading edges, rather than being continuously variable as in the full-scale fighter aircraft, must be manually changed to one of two distinct settings. One setting is denoted "maneuver wing"; the other is denoted "cruise wing." The direct force canard controls, rather than providing an arbitrary mixture of direct lift and direct side force, must be selected for one or the other function by the ground-based pilot.

Ground facility

Another feature of the RPRV operation that reduces program costs is the ground facility, which contains a cockpit with controls and displays typical of conventional fighter aircraft. A general purpose minicomputer is also available for mechanizing a control system on the ground. Figure 2 illustrates the arrangement between the vehicle and the ground facility. The pilot and the ground-based minicomputer send signals to the vehicle control surface

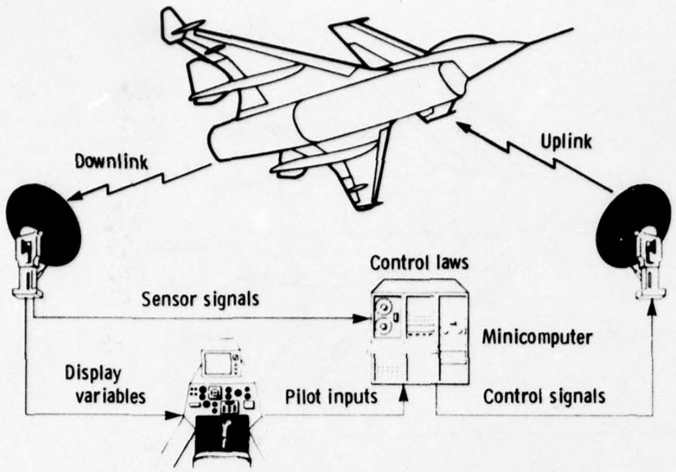


Figure 2. Conceptual layout of RPRV operation with ground facility.

actuators through the uplink system. Resultant vehicle motions are then sensed and sent back to the pilot and the control system computer by way of the downlink, thus providing closed-loop control of the vehicle. Reference 4 provides additional details on this operation.

A high level FORTRAN compiler is available for the ground-based minicomputer, providing an order-of-magnitude savings in coding and software validation costs as compared with flight computer software.

Hardware cost savings are possible because the facility is already in existence and the facility equipment costs are divided among a number of programs.

Reliability specification

The system reliability specification imposed on the HiMAT RPRV was less demanding than that appropriate for manned aircraft but more demanding than that normally associated with a drone. The contract specification that "no single failure shall cause loss of the vehicle" was imposed to protect the vehicle following a first failure, but protection against subsequent failures was not required. Consequently, the resulting system configuration is considerably different from what one would expect for an active control system in a manned aircraft.

The primary system, which uses the ground-based minicomputer, is functionally similar to that of a full-scale fighter aircraft with respect to the control laws. It is a simplex system with in-line monitoring. The backup system is in no way similar to that appropriate for a manned vehicle. It is a semiautomatic system that must provide safe return and landing capability, independent of the ground facility. An alternate command station in a chase airplane is used to make autopilot-type commands typical of drone operation. An onboard, microprocessor-based control system provides the necessary autopilot functions for stabilization, orbit, cruise at wings level, and approach and flare.

INTEGRATED ACTIVE CONTROLS DESIGN

Design Procedure

As in conventional airplane design procedures, the active controls design involved an iterative process. Figure 3 illustrates one cycle of the process. A set of requirements and a startup configuration were used to

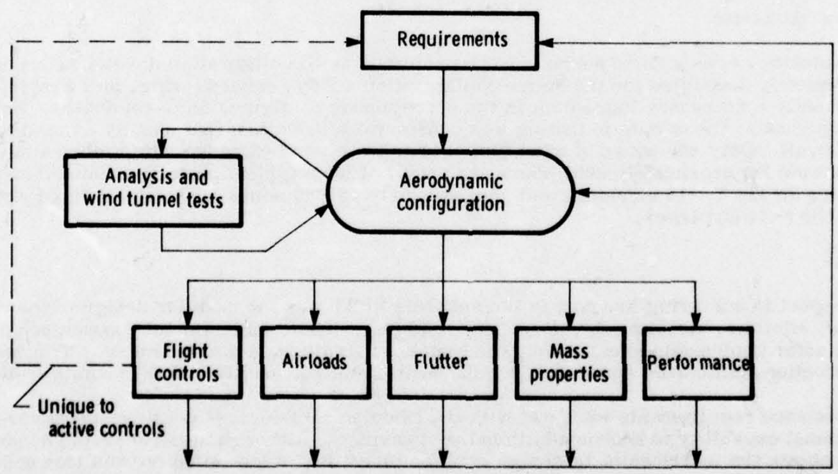


Figure 3. One cycle of iterative integrated design process.

define wind tunnel models and analytical aerodynamic models. Test results from these models formed the basis of a revised aerodynamic configuration and provided the necessary data to update or begin the various support activities, including flight control design, airload and flutter analysis, mass properties definition, and performance analysis. Results of the various analyses resulted in either confirmation of an acceptable configuration or an adjustment to the requirements that governed the subsequent iteration. The dashed line in the figure represents the need for early definition of control requirements prior to the iteration cycle. It corresponds to the aerodynamic configuration developer's asking the controls engineer, "How much control power is needed?" The controls engineer responds with a question: "What do I have to stabilize?"

It is interesting to note that the function of assuring the final level of adequate stability and control was performed by the flight controls engineers, rather than by the aerodynamics engineers as is the usual case with conventional airplane designs. In addition, the flight controls group was involved in the configuration development from the start rather than being consulted later in the design process.

With the exception of the need to establish control power sizing requirements before sufficient information was available to state such requirements, and the transfer of stability and control responsibility, the design process was rather standard. Although the flight controls engineers received the aerodynamic characteristics corresponding to the most recent configuration data set, the flexibility corrections were not available until the new iteration on airloads was completed. Because the flutter analysis lagged behind the other analyses, its iteration cycle was longer.

Design Experiences

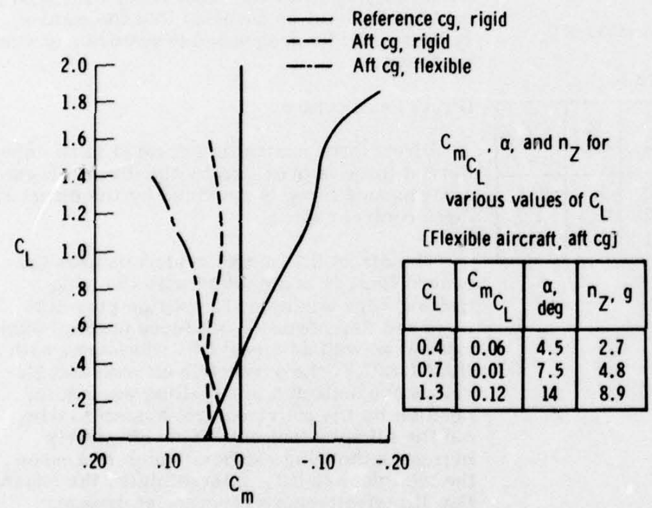
Three aspects of the configuration development stand out as unique to the active controls philosophy in the design. Two of these involved relaxed static stability; the third, direct force control.

Relaxed longitudinal static stability

In the early configuration development, the major problem related to relaxed static stability was the determination of control power requirements. Initially, based on advanced manned strategic aircraft design experience (ref. 5), 10-percent negative longitudinal static margin was selected as a limit for the rigid airplane. This limit was later increased to 15-percent negative based on the following rationale.

The landing condition in a wind gust was found to be critical. A ground rule was established that 20-percent control surface travel should be available for stability augmentation, after accounting for trimming to the landing condition and encountering a wind shear of reasonable intensity. The wing trailing edge pitch control surfaces (elevator and symmetric elevons) have a 30° trailing edge down limit; therefore, a deflection of no more than 25° trailing edge down could be allotted for trim plus wind shear. For a landing speed of 80 meters per second and a 15-percent negative static margin, a wind shear equivalent to a $\Delta\alpha$ of 5.7° or a Δn_Z of 0.4g could be tolerated under the above restrictions. This was considered a reasonable combination of landing speed and wind shear.

At higher angles of attack, a sharp nonlinearity in $C_{m_{C_L}}$ actually increased the static margin to more than 30-percent negative at some low subsonic flight conditions. To illustrate the degree of instability, including the effects of cg changes and flexibility corrections, figure 4 shows C_L as a function of C_m for the cruise wing at Mach numbers of 0.9, 0.7, and 0.2.



C_L	$C_{m_{C_L}}$	α, deg	n_Z, g
0.4	0.06	4.5	2.7
0.7	0.01	7.5	4.8
1.3	0.12	14	8.9

(a) Mach 0.9, h = 9140 meters.

Figure 4. Longitudinal stability of cruise wing HiMAT vehicle.

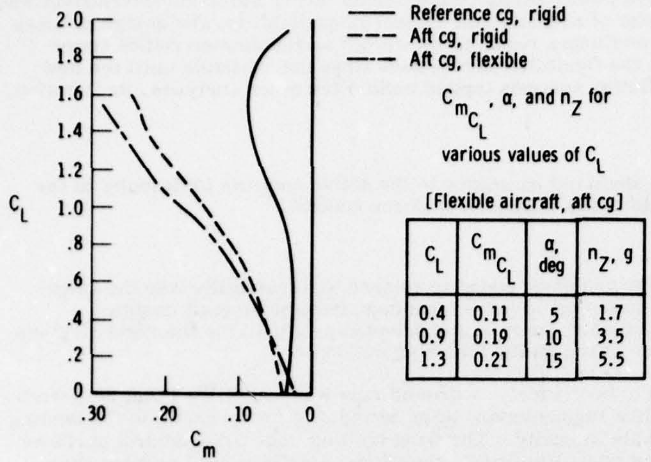
The data for Mach 0.9 are presented in figure 4(a). The curves show the flexibility effects for the flight condition at an altitude of 9140 meters with the indicated instability. The rigid curve corresponds to the reference cg position (0-percent $\bar{c}_w$) and is obviously stable. With the cg at 10-percent $\bar{c}_w$, which corresponds to the farthest aft cg position, the curve shows essentially neutral stability. The inserted table presents the magnitude of instability for several values of C_L. In addition, the approximate α and n_Z values corresponding to each C_L value are listed for a weight of 1390 kilograms. The instabilities shown range from 0.06 at a C_L of 0.4 to 0.12 at a C_L of 1.3 ($n_Z \approx 8.9g$).

Figure 4(b) illustrates similar data for the Mach 0.7 condition at an altitude of 9140 meters. The instabilities range from 0.11 to 0.21 for the C_L values shown. The Mach 0.2 data are presented in figure 4(c) and the cg effects are shown. (Flexibility effects are negligible at this Mach number.) For an altitude of 7620 meters, the instabilities range from 0.10 to 0.31. At such extremely negative stability levels, the pitch control surface will always reach its limit at some sufficiently large angle of attack; thus, stability augmentation will be lost. It should be noted that control power saturation does not occur in the maneuver portion of the HiMAT RPRV flight envelope and hence, does not restrict the demonstration of the highly maneuverable capabilities. However, to preserve vehicle control during low dynamic pressure, high angle-of-attack flight, an angle-of-attack limiter was planned for incorporation in the control system.

Some penalties were incurred because of the RSS system requirements. Larger hinge moments, resulting in larger actuators, were required. This necessitated going outside the wing mold lines on the RPRV and probably would cause a similar problem on a full-scale fighter aircraft. Although not quantified, the weight for the larger hydraulic system was greater than that required for a conventional design.

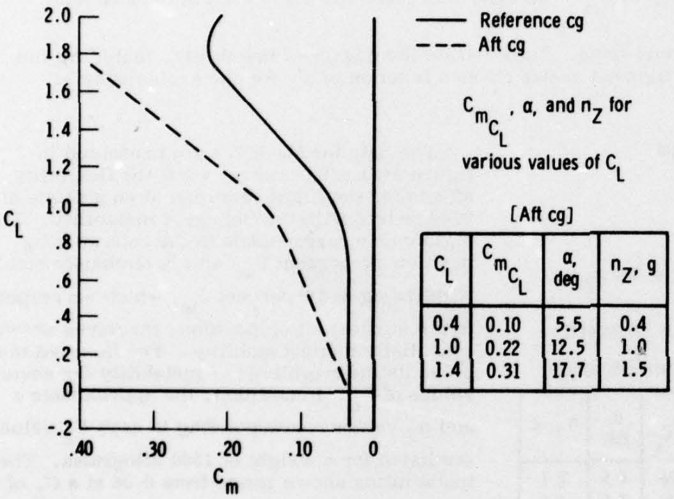
Relaxed directional static stability

As in the longitudinal RSS design, incorporation of RSS in the directional axis required early estimates of control power requirements. Initially, a goal of neutral directional stability based on a rigid airplane was established. Any further decrease in the stability level was not warranted on the basis of performance improvement alone. Even at neutral stability, an initial estimate of the control power required to provide adequate directional stability was needed. The controls engineers decided to state their requirement in terms of $C_{n_{\delta_r}}/C_{n_\beta}$ evaluated at an angle of sideslip of 2°. $C_{n_{\delta_r}}$ is negative and C_{n_β} is normally positive, so the



C_L	$C_{m_{C_L}}$	α, deg	n_Z, g
0.4	0.11	5	1.7
0.9	0.19	10	3.5
1.3	0.21	15	5.5

(b) Mach 0.7, h = 9140 meters.



C_L	$C_{m_{C_L}}$	α, deg	n_Z, g
0.4	0.10	5.5	0.4
1.0	0.22	12.5	1.0
1.4	0.31	17.7	1.5

(c) Mach 0.2.

Figure 4. Concluded.

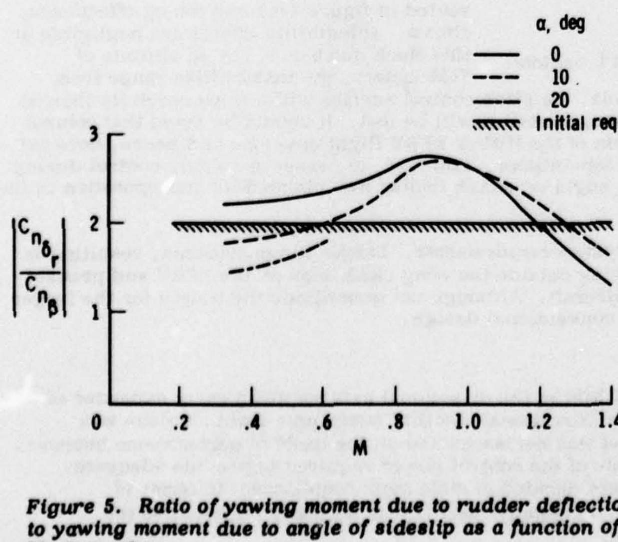


Figure 5. Ratio of yawing moment due to rudder deflection to yawing moment due to angle of sideslip as a function of Mach number.

$C_{n_{\delta_r}}/C_{n_\beta}$ ratio is negative. For a neutrally stable airplane, C_{n_β} is zero and the $C_{n_{\delta_r}}/C_{n_\beta}$ ratio goes to infinity. In other words, very little rudder effectiveness is needed to stabilize a neutrally stable airplane in the directional axis. The control power requested by the controls group and based on the rigid airplane was ultimately about 50-percent higher than that provided. Figure 5 shows the wind tunnel values for a range of Mach numbers and angles of attack as compared with the initial request. Even though the control power is below the initial request, simulation studies have indicated it is adequate.

The flexibility effects played a major role in the design. Wind tunnel data, corrected for flexibility and cg effects, predict a nonlinearity in C_{n_β} (fig. 6), resulting in negative directional stability to 1.5° of sideslip. This nonlinearity is caused by the flexible characteristics of the vertical tails and the winglets. Because these surfaces are aft of the cg, the flexibility effects tend to negate the directional stability the surfaces would otherwise provide.

The tendency to trim directionally at nonzero angles of sideslip would have been undesirable without active controls. However, with active controls it could be assumed that the control system could be programed to remove any such tendency.

Direct force control

Direct force control is provided in two axes. Normal force is provided by the direct lift control system; side force is provided by the direct side force control system.

The direct lift control system utilizes the canard flap, in conjunction with the wing trailing edge surfaces, to provide pure lift. A canard flap normally produces nose up pitching moment as well as direct lift. However, with the HiMAT RPRV, the symmetric elevons and elevators are deflected to a trailing edge down position by the active control system to trim out the pitching moment. This effectively increases the wing camber, which increases the total vehicle lift. Interestingly, the canard flap lift effectiveness reverses at dynamic pressures above 30,000 N/m^2 because the canard flap trailing edge down deflection causes downwash, which reduces the angle of attack at the wing, thereby reducing lift. However, a positive pitching moment is still generated, which results in the same downward deflections of the wing trailing edge surfaces through the active control system, and hence, direct lift as before.

Direct side force is achieved by deflection of the antisymmetric canard flap in conjunction with the rudders and antisymmetric wing trailing edge surfaces. To generate the side force with the canard flaps, dihedral was required in the canards. The dihedral ahead of the cg was destabilizing directionally. The wingtip ventrals were added to return the rigid airplane to neutral directional stability at the 0-percent $\bar{c}_w$ reference cg.

PRIMARY CONTROL LAWS

The primary control laws were designed to meet the basic military handling qualities

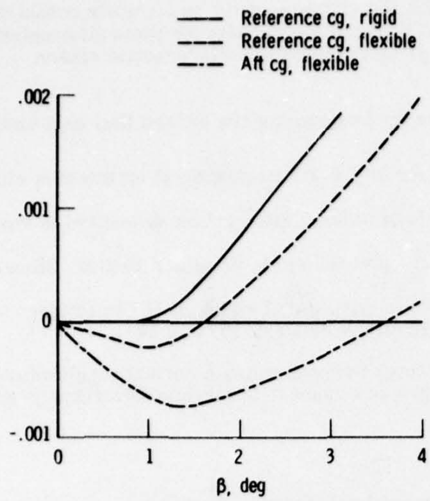


Figure 6 Yawing moment in stability axis system as a function of angle of sideslip. Mach 0.9; h = 3050 meters.

specification from MIL SPEC-F-8785B(ASG) (ref. 6). Therefore, the relationship between the control law design and the configuration development for the HiMAT RPRV should have been similar to the relationship for a full-scale manned vehicle. The design philosophy chosen is discussed, and the longitudinal and lateral-directional primary control laws are described.

Design Philosophy

The most significant assumption was that a full-time digital fly-by-wire system would be available to connect the pilot's inputs and a standard set of motion sensors to all the control surfaces. The digital computer was assumed to be sufficiently fast and large to handle whatever control laws were necessary to meet the handling qualities specifications. Reference 6 was to be used as a guide, with piloted simulation as a means for final refinement. The design goal was to provide constant handling characteristics throughout the flight envelope.

The control laws were initially designed in the continuous domain using classical design methods and later were converted to a discrete set.

Longitudinal Axis

Control law structure

The longitudinal axis control system is shown functionally in figure 7. The pilot's pitch stick input, δ_{e_p}, generates a normal

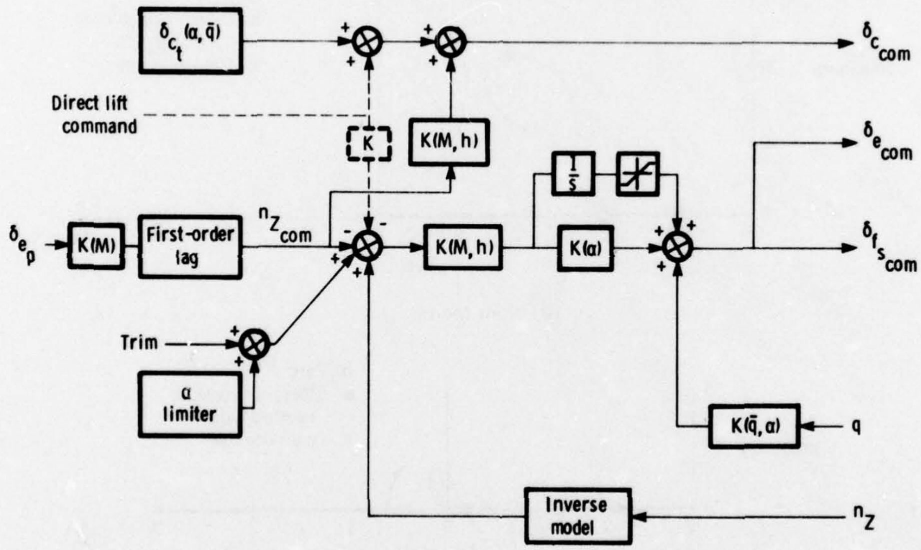


Figure 7. Longitudinal primary control law structure.

acceleration command, $n_{Z_{com}}$, through a first-order shaping filter and a gain. The value of $n_{Z_{com}}$ is compared with a filtered normal acceleration feedback, and the error is then routed through an integral-plus-proportional network in the forward loop to the elevators and symmetrical elevons. Note that many of the gains are functions of one or two variables. The forward loop gain, for example, is programed as a function of Mach number and altitude. An integral-plus-proportional network, programed as a function of angle of attack, provides neutral speed stability within the limits of the integrator limiter. Pitch rate is fed back as an inner loop to provide pitch damping. The elevators and symmetric elevons are driven in unison as the primary pitch control effectors. In the normal acceleration feedback path, an inverse model places zeros at a desirable location for the closed-loop short period to close on as the gain is increased.

A crossfeed is provided from the normal acceleration command, $n_{Z_{com}}$, to the canard. This crossfeed is needed to quicken the normal acceleration command augmentation response at high altitude, transonic flight conditions.

The angle-of-attack limiter introduces a nose down pitching moment command when a reference angle of attack, α_ℓ, is approached or exceeded. The α_ℓ is programed as a function of Mach number: At Mach 0.3, α_ℓ is equal to 10°; at Mach 0.8 and greater, α_ℓ is equal to 19°; and between these two points, α_ℓ is interpolated linearly.

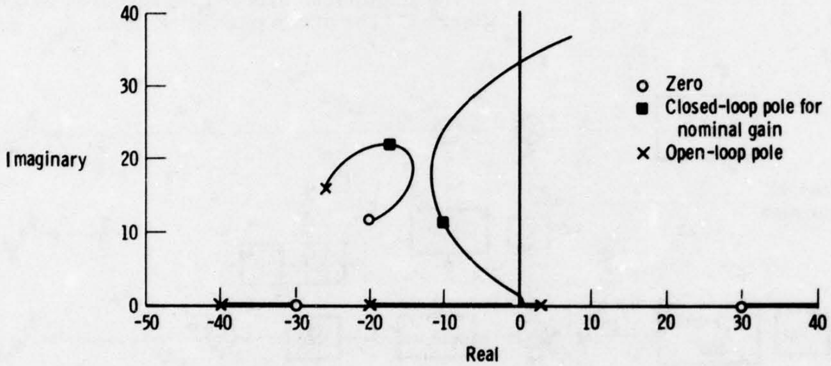
The angle-of-attack limit was based on the angle of attack at which the airplane could be abruptly rolled to a 30° bank angle using the antisymmetric elevons without reaching the limits of the elevons for pitch augmentation. If an elevon reaches its limit, the symmetric elevon commands have priority over the antisymmetric elevon commands.

An attempt has been made to straighten the nonlinear $C_{m_{C_L}}$ curve by programing the canard flap as a function of angle of attack. This is reflected in the δ_{c_t}, which is only active for $M \leq 0.7$. Beginning at an angle of attack of 10°, the δ_{c_t} is a trailing edge up command proportional to the angle of attack until surface saturation is reached at an angle of attack of 16°. There is close interaction between this δ_{c_t} and the angle-of-attack limiter, since they are active concurrently. The limiter begins making nose down inputs at an angle of attack of 10° to 12° for $M \leq 0.4$, whereas the canard flap makes nose down inputs at angles of attack between 10° and 16°.

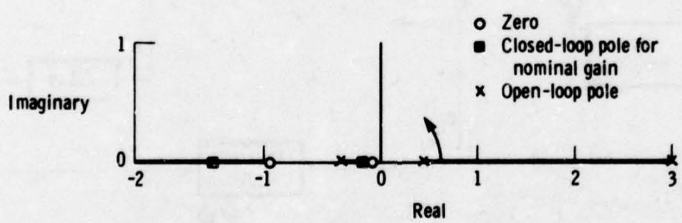
Also shown in figure 7 is a separate direct lift command (dashed line) that generates a normal acceleration command, which is acted upon by the closed loop through the elevators and symmetric elevons described previously. In addition, a signal feeds directly to the canard flap.

Dynamic characteristics

As mentioned previously, an inverse model was included in the normal acceleration feedback to give a set of zeros at a desirable location for the closed-loop poles to close on. Root loci were generated by varying the normal acceleration feedback gain with the pitch rate feedback loop closed and assuming the first-order actuator lags to be 40 radians per second. Figure 8 presents a root locus for a high dynamic pressure flight condition of



(a) Root locus.



(b) Expanded view of root locus near origin.

Figure 8. Root locus for increasing normal acceleration feedback with pitch rate loop closed. Mach 0.9; h = 760 meters.

Mach 0.9 and an altitude of 760 meters. The closed-loop root locations for five representative flight conditions are shown in figure 9. The root locations and, in particular, the real zero pairs vary considerably with flight condition. Despite this variation, the dynamics of the transient response to a pilot's command is relatively invariant, even though the static response is different. An example is given in figure 10, which shows simulator time responses for identical pilot inputs at two flight conditions.

Lateral-Directional Axes

Control law structure

In the lateral-directional axes, the control laws are conventional, with the exception of an integral-plus-proportional network on lateral acceleration and the addition of a rudder pedal-to-antisymmetric elevon interconnect. Three sets of controllers are available: ailerons, antisymmetric elevons, and rudders. Figure 11 presents a block diagram of the roll-yaw normal mode control laws. As in the longitudinal axis, many of the gains are programed as functions of several variables to maintain nearly constant handling qualities.

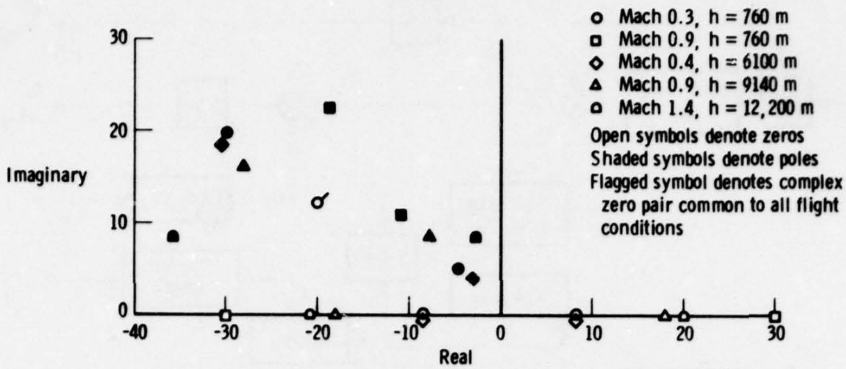


Figure 9. Closed-loop roots in longitudinal axis at selected flight conditions. Closed-loop pole-zero pairs near the origin are similar for all flight conditions.

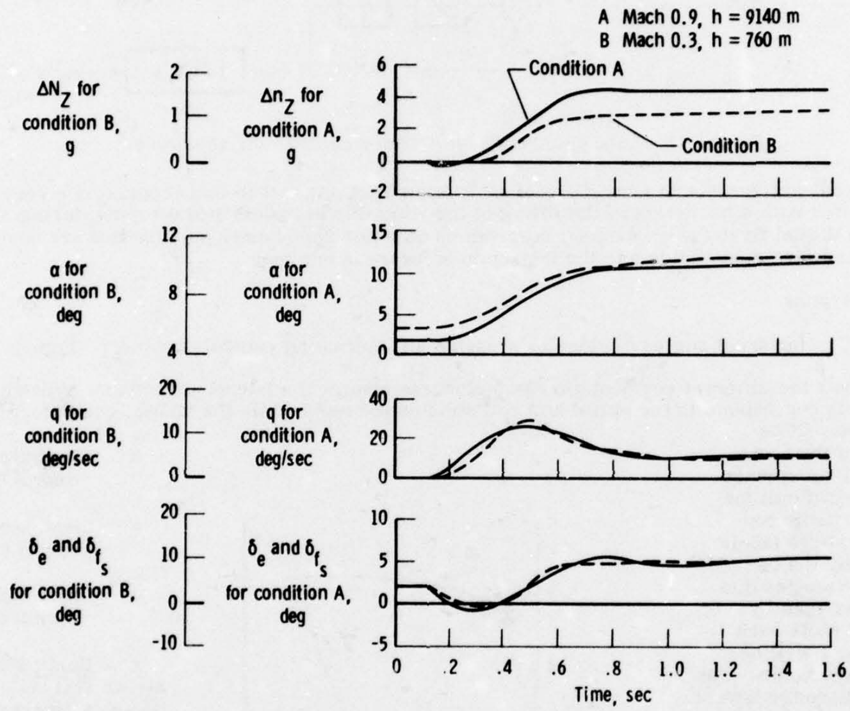


Figure 10. Time response for 3-centimeter pilot pitch stick step command at two flight conditions.

Beginning at the top half of figure 11, the pilot's roll stick signal, δ_{a_p}, is routed through a gain programed as a function of Mach number, then low passed to generate a roll rate command. The command is routed to the ailerons through a scheduled gain. The gain is constant for dynamic pressures less than 38,400 N/m^2. The gain is gradually reduced at higher dynamic pressures such that it is zero for dynamic pressures greater than 57,000 N/m^2 (aileron roll effectiveness changes sign due to flexibility at high dynamic pressure). The roll rate command is also routed to the antisymmetric elevons, which are used to augment roll damping through roll rate feedback. A roll stick-to-rudder interconnect is provided to decrease a strong adverse yaw. A lag on the interconnect with a 1-second time constant provides compatibility between the interconnect and the augmented roll subsidence mode.

Rudder pedal inputs, δ_{r_p}, are converted to directional commands in a manner similar to that for the roll stick signal. Augmentation of the Dutch roll is accomplished through lateral acceleration and yaw rate feedback. The limited integral-plus-proportional network on the lateral acceleration feedback is a direct result of the nonlinear C_{n_β} curve discussed in an earlier section. Without the integrator, directional trim may occur at non-zero angles of sideslip. With the active control philosophy, the control system is expected to provide directional trim at an angle of sideslip of 0°. If the active control system were not available, it would be necessary to stiffen the vertical tail structure to eliminate the C_{n_β} nonlinearity.

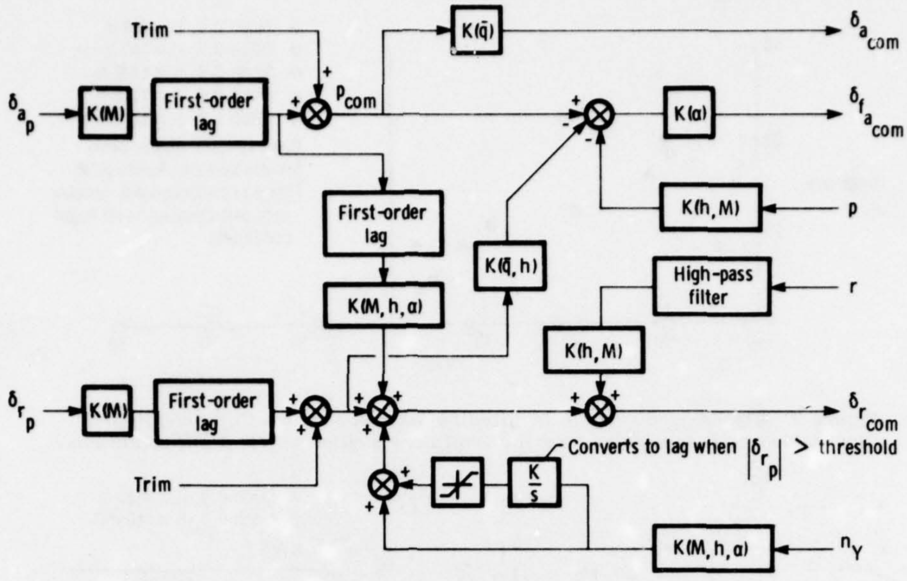


Figure 11. Lateral-directional primary control law structure.

The rudder pedal-antisymmetric elevon crossfeed is somewhat unusual in that it corrects a very minor annoyance associated with a hesitation in the lifting of the wing due to rudder pedal inputs during a crosswind approach. With a digital fly-by-wire system, corrections of minor annoyances such as this are essentially free in that the hardware is available and the impact on software is minimal.

Dynamic characteristics

The negative C_{n_β} for small angles of sideslip presents an interesting controls problem. Figure 12 shows, in root locus form, how the different augmentation loop closures change the lateral-directional dynamics. The two real, unstable roots correspond to the spiral and roll subsidence roots while the stable, oscillatory pair represent the Dutch roll roots. Note that the labeling of the roots as Dutch roll, roll subsidence, or spiral becomes difficult for these unusual, unstable configurations. The above labels were determined by the behavior of the mode shapes due to the roots, rather than by tracing the loci of roots back to their origins for a well behaved stable configuration. The initial closure, where the lateral acceleration feedback includes the integral-plus-proportional network, moves the unstable roots into the stable left half plane while decreasing the stability of the oscillatory pair. The second closure, where high-passed yaw rate is fed back, improves the damping as expected. In the final closure, the addition of the roll rate feedback improves the roll subsidence time constant by moving the real root from -2.75 to -3.95.

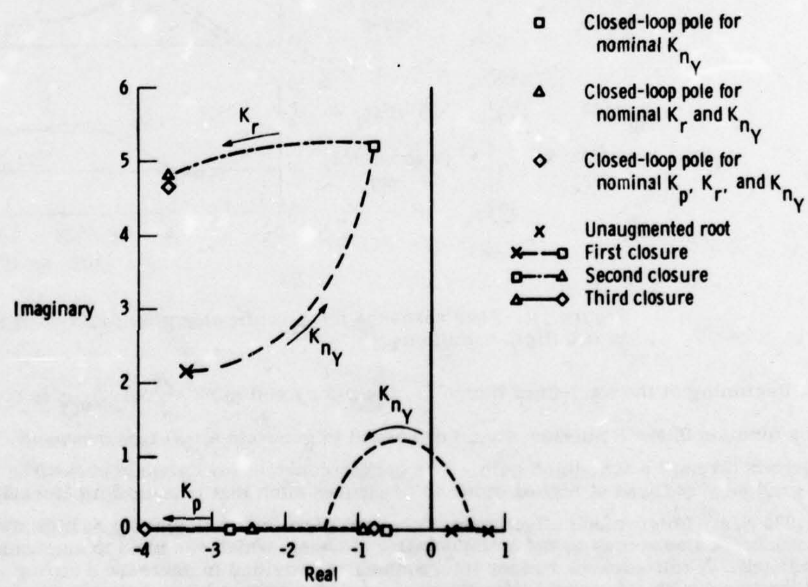


Figure 12. Effects of augmentation loop closures on the lateral-directional roots. Mach 1.0; h = 3050 meters.

Extensive programing of control system gains as functions of flight condition parameters was usually required. As an example, a comparison of the pole-zero root contours for the bank angle due to roll stick transfer function with and without gain variations with angle of attack and without roll-to-yaw interconnect is shown in figure 13. It is evident that the interconnect maintains roll control for high angles of attack; in other words, no sign change occurs in the numerator of the bank angle due to roll stick transfer functions. The angle-of-attack gain variation case shows that the zeros remain close to the corresponding poles, whereas for the fixed gain with angle of attack case, the poles and zeros become greatly separated. When the poles are close to the zeros, the roll rate response is well behaved; when the poles are well separated from the zeros, a large component of Dutch roll is present in the roll rate response.

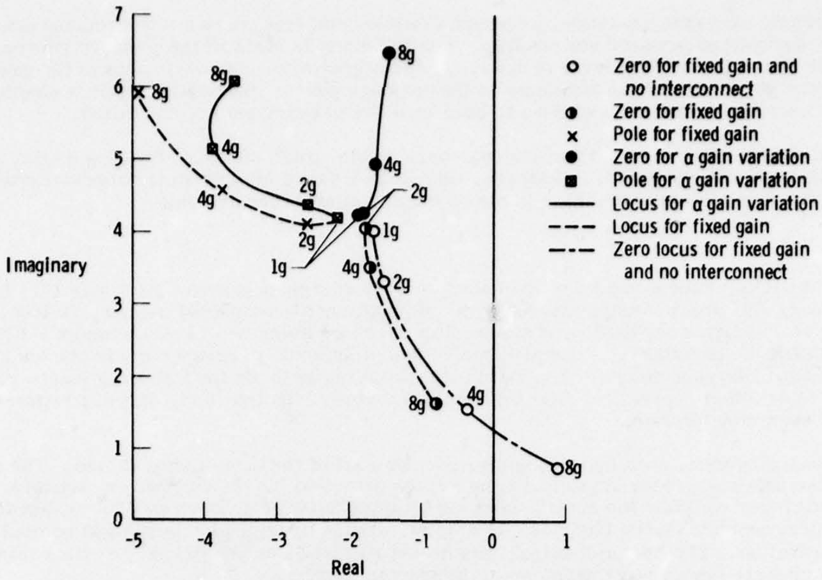


Figure 13. Effects of angle-of-attack gain schedule on bank angle due to roll stick transfer function roots.

CONTROL SYSTEM IMPLEMENTATION FOR RPRV OPERATION

The procedures followed up to this point are generally applicable to a full-scale manned aircraft. The information presented in this section, however, is unique to an RPRV operation. No attempt is made to generalize this information to a manned aircraft.

Primary Control System

Figure 14 presents a diagram of the principal elements in the control system. Separate input/output interfaces are included in each of the interfaces with the computers, although none are shown in the figure. There are three flight-safety-critical sensor sets: two within the primary system and one in the backup system. Sensor information is transmitted to the ground by way of the flight test instrumentation system. The control laws are implemented on the ground-based minicomputer, operating on the motion sensor and cockpit command information. The resulting control surface commands are transmitted to the airplane at the rate of 53.3 samples per second. Both receiver/decoders are required (eight control surface commands at four commands per unit) to transmit the commands to the primary microcomputer, which forwards the commands to the appropriate control actuators.

The control laws involve substantial use of nonlinear gain scheduling on multiple flight condition parameters. The principal elements are discussed in the following sections.

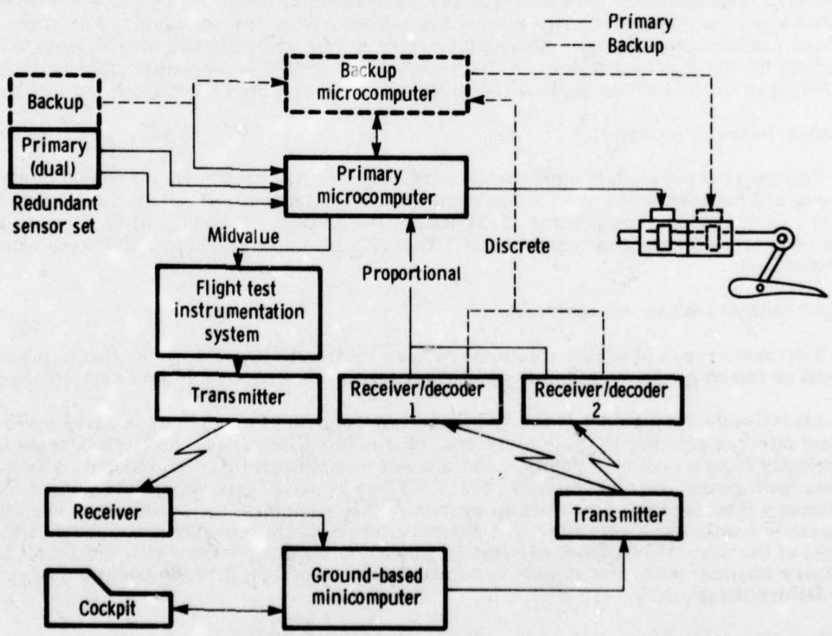


Figure 14. Arrangement of flight control system.

Sensor redundancy management

The flight-critical flight control sensors, the rate gyros (p, q, and r), and the normal and lateral accelerometers (n_Z and n_Y) are triplexed. The outputs of these sensors are transmitted to the primary microcomputer where the midvalue is selected. This midvalue is transmitted to the ground for primary system control loop closure. The other two values are monitored against the midvalue to within a specified tolerance for failure detection. If an out-of-tolerance condition is detected, the ground is notified as to which sensor has failed, thus aborting the mission, but the RPRV remains on the primary system.

The air data sensors (dynamic pressure, pressure altitude, and free-stream temperature) are dualized at the transducer and are designated primary and backup. A comparison is made in the primary microcomputer. When a difference greater than a prescribed level is detected, a disagreement discrete is sent to the ground. The ground-based minicomputer checks the pressure or temperature sensor information against simple models to determine which sensor has failed and switches to backup if the primary sensor has failed.

Sensors for non-flight-safety-critical parameters (bank angle, pitch attitude, heading angle, angle of attack, and angle of sideslip) are not redundant. However, the ground-based minicomputer checks these simplex sensors to determine when the data exceed reasonable limits or change at an excessive rate.

Microcomputers

The primary and backup microcomputers are based on 8080 microprocessors. Each has 1024 8-bit bytes of random access memory and provisions for 22,000 bytes of programable read-only memory (PROM). The primary microcomputer has 16,000 bytes of PROM, and the backup microcomputer has 14,000 bytes of PROM in addition to a 7-byte intercom between computers. A comprehensive self-diagnostic program runs in the background to the control software in both microcomputers. The diagnostic program includes the following tests: memory check sum, scratch pad, instruction repertoire, real-time clock accuracy, limited input/output wraparound, hardware multiplication, and computer intercom.

While in the primary system, each microcomputer carries part of the computational load. The primary microcomputer does all the datalink processing and all the failure detection for the computers, sensors, and actuators. The backup microcomputer contains the control laws for an integrated propulsion control system that replaces some of the mechanical controls on the basic J85-21 engine. It also provides backup flight control system synchroniztion information. The backup control laws do not run while on the primary system, but constants must be updated to prepare for a smooth transfer to the backup system.

Telemetry downlink and uplink

This system consists of the downlink and uplink telemetry information for control of the RPRV. The telemetry links are essentially line-of-sight transmission paths. It is estimated that flight operations will be limited to a range of approximately 60 kilometers at an altitude of 1500 meters and a range of approximately 200 kilometers at an altitude of 14,400 meters.

The downlink system provides aircraft response variables to the ground station at 220 frames per second. Approximately 202 data parameters, including 25 flight control data words, are packed into 75 words per frame. Parameters that contain useful, high frequency information are sampled at 220 samples per second; parameters of lower frequency signal content are sampled at 55 samples per second. The pulse code modulation system is a 10-bit system plus parity, although parity is not checked. A 3.7-meter parabolic receiving antenna, slaved to a radar tracking antenna, receives the transmitted signal.

The uplink system contains 16 bits per data word (a 10-bit proportional command signal and six discrete signals). Transmission is at a rate of 106.66 frames per second with four data words per frame. Since eight uplink words are required, two frames are necessary for transmission. The effective rate is 53.3 samples per second per command signal. Although two parity bits are transmitted with each word, only one bit per frame is checked by the microcomputer. A discrepancy is handled as improper information, and the system is automatically transferred to the backup mode if the discrepancy is repeated a prespecified number of times.

Ground-based minicomputer

The general purpose ground-based minicomputer has 32,000 16-bit words of memory, with a memory cycle time of 330 nanoseconds. A set of peripherals is available to support the minicomputer, including a magnetic tape drive, card reader, line printer, 2.34-million-word disc memory, and high speed paper tape reader and punch. The control laws are programed in FORTRAN IV with the input/output software written as assembly language subroutines.

Flight control surface servoactuators

Two basic types of servoactuators are used on the RPRV: fail-safe, through hydraulic locking to a predetermined or faired position; and fail-operative, through a dual-redundant (active-standby) implementation.

All fail-safe simplex actuators (canards, ailerons, and elevators) use a cross-ship monitoring technique to detect failures relating to their individual channels. Since these surfaces operate either in unison or antisymmetrically from a centered position and are not mechanically interconnected, a simple comparison of their position linear voltage differential transformers (LVDT's) is used. An out-of-tolerance difference in any of the three pairs initiates a total switch to the backup system. This switching is, however, reversible. In the fail-operative duplex actuators (rudder and elevon), an airborne microcomputer comparison (model) testing technique is employed. A model of the servovalve spool position is generated for comparison with the actual LVDT-measured position. If the primary channel fails, the system is automatically transferred to the backup mode. This switching is reversible if the failure disappears.

Backup Control System

The backup control system was designed to recover and return the vehicle to a stable attitude after a primary system failure; place the vehicle in a constant bank angle orbit mode until commanded to exit orbit; permit remote control of the vehicle from the ground or a chase aircraft by discrete command; provide emergency landing capability; and provide a glide mode for the best range with the engine out.

Functional description

The above functions could be accomplished through the use of only the elevons, rudders, and throttle. An autopilot was designed with the nine operating modes described below.

The RECOVER mode returns the vehicle to a stable attitude after a switch to the backup system. Although the attitude gyros are not required, they are used to update the direction cosines, which are used to transform rate gyro outputs to attitude signals. Once recovery is complete, the system automatically switches to the ORBIT mode. The RPRV climbs or descends to a prespecified altitude, maintaining a constant turn rate.

Either the ground station or the chase airplane can initiate the EXIT ORBIT mode. The RPRV then goes to wings-level flight, maintains altitude, and automatically transfers to the STRAIGHT AND LEVEL mode. This mode maintains a straight track on the last heading and holds altitude. The DIVECLIMB and TURN modes may be commanded by the pilot to dive or climb at a predetermined, altitude-dependent altitude rate, or to turn at a bank angle of 35°.

Landing can be initiated by either command station by transferring to the LANDING mode. An altitude rate of descent, which is a function of radar altitude, is then maintained by the autopilot. The altitude rate can be modified by the pilot. A MACH COMMAND mode is also selectable for assisting with speed control during the landing approach.

If the engine is out, the ENGINE OUT mode is automatically selected. The pilot can make discrete speed changes and has access to the TURN MODE to make heading changes.

Implementation

The backup system uses sensors that are part of the redundant sets discussed previously and shown in figure 14. One sensor of each set is designated the backup sensor. If the backup sensor is in a triplex set, it is routed to both microcomputers. If the backup sensor is in a duplex set, it is connected only to the backup microcomputer. The primary microcomputer receives the backup sensor information indirectly through the computer intercom from the backup microcomputer. In either case, the backup microcomputer receives the backup sensor data directly.

The backup microcomputer accomplishes all necessary processing while in the backup mode. It processes all the uplinked discrete commands, the full set of backup control laws, and a subset of the integrated propulsion control system control laws. If the primary microcomputer is operable, it continues to process downlink data and a subset of failure detection software.

The rudder and elevon servoactuators are fail-operative, active-standby devices as described previously. The standby channels are switched on if transfer is made to the backup system.

Through a set of toggle switches, either the ground cockpit or a chase airplane can command mode changes within the backup system. The ability to make discrete changes to the altitude rate schedule in the LANDING mode is also provided at these command stations. The commands are in the form of discretes received by either receiver/decoder. A full set of commands can be received even if one of the receiver/decoders has failed.

CONCLUDING REMARKS

The design of a HiMAT RPRV has been completed, with active controls, a major new technology, incorporated in the design. The HiMAT RPRV is part of a new method for bringing advanced aircraft technologies to a state of readiness. The method involves the design and flight test of an RPRV based on a full-scale manned vehicle design incorporating the technologies of interest.

An examination of the active controls design process that resulted in the HiMAT RPRV reveals the following factors, which probably have general applicability to the design of full-scale operational airplanes of similar planform:

1. The active controls design process differed from conventional design processes in that although the control power sizing requirements were needed at the start of the design iteration, sufficient information to make such specifications were only available at the completion of the iteration. Better rough order-of-magnitude guidelines are needed to begin the iteration cycle.

2. Although a maximum of 10-percent negative static margin was used as a guideline for relaxed longitudinal stability and was increased to 15 percent later, nonlinearities in $C_{m_{C_L}}$ led to more than 30-percent negative static margin for some high angle-of-attack flight conditions at low Mach numbers. As a result, an angle-of-attack limiter was required to assure adequate excess control authority to stabilize the aircraft.

3. Neutral directional stability was selected as a limit for the rigid airplane. However, flexibility effects caused negative stability for small angles of sideslip. A relaxed directional static stability system was required with special provisions to prevent trimming to nonzero angles of sideslip.

4. Some penalties were incurred because of the active control functions. Actuators and hydraulic systems were larger than those required for an aircraft of conventional design. Installation of some actuators outside the wing mold line was required. Use of canard flaps to generate direct side force led to increased canard dihedral, and the addition of wingtip ventrals was necessary to compensate for the destabilizing canard dihedral effect.

5. Direct lift was provided by the canard flap and wing trailing edge surfaces' working together. The active control system deflects the wing trailing edge surfaces to counteract the nose up pitching moment due to a downward canard flap deflection. This effectively increases wing camber without inducing a net aircraft pitching moment, resulting in direct lift.

6. The primary control system was designed under the assumption that a full-time digital fly-by-wire system was available to interconnect the pilot's commands and the motion sensors with all the control surface actuators. The resulting control laws involved substantial scheduling of nonlinear gains as functions of multiple flight condition parameters.

Although the control system implementation for RPRV operation cannot be generalized to an operational piloted airplane, a versatile system suitable for remotely piloted flight research was defined. The system makes effective use of a ground-based minicomputer and two onboard microcomputers.

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ACKNOWLEDGMENT

The authors are indebted to Marshall Roe of Rockwell International for his contributions concerning the aerodynamic configuration development experience with the HiMAT RPRV.

PROPULSION-FLIGHT CONTROL INTEGRATION TECHNOLOGY

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SUMMARY

This paper describes the concept of propulsion-flight control integration technology (PROFIT). The PROFIT concept is to be implemented on a high performance supersonic twin-engine aircraft which will make possible the evaluation of a wide variety of integrated control concepts. The aircraft's inlet, engine, and flight control systems are to be integrated with a digital computer. The airplane control hardware is to be modified to provide the necessary capability for control research; software will be used to provide flexibility in the control integration capability. This paper describes the background for flight and propulsion control system development and probable future trends. It also discusses the PROFIT concept, design philosophy, and approach. Examples of integrated control research that have application to future aircraft designs are also presented.

1.0 INTRODUCTION

Many high performance aircraft exhibit strong coupling between the propulsion system (inlet, engine, and nozzle) and the airframe. Such aircraft include supersonic cruise, hypersonic, and short and vertical takeoff and landing aircraft and highly maneuverable fighters. The achievement of optimum performance and cost effectiveness for these aircraft requires the propulsion control systems and flight control systems to operate in harmony with each other. Studies have indicated that the integration of the propulsion and flight control systems improves aircraft performance, reduces pilot workload, increases safety and reliability, and reduces costs (Refs. 1 to 3). However, the integration of propulsion and flight control systems is complex because of the large number of system inputs and outputs, the nonlinear nature of some of the system components, and the difficulty of accurately modeling the interactions between the components.

To study these problems, a program called PROFIT (for propulsion-flight control integration technology) has been conceived. In the program, a flexible flight research facility is to be developed that is capable of evaluating a wide variety of integrated control concepts. The PROFIT system would be implemented on a high performance supersonic twin-engine fighter aircraft capable of addressing the integration problems associated with highly maneuverable fighters and supersonic cruise aircraft. This paper describes the design philosophy for the PROFIT concept and describes some of the research that could be conducted with the PROFIT airplane.

2.0 BACKGROUND

2.1 Control Requirements

The evolution of flight and propulsion control systems has followed the trends shown in Figure 1. The early jet-powered aircraft developed in the 1940's had simple mechanical flight control systems and simple turbojet engines with hydromechanical fuel flow controls.

The next generation of aircraft, introduced in the 1950's, had afterburning turbojet engines to provide greater thrust and analog electronic stability augmentation systems to provide acceptable handling qualities at transonic speeds.

The aircraft introduced in the 1960's typically had variable-geometry inlets, autopilots, and air data computers. The turbofan engine was introduced. However, there was little or no integration between the propulsion and flight control systems.

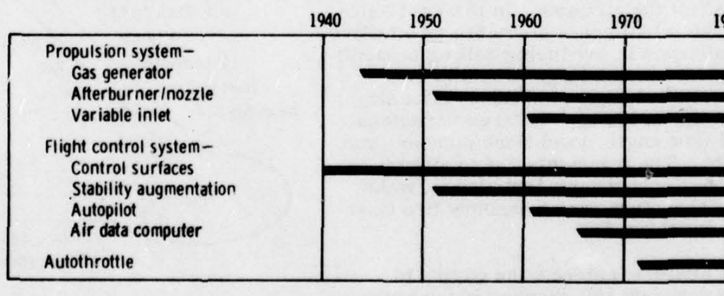

Figure 1. Evolution of flight and propulsion control systems.

The first step in integration, the autothrottle, has been used on some of the aircraft introduced in the 1970's, primarily for the landing approach. For other flight regimes, however, the propulsion and flight controls operate independently.

This gradual increase in the complexity of the flight and propulsion control systems has resulted in a steady increase in the number of controllers, as shown in Figure 2 and the adjacent table. The jet fighters of the late 1940's had only aileron, elevator, rudder, and fuel flow controls. The afterburning turbojet engines of the 1950's added afterburner fuel flow, primary nozzle, and variable stators or bleeds to the engine control task, and the hydromechanical controls became quite complex. The variable-geometry inlets of the 1960's necessitated the development of another control system, and spoilers came into common use. The control requirements of the afterburning turbofan engine pushed hydromechanical control to its limit. The aircraft of the 1970's incorporate such new features as maneuver flaps, digitally controlled inlets, and even more complex engines with supervisory digital controls.

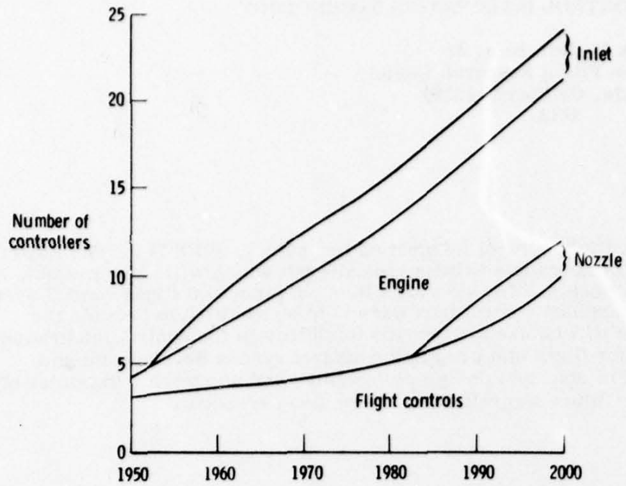

Figure 2. Number of controllers for propulsion and flight control systems of high performance aircraft.

	Controllers
Inlet	Bypass Spike or ramps Variable capture, bleeds
Engine	Engine fuel flow Afterburner fuel flow Primary nozzle Bleeds Variable stators Inlet guide vanes Secondary nozzle Variable turbine area Variable bypass ratio
Nozzle	Vectoring, reversing Noise suppressor
Flight controls	Elevator Aileron Rudder Spoiler Flap/slat Canard Direct side force surfaces Surfaces for active control technology

The next generation of aircraft may incorporate direct lift and side force control surfaces, which must also be integrated into the flight control system. Future variable-cycle engines may incorporate variable turbine area and variable bypass ratio to improve off-design performance, as discussed in Reference 4.

In the long term, active control technology is expected to require additional control surfaces for such functions as active flutter suppression, gust alleviation, structural mode suppression, and ride smoothing. The engine may have as many as 10 control parameters (Ref. 5), and the nozzle may be used for thrust vectoring and reversing or noise suppression. Obviously, the large number of controllers creates a challenge in terms of control.

2.2 Interactions

Strong interactions may be expected between the engines, inlets, and airframe in all high performance aircraft (Ref. 1). Figure 3 shows these potential interactions.

Inlet-airframe interactions are relatively mild in some aircraft, but they may be severe in supersonic cruise aircraft like the XB-70 and YF-12 aircraft, where the inlets generate large amounts of thrust (Ref. 1). Inlets cause pitching, rolling, and yawing moments which affect the airframe. In the YF-12 airplane, the inlet bypass doors are as effective as the ailerons in producing rolling moments. Drag forces may be large when the inlets operate at off-design conditions. The airframe subjects the inlet to large variations in local flow angle, local Mach number, and pressure. The transients due to altitude variations, atmospheric turbulence, wake vortex encounters, and weaponry fire must also be considered.

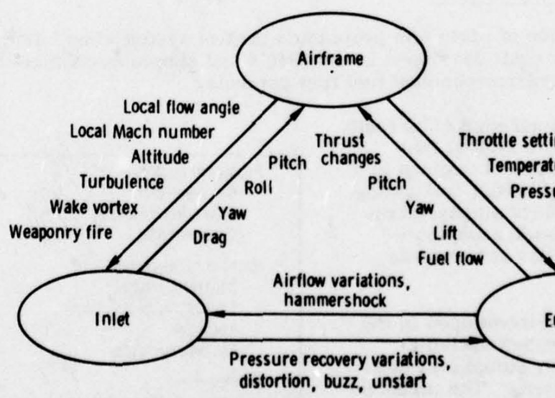

Figure 3. Engine-inlet-airframe interactions.

The airframe subjects the engine to throttle demands and changes in temperature, pressure, and Mach number as the flight conditions change. As the engines consume fuel, changes in the airframe center of gravity may become significant. Thrust changes may affect airframe trim in the pitch and yaw axes, particularly for wing-mounted engines. Engine-airframe interactions are especially strong in powered-lift or vectored-thrust airplanes.

Airflow transients in the engines cause airflow transients in the inlets, which may be quite rapid and, in the case of the hammershock from compressor stalls, violent. The controls and structure of the inlet must be capable of handling these transients. In return, the inlet supplies the engine with air of varying pressure recovery and distortion, and severe transients due to inlet unstarts and buzz are possible.

The designer of a new aircraft is therefore faced with complex engine, inlet, and flight control requirements, as well as a list of potential interactions that is quite imposing.

2.3 Digital Control

The potential of using digital computation methods for propulsion and flight control has been evident for several years, but it was only in the 1960's that the development of high-speed flight-qualified computers made digital flight and propulsion control possible.

2.4 F-8 Digital Fly-By-Wire

The F-8 digital fly-by-wire (DFBW) program was the first program in which a full-authority digital flight control system was tested without a mechanical backup. It incorporated not only flight control, but autopilot, air data, and stability augmentation functions. Figure 4 shows the major hardware elements of the DFBW system. The system is triply redundant, and a major portion of the software is devoted to redundancy management, fault detection, and fault tolerance. More information on the F-8 DFBW program is given in Reference 6. Propulsion control was not required for the DFBW program, since interactions between the propulsion system and airframe are minimal for the F-8 configuration, which has a single fuselage-mounted engine.

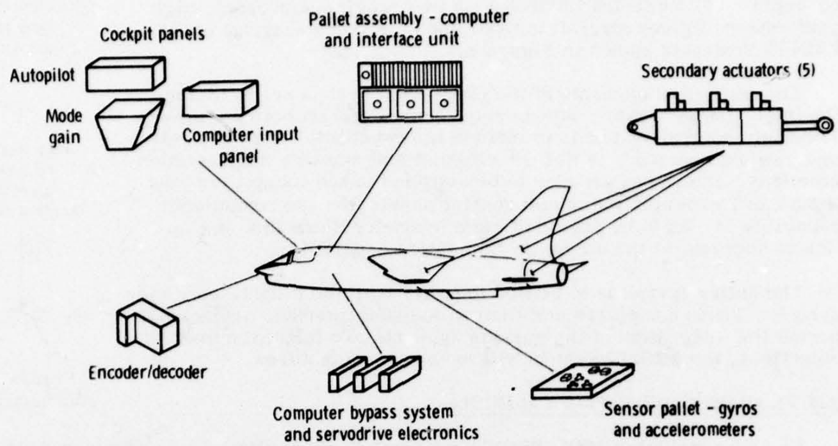

Figure 4. F-8 DFBW hardware elements.

2.5 F-111 Integrated Propulsion Control System

The first flight test program to use digital computers for propulsion system control was the integrated propulsion control system (IPCS) program (Refs. 7 and 8). As shown in Figure 5, this system provided full-authority digital integrated control of the left engine, afterburner, and variable-geometry inlet of an F-111E airplane. No redundancy was provided in the digital control; however, the hydromechanical fuel control was retained for backup. No flight control capability was incorporated, except for an autothrottle on the digitally controlled left engine. Significant performance improvements were realized during the IPCS program; these are summarized in Figure 6. Operation free of compressor stalls was obtained at high Mach numbers and at low Mach number-high altitude conditions where the unmodified right engine did stall. Thrust and specific fuel consumption were improved at the flight conditions indicated by the cross-hatched region. At the flight conditions indicated by the circle symbol, this resulted in an increase of approximately 16 percent in the dash range at Mach 2.2. Throttle response was improved over the entire flight envelope, and idle thrust was lowered. Some of the performance improvements were the result of control integration, while others were due to the improvements in control that were possible because of the additional sensors and computational capability of the digital computer. There was little interaction between the propulsion system and the airframe of the F-111E airplane. However, there were strong interactions between the inlet and the engine and between the gas generator and the afterburner. The IPCS reduced the severity of these interactions substantially.

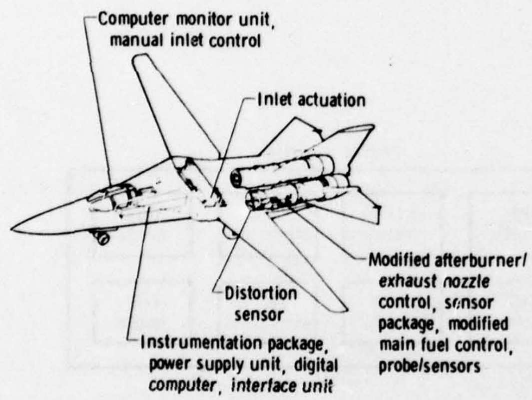

Figure 5. IPCS equipment installed on F-111E airplane.

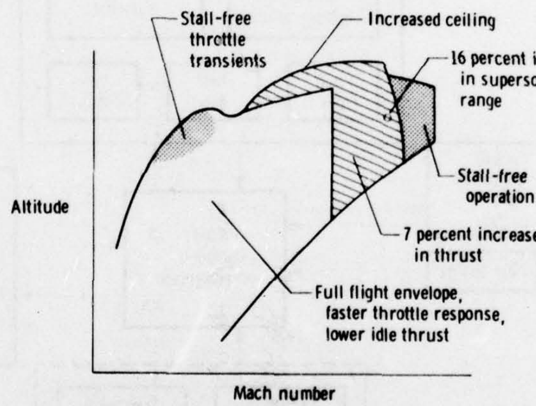

Figure 6. Summary of performance improvements from IPCS program.

The reliability and flexibility of the IPCS electronics were excellent, and it was obvious that the electronic hardware had additional capability that made it suitable for other applications.

2.6 YF-12 Cooperative Control

Strong interactions between the propulsion system and the airframe have been observed with supersonic cruise airplanes such as the XB-70, Concorde, and YF-12 aircraft. These interactions manifest themselves in difficulty in holding precise altitude and Mach number, reductions in stability, and severe pitch, roll, and yaw transients from inlet unstarts or compressor stalls. The severity of the interactions tends to increase with increasing Mach number. A cooperative control program with the YF-12 airplane (Ref. 9) has as objectives taking advantage of favorable interactions and minimizing unfavorable interactions. As shown in Figure 7, the cooperative control program is to digitally implement the autopilot, autothrottle, air data, and inlet control functions on the YF-12 airplane. Engine control is to be limited to the autothrottle. More information on the YF-12 cooperative control program is presented in Reference 9.

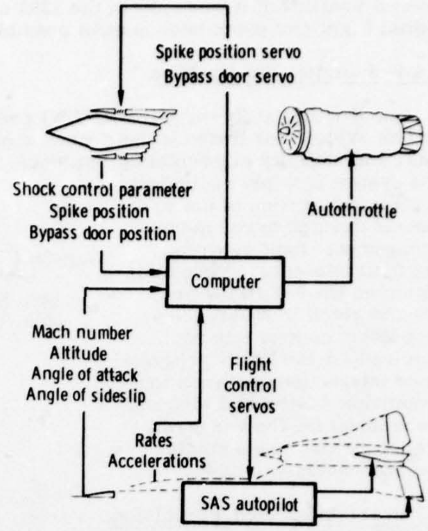

Figure 7. YF-12 cooperative control schematic.

3.0 PROPULSION-FLIGHT CONTROL INTEGRATION TECHNOLOGY CONCEPT

To test the PROFIT concept, a flight research facility is to be used that is capable of evaluating a variety of integrated control concepts. For maximum flexibility, a twin-engine supersonic high performance fighter aircraft is to be used. A block diagram of the PROFIT system is shown in Figure 8.

The controlled elements of the propulsion system are to include the inlet, gas generator, afterburner, and nozzle on both engines. The flight control system is to include the capability for pitch, roll, and yaw axis control, as well as autopilot and stability augmentation functions. Interfaces are also to be supplied to the cockpit for pilot inputs and a cockpit status and control panel. Remote computation capability is also to be provided via a telemetry down link and up link to supplement the onboard computation capability.

The entire system is to be tied together with the PROFIT computer system. These computers and their associated interface devices will permit the integration of the various systems. To fulfill the program objectives, the PROFIT system will have several features.

3.1 Flexibility and Varied Capability

The primary value of the PROFIT research flight facility is to evaluate control schemes that are difficult or impossible to evaluate adequately on the ground and to demonstrate that particular control concepts are ready for use in production airplanes. The development of a flight test program for a single or limited number of objectives is costly and time consuming and difficult to justify. Therefore, a primary objective of the PROFIT concept is to

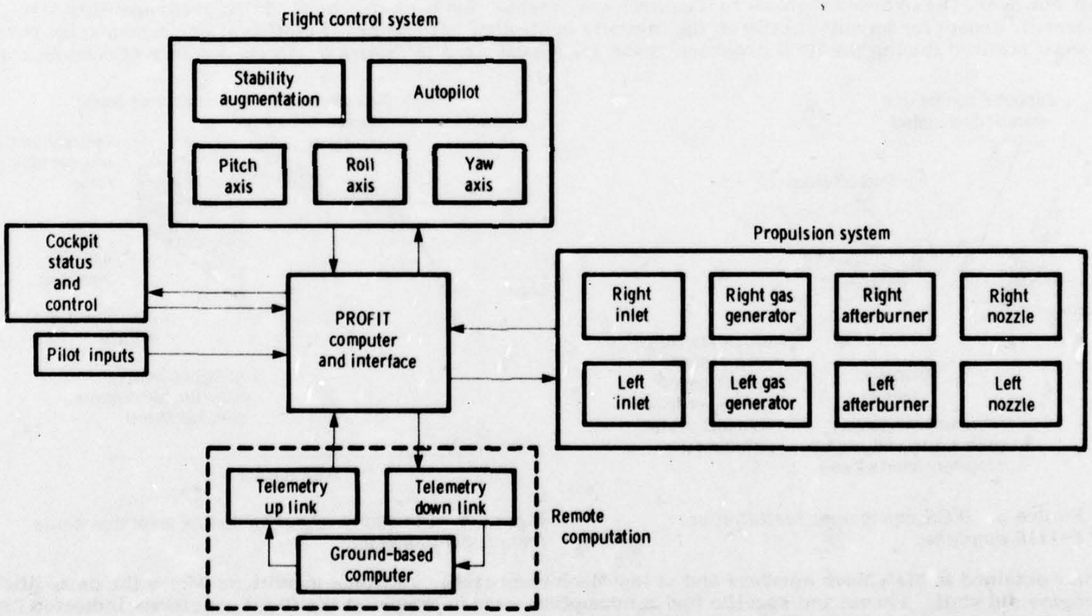

Figure 8. Block diagram of PROFIT system.

provide a facility with the flexibility to evaluate a wide variety of control schemes. The capability for variety is to be achieved by using a combination of hardware and software flexibility. Control hardware for engines, inlets, and flight control systems is to be modified to operate under the control of a digital computer. The modifications are to be designed so as not to limit the control capability of the PROFIT system. Full authority will be provided in most areas. The electronic hardware from the IPCS program will be used for the PROFIT program. This hardware proved to be reliable and has adequate expansion capability to handle all foreseeable requirements. Most of the control integration will be performed in software. Complete changes in control modes can then be made with only software changes.

3.2 Flight Safety Considerations

The control system modifications for the PROFIT program are not to be flight safety critical, at least in the initial phases. The cost in resources and time of making flight-critical modifications is large, and requires extensive documentation.

For non-flight-critical systems, backups for the modified systems can be achieved by careful design, often by using existing systems or limiting the authority of the control modifications. In the case of changes to the propulsion system of a twin-engine airplane, one side can be fully modified and changes to the other side can be minimal or non flight critical. The authority of some mode of the flight control system is often limited so that its inputs can be overridden by the pilot. Certain tests may also be conducted in regions of the flight envelope where failures are not flight critical.

Once a control concept is verified, the development of a flight-critical implementation with adequate redundancy may be desirable. This prevents large expenditures of resources to implement control schemes that are found to be undesirable.

3.3 Remote Computation

One of the difficulties involved in achieving the desired computational capability for control research is handling the programming requirements with the airborne computers, which have a limited capacity. If several control schemes are to be tested during a flight, either all the control algorithms must be entered in a single large program or the software must be changed during flight. Additional computational capability can also be provided by using a ground-based computer that communicates with the aircraft through telemetry up links and down links. This capability, although not necessitated by the PROFIT system, increases its capability and flexibility. If it had been necessary to develop this capability along with the PROFIT system, it might not have been attractive. However, the down link has existed for many years at the Dryden Flight Research Center in the form of data telemetry, and the up link has been in use since 1974 for remotely piloted research vehicles (Ref. 10). (This feature was also used during the F-8 DFBW program (Ref. 6).) A schematic of the PROFIT remote computation system is shown in Figure 9. Telemetered data from the onboard computer and also from sensors not necessarily available to the onboard computer are fed into the ground-based computer. Control algorithms, which may be written in Fortran or other higher level language, are executed, and the resulting commands are uplinked to the airplane. The PROFIT computer on the airplane receives the commands and some synchronization and validity codes, checks the commands for flight safety, and sends them to the appropriate pilot displays or actuators.

Remote computation is ideal for low sample rate energy management or profile optimization routines, which involve large programs and time-consuming calculations but only two or three command signals. It is also well suited to highly experimental control schemes, since a researcher would be able to modify the control laws easily without being able to adversely affect the normal control mode software. However, the control of such systems as afterburning turbofans, which have several moderately high bandwidth control loops, would be difficult, and would be better done on board.

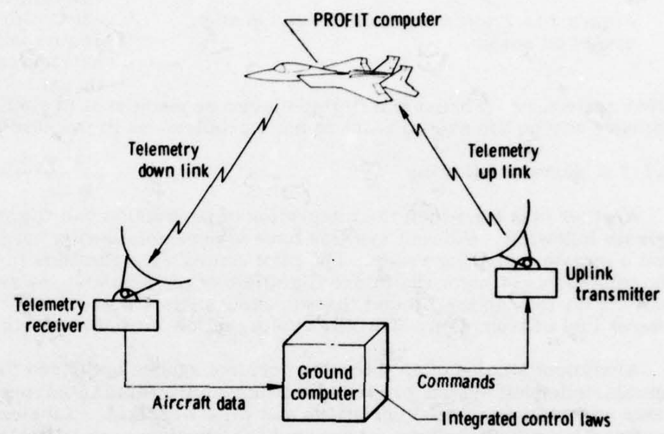

Figure 9. Schematic view of PROFIT remote computation.

3.4 Engine Control Research Facility

An important feature of the PROFIT concept is the capability to perform engine control research. The verification of an engine control mode is not complete until the engine has been subjected to rapid variations in temperature and pressure that cannot be simulated in a ground facility. Real flight transients such as compressor stalls, inlet dynamic distortion, unstart, and buzz are also impossible to simulate adequately on the ground. The PROFIT research facility will incorporate an engine with full-authority digital control of all of the control variables, with sensors at the critical engine locations.

4.0 CONTROL RESEARCH

The PROFIT program is designed to produce a research facility capable of evaluating a wide variety of control concepts. A few of the possible subjects of investigation are listed below. They fall into two categories: control integration research and engine control research.

4.1 Control Integration Research

Many control schemes are conceivable for the integration of propulsion and flight control. Within the propulsion system, there are integration schemes that affect propulsion system-airframe interactions.

4.1.1 *Engine-inlet-nozzle integration*

A relatively straightforward optimization of inlet drag, aft end drag, and engine performance for partial power is shown in Figure 10. For partial power operation, the engine manufacturer optimizes the engine control for minimum specific fuel consumption. The airframe manufacturer may also be able to optimize the engine for low inlet and aft end drag for the airflow the engine demands at one flight condition, usually high altitude cruise. For other flight conditions, which may be dictated by air traffic control requirements or involve external stores, non-standard-day ambient air temperature, or operational constraints, the match will not be optimum. An integrated control system with authority to control the inlet, engine, and nozzle can optimize the performance of the propulsion system throughout the flight envelope. Improvements of up to 5 percent in propulsion system thrust may be attained.

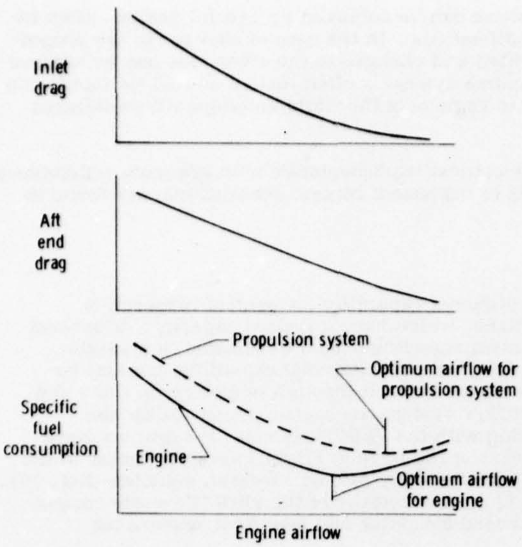

Figure 10. Engine-inlet-nozzle integration at partial power.

4.1.2 *Trajectory optimization*

Many integrated control schemes involve some form of trajectory optimization, that is, a control algorithm that controls flightpath and engine throttle setting to optimize some flight phase.

4.1.2.1 Minimum noise takeoff

One example of trajectory optimization is the minimum noise takeoff mode. This mode is designed to optimize the takeoff flightpath and throttle setting for minimum noise. Open loop programmed throttle noise reduction, an extension of presently used cutback procedures, has been proposed for supersonic transports (Ref. 4). In the minimum noise mode the onboard computer takes information about temperature, airspeed, ground speed, altitude, weight, runway length, and community type, and continually calculates the trajectory and throttle setting that minimize noise. The resulting outputs may be used to drive autopilot actuators or merely be displayed for the pilot.

The results are minimum noise, not just for a standard day with no wind and a certain gross weight, but for the existing conditions. Crew workload is reduced, improving safety during the critical takeoff and initial climb. If an engine fails during reduced thrust operation, the system automatically increases all other engines to maximum thrust and, if the crew so desires, flies an engine-out climb trajectory. Thrust and flightpath can be modulated in such a way as to minimize fuel consumption or improve engine life even if noise is not a problem, as in the case of an overwater climbout.

4.1.2.2 Terrain following

Another task for which the integration of propulsion and flight control systems appears to be advantageous is terrain following. Current systems have a terrain-following computer which is coupled to the flightpath controls and a terrain-following radar. The pilot controls the throttles independently. This causes some problems. Since the pilot does not know the future flightpath of the airplane, he must continually vary the throttle setting to maintain the desired speed. Rapid throttle excursions are necessary, some of which involve afterburning, with its poorer fuel consumption. Throttle cycling at low altitude and high speed greatly reduces engine life.

Analytical studies of an integrated control system optimized for terrain following have been conducted. The terrain-following system predicts throttle requirements in advance, allowing slower and smaller throttle excursions and reducing speed variations and pilot workload. Substantial reductions in throttle rate, with the accompanying fuel savings, have also been predicted, as indicated in Figure 11.

This integrated control concept could be demonstrated with an airplane equipped with a terrain-following radar, a computer that could handle the optimal control laws, and an autothrottle. It could also be demonstrated with the PROFIT airplane by using the arrangement shown in Figure 12. The aircraft could be flown over a simulated ground contour stored in a ground-based computer. Data from the NASA precision ground tracking radar would indicate airplane position with respect to the simulated ground contour. The terrain-following radar and optimal integrated control laws would process the aircraft position data and other telemetered data and calculate the desired flight control and throttle commands. These would be transmitted to the airplane for pilot display or use as actuator commands. Since the airplane could be flown well above the ground, the hazards of low altitude flight could be avoided. The ground-based computer programming could all be done in Fortran.

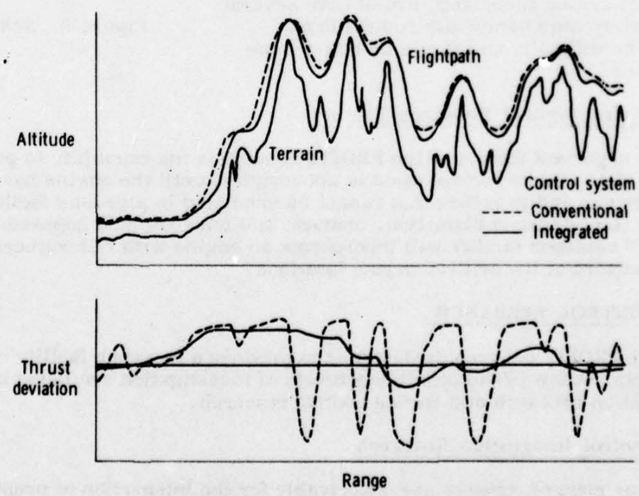

Figure 11. Comparison of altitude and thrust deviations for conventional control and integrated control optimized for terrain following.

4.1.2.3 Energy management

Whenever an airplane is supposed to proceed from one point in its flight envelope to another, there is an optimum path for the transition. This path varies according to the optimized parameter. For example, a pilot may desire to optimize fuel consumed, time consumed, or distance covered, depending on the mission. Until recently, an experienced pilot could do a good job of optimizing performance himself. However, in very high thrust-to-weight-ratio aircraft that can sustain high normal load factors at less than maximum thrust, a pilot's ability to estimate the optimum path is greatly reduced. Reference 2 discusses the flight and propulsion control aspects of energy management. Some of these concepts could be flight tested in the PROFIT program. The

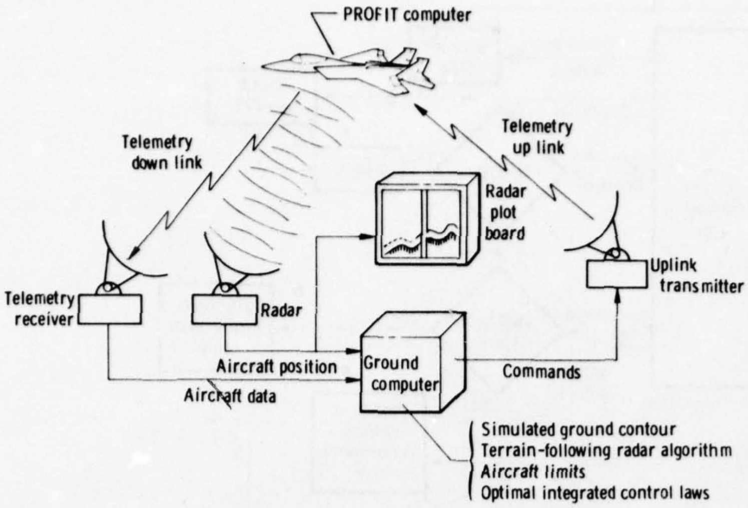

Figure 12. PROFIT concept as applied to optimal terrain-following problem.

results would determine whether the performance improvements justified the addition of a PROFIT system to an airplane. Methods of display and mode selection could also be studied. Pilots who are not familiar with the PROFIT airplane's capabilities would probably give a more accurate indication of the benefits of the system than pilots with experience in the PROFIT airplane, and should participate in the evaluation. For a production airplane, the long-term benefits of a PROFIT system would include not only improvements in performance but also improvements in pilot effectiveness and reductions in training requirements.

An example of the simulation results for an energy management application from Reference 2 is shown in Figure 13. An aircraft operating at an altitude of 3000 meters and a Mach number of 0.8 with 3180 kilograms of fuel is required to intercept another aircraft 50 kilometers ahead flying at at an altitude of 1500 meters and a Mach number of 0.9. Three different trajectories are shown. In the level acceleration case, the interceptor's fuel is expended after 205 seconds (point A) while the aircraft is still 20 kilometers behind the target aircraft. When the energy management algorithm is used to compute the minimum time trajectory, the interception is made in 240 seconds (point B) with 1360 kilograms of fuel remaining. The minimum fuel trajectory involves climbing to a higher altitude. In this case, the interception is made in 272 seconds (point C) with 1745 kilograms of fuel remaining.

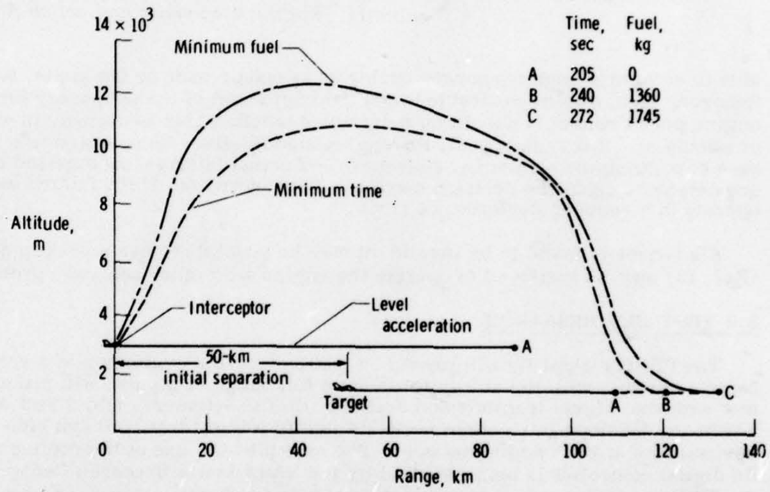

Figure 13. Energy management for intercept mission. Simulation data from Reference 2.

4.2 Engine Control Research

The engine of a high performance airplane is required to produce high thrust, to have low fuel consumption, and to respond rapidly to throttle changes. These requirements tend to necessitate a complex control system—one that controls several variables. The full-authority digital control for the TF30 engine in the F-111 IPCS program resulted in substantial gains in engine transient response and stability. Later technology engines incorporate more control variables than the TF30 engine. Some of these engines have limited-authority supervisory digital controls in addition to the hydromechanical controls, but no production engine has an all-digital control. Some of the performance gains to be expected with an all-digital engine control are increases in thrust, decreases in fuel consumption, improvements in stability and transient performance, reductions in maintenance requirements, and improvements in reliability. These improvements result from the control of more variables, additional sensors, more accurate controls, closed loop controls in place of open loop schedules, better matching of components, and software which will switch to other control modes in case of certain failures.

4.2.1 *Engine multivariable control*

A primary objective of the PROFIT program is to evaluate multivariable engine control (Ref. 11). This method of control, which was developed and evaluated on a hybrid simulation, will be ground tested on an engine in the NASA Lewis Research Center Propulsion System Laboratory. The engine control was developed by using linear quadratic regulator theory. It has been proposed that the PROFIT flight tests extend the multivariable control analysis to include the inlet as well as the engine.

4.2.2 *Engine problem detection and action*

A number of potentially useful engine functions are based on a common requirement of knowing the relationships between all of the engine states. These functions include failure detection, diagnostics, self trimming, and the use of self-correcting software. Each of these functions involves decisions about the overall health of the engine, and degraded or failed sensors, actuators, or engine components must be identified. Figure 14 shows a conceptual flow diagram of a system that performs these functions. Engine sensors are checked for validity first. If all sensors are determined to be valid, a diagnostics routine is entered. The routine must be

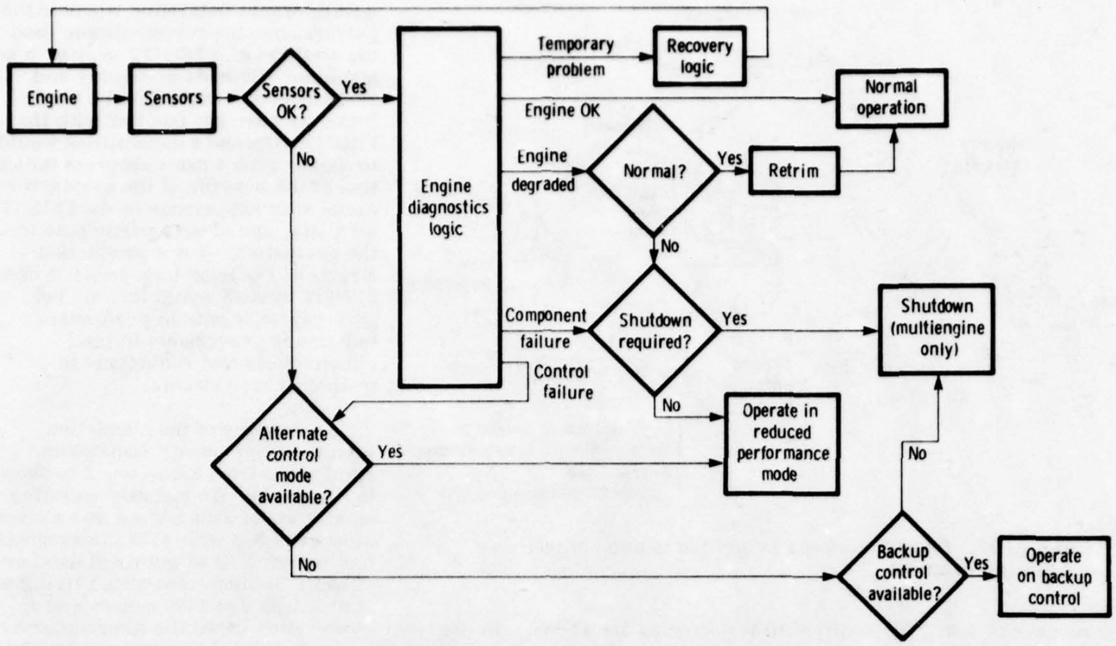

Figure 14. Engine diagnostic and action flow chart.

able to recognize such temporary problems as compressor or fan stalls, which do not represent actual failures. Recovery logic may be desirable to get the engine out of the temporary problem. If degradation is detected in engine performance, it should be determined whether this is normal, in which case the engine may be retrimmed, or abnormal. If it is abnormal, the engine should either be operated in a reduced performance mode, or, in the case of multiengine airplanes, shut down. If actual failures (as opposed to degradation in engine performance) are detected, again the decision may be to shut down, or, if the failure is in certain components, to continue to operate in a reduced performance mode.

If a sensor is found to be invalid, it may be possible to use a backup control mode. Self-correcting software (Ref. 12) may be employed to operate the engine with other sensors, probably at reduced performance.

5.0 TEST BED RESEARCH

The PROFIT airplane will provide a test bed for the evaluation of a variety of hardware concepts. The digital control system, with the ability to accept a full range of inputs, will make an ideal test bed for such things as new sensors, signal transmission systems, digital actuators, and direct digital transducers. Many of these new hardware developments cannot normally justify a flight test, but can ride along and provide data in the flight environment at little additional cost. For example, the use of fiberoptics as an interface between an engine and its digital controller is being studied by the NASA Lewis Research Center, and is planned for flight testing on the PROFIT airplane, as are some advanced direct digital sensors and actuators.

6.0 CONCLUDING REMARKS

The propulsion-flight control integration technology (PROFIT) concept is to be used to develop an airplane capable of studying a variety of integrated control concepts. The PROFIT system will incorporate engine, inlet, and flight controls that are integrated with a digital computer, and it will be implemented on a high performance supersonic twin-engine fighter airplane. The airplane control hardware will be modified to provide the necessary capability for control research, while software will be used to provide a flexible integration capability. The PROFIT system will be designed so as to not be critical to flight safety, reducing cost and risk. A remote computation capability using a telemetry down link, ground-based computer, and telemetry up link will be provided to supplement the airborne computer. One engine will have full-authority digital control of all of the engine variables, and will be available as an *engine control research facility*. Other subjects for control research are engine-inlet-nozzle integration, trajectory optimization, and multivariable engine control. An investigation of these and other subjects should aid in the development of control technology for future aircraft designs.

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PART III

TRANSPORT APPLICATIONS

ACTIVE CONTROLS FOR CIVIL TRANSPORTS

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SUMMARY

The principles involved in A.C.T. for civil transports are described and estimates are made of the probable benefits. The A.C.T. functions, manoeuvre load alleviation, gust load alleviation, relaxed stability, flutter suppression, ride quality improvement and fatigue improvement are discussed in turn and the problems and benefits outlined. It is concluded that load alleviation approaching 50% may be accepted as a target and that D.O.C. savings in toto of about 7% are possible and worthwhile.

THE TARGET

The motivation for research in the civil transport field is very largely to improve the aircraft economics as seen by the airlines. This must be done, to be acceptable, without any degradation of safety, reliability and maintenance costs. Civil transport technology is fortunate in that there is a commonly accepted measure by which improvements offered on the market can be assessed - namely D.O.C. (Direct Operating Cost). Unfortunately whilst there is a standard D.O.C. formula, it is much criticised in detail by almost every interested party each of whom has his own private variant. These variations turn largely on the way in which the costs of financing new aircraft are accounted for and the differences in D.O.C. between the formulae can be significant. However, in assessing the worth of various technological improvements and putting them in the right 'pecking-order', the various D.O.C. formulae do not differ significantly.

The salient result arising from an examination of a particular aircraft using one such D.O.C. formula is that in order to reduce D.O.C. by 1%, fuel usage has to be reduced by 3 to 4%. No other single item comes close to this, the next being reduction in airframe manufacturing cost which is around 6%.

Accordingly all those technical innovations which aim at reducing aircraft empty weight, aircraft drag, reduced engine bleeds and of course improvements in engine economy bear directly on reductions in fuel useage and are worth pursuing.

Of these innovations, no single technology offers significantly more than Active Control Technology in terms of D.O.C. benfits. A.C.T. in this context is essentially wing load alleviation together with relaxed stability. These benefits are around 4 - 7% on D.O.C. for a medium range (2000 n.m.) 200 seater aircraft when it is assessed at constant pay load and it is assumed that the technology is used to the full extent possible and the configuration is adjusted.

Analysis of a typical transport aircraft at constant Aspect Ratio leads to the conclusion that a target of 50% design wing BM reduction can be set using existing types of controls (ailerons, spoilers, flaps). This leads to some 15% reduction in wing weight, 3½% reduction in AUWE which directly leads to 2-3% reduction in D.O.C. or approximately double this if the aircraft is stretched with a given engine size for the same performance standard.

Analysis also leads to the conclusion that by relaxing the longitudinal stability to just above neutral and so allowing the C.G. to move further aft, the tail size and the tail load to trim can be reduced. When these are worked through, a D.O.C. gain of 2% results or approximately double this if the aircraft is stretched with a given engine size for the same performance standard.

If these two A.C.T. functions are implemented together the gains are not directly additive but something approaching their sum can be expected i.e. approaching 4 - 5% D.O.C. gain or 7 - 9% if the aircraft is stretched.

These gains are highly significant notwithstanding their small numerical description. A 1% D.O.C. gain on a DC9, BAC -111, Boeing 737 amounts to approximately a saving of £25K per aircraft per year on a typical 3000 hour utilisation or £0.5m over the 20yr life of any one aircraft.

These figures are fairly large and as targets they sustain the researchers in their quest for improvements and amply justify the research costs. In the A.C.T. field the costs of any one research group bringing these benefits to fruition is likely to be around £1½ - 2m. The Return on Investment estimated as a ratio of these figures is very satisfactory for any reasonable number of aircraft.

A.C.T. FUNCTIONS

Formally, in the A.C.T. field, the following functions are usually included.

1. Load alleviation.
   1.1. Manoeuvre load alleviation.
   1.2. Gust load alleviation.
2. Relaxed Stability.
3. Flutter suppression.
4. Ride-quality improvement.
5. Fatigue life improvement.

These will now be discussed in turn with a view to describing the possibilities and the pitfalls.

1.1. Manoeuvre Load alleviation

Transport aircraft are required to be designed to a given manoeuvring G. This value is part of the particular specification of the aircraft being designed, may be a function of weight and may be negotiable with the particular country's certifying Authority under whose rules certification is to be sought. This value of G is obtainable through the use of controls and is defended on the grounds of a desired turning radius and a desired dive pull-out ability. The arguments for accepting one value or another relate to measurements over the years from V-G recorders and the like and the larger values relate especially to inadvertent or emergency manoeuvring. It may well be that with a properly designed and engineered control limiting system (akin to a military manoeuvre demand system) inadvertent or extreme emergency manoeuvring excess G could be reduced without adversely affecting the normal turning or pull-out ability or other desired manoeuvres. Such a system is not discussed here but would in any case be a simple application of the systems to be discussed.

Accordingly it is assumed that the usual 1.5 excess G ability is desired giving as a limit load condition 1 + 1.5 = 2.5 G.

Manoeuvre load alleviation is not therefore to be construed as an alleviation of the total load but as an alleviation of the wing Bending Moments for the same total load by the expedient of shifting the centre of pressure of the additional load inboard.

The aim is to shift the C.P. inboard sufficiently to allow the design BM to be reduced by 50%. Since the gust BM is typically equivalent to some 3G and the manoeuvre BM to some 2.5G the gust case usually designs. In these circumstances the target BM is equivalent to 1.5G and the manoeuvre BM therefore needs to be reduced only by $1 - \frac{1.5}{2.5} = 40\%$ to match.

Considering as an example an aircraft having a wing weight of 10% A.U.W. and wing fuel weight 20% A.U.W., the nett manoeuvre BM will be approximately 70% of the air-load BM. Hence to achieve a reduction of 40% in the nett value requires 28% reduction in the air load BM.

The question now arises as to whether the air load can be manipulated so that this 28% reduction be achieved.

The means conveniently available are the existing controls namely aileron, flaps and spoilers. These conventional controls can in principle be rearranged and resized in any specific case and new controls e.g. leading edge devices can be called up.

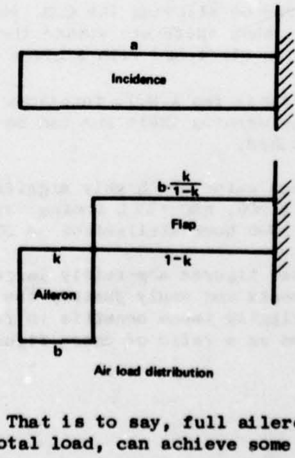
To introduce the possibilities at their simplest consider what can be achieved with a typical tip-aileron of 0.3 span, the rest of the wing having flaps or flaperons. Deploying up-aileron introduces relieving down loads at the tip and to preserve the total air load, to achieve the required 1.5 excess manoeuvring G, the flaps or flaperons will be deployed downwards. If it be arranged that these loads exactly balance then there will be no incidence adjustment but there will be changes in the balancing tail load. If it be arranged that they do not, then there will be an incidence adjustment. This latter situation is generally undesirable not least because greater control angle is clearly needed and this will be a limiting factor.

Simplifying the air load distributions involved, as in the figure, yields for the Root BM from the air load

Without A.C.T. :- $\frac{al^2}{2}$

From A.C.T. :- $\frac{bkl^2}{2}$

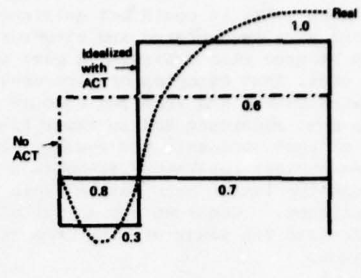
Nett :- $\frac{al^2}{2}\left(1 - \frac{b}{a}k\right)$



Air load distribution

Hence it is seen that $\frac{b}{a}k$ measures the alleviation. Now $\frac{b}{a} = \frac{a_2\beta}{a_1\alpha}$ where β is the control angle and α is the wing incidence at the 2.5G condition. For cruise at say 600 f/s for an aircraft with a wing loading around 100 lbs/sq. ft., $a_1\alpha$ at 2.5G will be about 0.6 which for an a_1 of 5 say, indicates an incidence of some 7°. The maximum aileron angle available in the up sense is typically about 20° and a_2 is around $\frac{1}{2}a_1$. Hence the maximum value of $a_2\beta$ is about 0.9 when $\frac{b}{a}k$ is then 0.45. That is to say, full aileron at cruise compensated by corresponding flap movement to preserve the same total load, can achieve some 45%

reduction in the air load BM at the root. The resulting distribution is illustrated in the figure. This alleviation varies with speed and Mach No. and at is about 20%.



Nett air load distribution

However this is a highly idealised value and will be much affected by the aileron aeroelastic effectiveness η. Since a tip aileron will normally reverse at about 1.2 V_D, at cruise its effectiveness will be about 0.4 and at V_A, the minimum speed at which 2.5G can be sustained, η will be about 0.85.

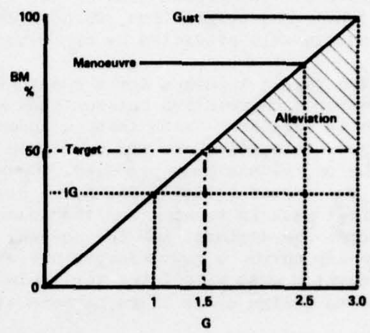
Hence in practise the reduction in air load root BM becomes 18% at cruise and about the same at V_A from full 20° of aileron.

These figures are to be compared with the figure of 28% required to achieve the 40% nett reduction in manoeuvre BM to match the 50% target reduction in design BM.

This simple example illustrates that a conventional tip aileron produces about the right amount of shift in C.P. of additional load if operated with flaps or flaperons. *Studies in depth have shown* that the target is severe but just achievable.

The thorny philosophical question of the safety factor enters here for it is clear that such a manoeuvre load alleviation system will cease to work linearly when the control reaches its stops and this will be equated to the limit load. If the ultimate load condition is interpreted as a rare but genuine load then the system will not give the extra protection between limit and ultimate load and the strength will be exceeded. If however the ultimate load condition is interpreted as a means of securing a reserve of strength beyond limit load strength and is not a genuine load then the alleviation system is not required to move beyond that needed for alleviating limit load i.e. not beyond full travel and will therefore be satisfactory.

The diagram emphasises the point that in the event of failure (*controls fail to move from neutral in response to* demand) the strength is only good for a 0.5G (unfactored) manoeuvre. Clearly this is restrictive but if the flight deck indications of failure were positive enough, it would be sufficient for an alerted pilot to proceed safely to his destination. Of course the 50% BM reduction argued *here is a* target only; a lesser target would involve lesser problems.



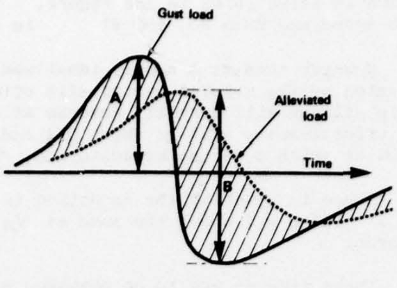
Different applications, but using the same principle are to allow the A.U.W. to increase without redesigning the wing or the span to be increased by the addition of larger tips without redesigning the wing. In both these cases a typical nett BM increase at the root would be about 10% which can be offset by the use of active controls. This represents a much more modest target than that of the previous section and is correspondingly easier to achieve.

To implement such a system involves an acceleration sensor, but a slow acting one, positioned on the aircraft centre line. The signal would be fed to the aileron, spoilers, flaperons which would also be slow acting. The pitching moments created as a by-product would need to be offset by the use of elevator. There are no particular difficulties in this systems design and some systems are about to be offered on the market. The major issue relates to the engineering of the flaps or flaperons to ensure that, although slow acting, they are nevertheless quick enough.

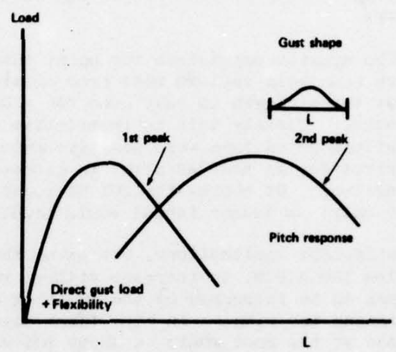
1.2. Gust Load Alleviation

At the outset there is a major question to face. Traditionally the gust requirements for transport aircraft have been based on the idealized concept of an isolated gust whose characteristics have been adjusted to give accord with measured data from instrumented aircraft. As aircraft characteristics changed, notably in the middle 50's from slow stiff vehicles to fast flexible vehicles, the data from the past was less relevant to the future and the traditional idealized gust concept did not necessarily account properly for the newer aircraft features, notwithstanding the rework of the measured data to provide adjusted values for the idealised gust. To account for these changes, the concept of continuous turbulence as described by power spectra was developed. The situation now exists wherein some nations' requirements call for the isolated gust description and others for the continuous turbulence description.

In considering gust load alleviation the task is considerably more exacting if the isolated gust concept is used as the gust description than if the continuous turbulence concept is used. Essentially this is because, in the continuous turbulence concept, response is described statistically in terms of averages whereas in the isolated gust concept attention is focussed on a specific response which is sensitive to the description of the isolated gust. Particularly, the response to an isolated gust must be sensitive to gust pattern and gust patterns are not well known or established. It is not difficult to imagine a gust pattern which at first causes a gust load alleviation system to act so as to alleviate the gust but which then changes sign too quickly for the system to follow thereby leaving the aircraft to contend with the alleviator acting temporarily in the wrong sense. Moreover it can easily be conjectured that if a system is designed to alleviate continuous turbulence only, it could act detrimentally if certain gust patterns were encountered and vice versa. Both descriptions should be used when designing a gust alleviation system for it seems clear that examples of more-or-less isolated gusts have been encountered and recorded (though not often reported) of design gust magnitude and to these has been attributed the cause of some accidents and equally clearly there are examples of more-or-less continuous turbulence of high intensity, although the latter have not yet been attributed the cause of any accident. Consider now a gust alleviation system designed to alleviate the loads arising from isolated gusts of design magnitude.

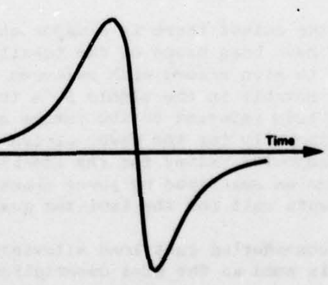


Such an alleviating system must either produce a load of opposite sign to the gust load at every instant i.e. attempt to cancel the load at source or must pitch the aircraft into the gust (or twist the wing into the gust!) so that the gust loads do not develop fully. In the former case, conventional leading or trailing edge controls can be used, and are, in effect, the only means available for short wave - length isolated gusts; in the latter case, elevator control can be used but will only be effective at the longer wave lengths since the time taken to pitch the aircraft is considerable. If an aileron control is used and is effective for short sharp gusts it will also be effective for the longer wave length or draught-type gusts, but a measure of elevator control will have to be added to compensate for the aileron pitching moments and to restore the handling qualities. For most transport aircraft the bending loads produced on the wing at the short sharp end are approximately equal to those produced at the long wave length end. If the current idealised (1-COS) gust shape of wave length L is used for illustration, the loads produced vary as in the illustration. The first peak is sensitive to the elastic characteristics, involves considerable "modal activity" but is largely a direct reflection of the instantaneous gust load modified by 'heave'. The second peak in the envelope is sensitive to pitching characteristics, CG position, pitch damping etc. and is well predicted by rigid-body analyses.



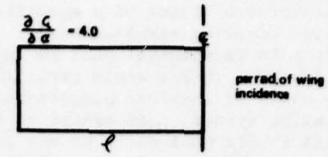
The figure is drawn for a constant gust velocity. If however a correlation between gust magnitude and wave length is accepted - such that on an equal probability basis high gust velocities are not associated with short wave lengths or, as has been proposed, $v_g \propto H^{1/3}$ with v_g = gust velocity and H the gradient distance, then the first peak is reduced and the second peak is increased to become the dominant one for design. (These ideas are not yet in any nation's requirements but are in debate). In this event a gust alleviator for isolated gusts would be easier to design since there is more time in which it can react.

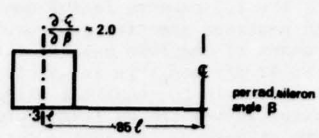
Illustration has been made of the idealized (1-COS) isolated gust but it may be expected that, for an aircraft with a gust alleviator to be certified in accordance with isolated gust requirements, other gust types would need to be considered. It would be ideal, if a gust alleviation system were able to alleviate gust patterns of any complexity. This may be too much to expect and it will then be necessary to shew that the system alleviates gust patterns associated with magnitudes above a certain probability level. Little work is available on such probabilities; however a "Vortex" gust represents a practical extreme, for it contains the rapid reversal feature, as in the figure, is known to exist in the atmosphere, and has an understandable origin in atmospheric vorticity as from convective storms. Its actual shape could take cognizance of any proposed correlation between magnitude and "wave" length. It should however be noted that the maximum rate with which the gust load can change is limited by the mechanism of circulation change on an aerofoil conveniently described by the Küsner function well known in two dimensions at least.



Vortex gust shape

There is a fundamental question as to whether a control can produce enough force fast enough to reduce sufficiently the loads arising from sharp gusts. An estimate of the control angle required is easily made, using the (1-COS) gust as an example. Consider a (1-COS) gust of magnitude v_g and wave length L. The wing incidence is then $\alpha = v_g/V$ radians. For a typical system with a rigid wing having a tip aileron occupying 0.3 of the span, the forces are proportional to $\frac{\partial C}{\partial \alpha}$ and $\frac{\partial C}{\partial \beta}$ as illustrated. For equal BM at the root i.e. full cancellation of the gust by the control in a static sense $2 \times 0{\cdot}3L \times 0{\cdot}85L \times \beta = 4 \times L \times 0{\cdot}5L \times \alpha$ whence $\beta/\alpha \doteq 4$

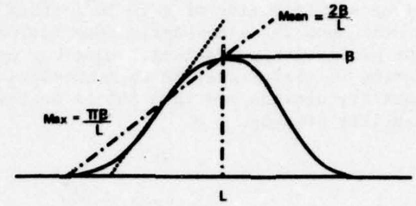




For a flexible wing the aileron will have effectiveness η and then for incremental gust BM cancellation $\beta/\alpha \doteq 4/\eta$

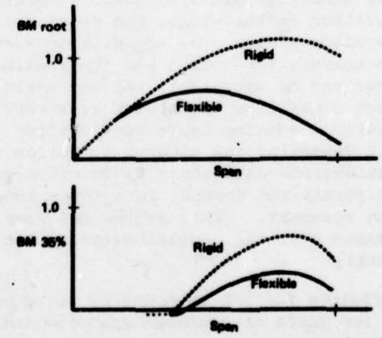
Hence for a gust, producing incidence $\alpha = v_g/V$ radians the aileron required is $\beta = 4/\eta \cdot v_g/V$ radians. When v_g = 60 f/s, V = 600 f/s and η = 0.4 say, as at cruise, $\beta \doteq 1$ radian. Clearly no aileron control can be worked to 1 radian; ⅓ radian is a typical maximum. Hence a typical tip aileron of 30% span working at 40% aerolastic effectiveness will be able to produce enough BM at the root to alleviate the BM from a 60 f/s isolated gust by ⅓.

The rate of application can be assessed by observing that since β follows (in this elementary illustration) the (1-COS) shape, as illustrated, the average rate is $\frac{2}{L}\cdot\beta$ and the maximum rate is $\frac{\pi}{L}\cdot\beta$ degrees/foot or $V\cdot\pi\beta/L$ degrees/second. If β is limited to 20° (⅓ radian), then for L = 60 feet and V = 600 f/s representing a short sharp gust the maximum aileron rate required is $\pi/60 \times 20 \times 600$ $\doteq 600°$ /second and the average is $2/60 \times 20 \times 600$ $\doteq 400°$ /second. These estimates incicate only that a high rate of application will be necessary to cancel the gust instant by instant. Studies of a much more detailed nature indicate that about 200°/sec suffices and this is within the compass of modern practice on hydraulic power units.



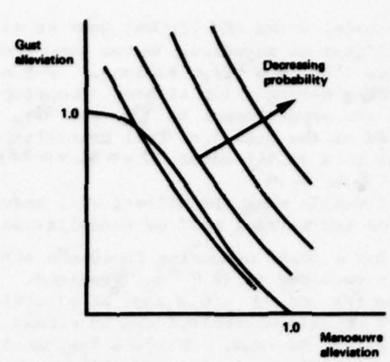
(1 — cos) gust shape

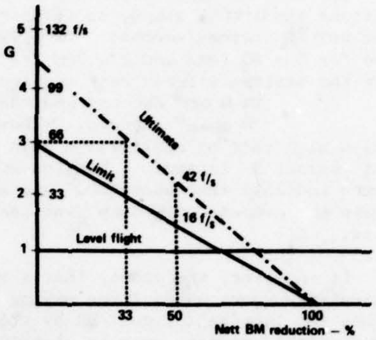
It is clear, therefore, that a typical aileron of a typical transport has a force making capability sufficient to reduce the incremental gust BM by about ⅓ provided it can be moved quickly enough; equally clearly if, say, twice the trailing edge can be given over to aileron (or rotating trailing edge of a flap) and/or if other controls e.g. spoilers can be brought into play, then a reduction in the incremental gust BM of about twice the figure or more is achievable. On a particular study a reduction of the gust incremental BM by 75% to 25% was an achievable target. It may be noted, as in the figure, that although the moment arm reduces as the aileron moves inboard its aeroelastic effectiveness increases and the most effective aileron position for relieving the ROOT BM is about mid span and for relieving BM at 35% span is about 75%. It must also be noted with regard to spoilers that their lift-making capability in both +ve and -ve senses in terms of rapid operation is not well established and spoilers may not therefore be a useful control for this purpose.



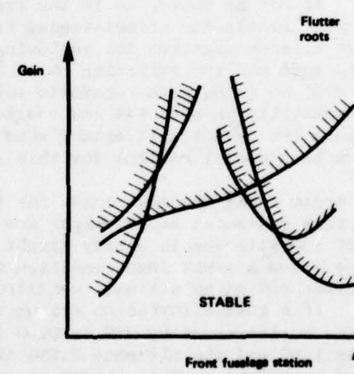
In broad terms the gust loads for the design 66ft/sec gust on a typical design at heavy weight are equivalent to approximately 2G and with the 1G steady flight condition the design up-gust case gives a limit load condition of approximately 3G. This becomes 4½G as an ultimate condition with a 1.5 reserve factor. If a gust alleviation system is deployed so as to reduce the 2G increment by 75% to 0.5G then the nett limit condition is 1.5G and the ultimate 2.25G and the nett BM in the gust case is halved. This target is consistent with the target possibilities in the manoeuvre load case and both systems must be deployed for the design BM to be so reduced.

Clearly there would be insufficient control movement to alleviate the design gust BM and the design manoeuvre at the same time. There is therefore an "interaction" curve to be developed as illustrated in the figure which would need to be assessed against the combined probabilities of a specific gust and a specific manoeuvre occuring simultaneously. The target figure of 75% reduction in incremental gust BM implies that a gust of 25% of 66 f/s i.e. 16 f/s would give limit load on the structure in the event of complete passive failure of the gust alleviation system. At cruise conditions of 30,000 ft and M = 0.84 a 16ft gust would be met about once in 10^4 nauticel miles i.e. perhaps once in every five flights. However if the whole of the 1.5 reserve factor can be attributed to structural strength reserves then the gust which would give ultimate load in the event of complete passive failure of the gust alleviator system is 42 f/s and this is likely to be encountered in cruise conditions one in 10^8 nautical miles. Failure of the gust alleviation system can be likened to failure of a member of a fail-safe structure. In such cases civil aircraft certifying authorities accept a residual strength sufficient for limit load. If this same philosophy be used then to achieve a residual strength sufficient for a 66 f/s gust treated as an ultimate condition would require, as seen in the figure, that a BM reduction of 33% only be made by the use of a gust alleviation system. If anything better than 33% is to be attempted, detailed consideration would have to be given to the reliability of the A.C.T. system to ensure that the combined risk of encountering the design gust with, at the same time, the system inoperative was acceptable. Some studies have indicated that for a 50% BM reduction and to meet a 1 in 10^9 risk of catastrophe a system requires a reliability of 1 in 10^4. This is a figure very dependent upon the atmospheric description used but in itself is not particularly exacting. One may conclude that a 33% reduction in design gust BM is achievable with no great system reliability demands and that 50% is achievable with increased reliability demands.

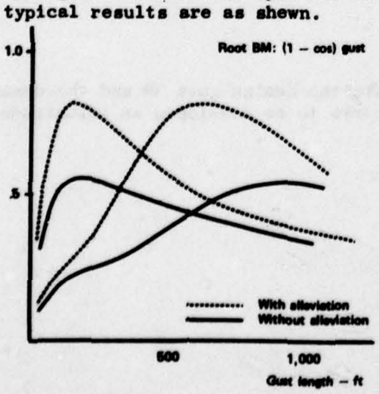


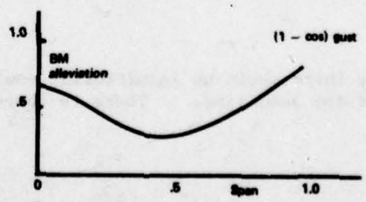


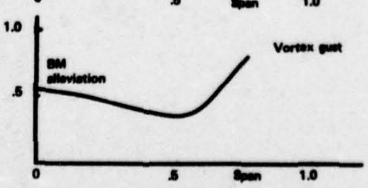
It remains only to consider what difficulties exist in designing control laws to drive the controls. One obvious philosophy to follow is to install a sensor where there is large wing deflection – at the tip say – use the signal as an error signal, and attempt by the use of aileron to 'null' it. This has not been found difficult when using simple dynamic models, two rigid modes plus two bending modes, and an accelerometer with no filtering is satisfactory in principle. But with more representative (more elaborate) models such a system is very prone to introduce flutter type instabilities. These will be sensitive to sensor position and system gain. It is likely to happen that the gains necessary to ensure adequate flutter stability are inadequate to achieve the alleviation for any position on the wing. For positions on the fuselage, an acceleration sensor does not pick up much wing motion but picks up the heave/pitch motion and the instability problem is much lessened and an acceptable region materialised as illustrated. For such a position the system acts more like a gust force sensor – by virtue of sensing heave acceleration – producing an open loop signal demanding the aileron to follow the gust "precisely". Another obvious philosophy is to use a gust vane to sense the gust directly and demand, in an open loop manner, a cancelling aileron movement. This system has been found to offer no advantages over the accelerometer system and is less robust physically.



Studies for one such design using aileron alone driven by an accelerometer have been conducted in depth for gusts of constant spanwise intensity and many shapes, (1-COS) and Vortex notable among them. Some typical results are as shewn.





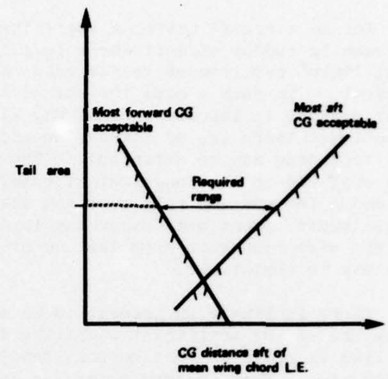


The system, designed on an isolated gust basis has been checked on a power spectral density basis with the specification as in current F.A.A. regulations. It transpires that the system behaves very well and the degree of alleviation is very similar in all respects. In particular it transpires that, in order to achieve the maximum alleviation afforded by a conventional aileron operated to full travel i.e. some 25% on Root BM, a high rate jack of much the same specification as for the isolated gust case is required. However the reverse situation has not been studied and it may be that a system designed on a power spectral density specification of turbulence would not give a satisfactory alleviation on specific isolated gusts.

2. RELAXED STABILITY

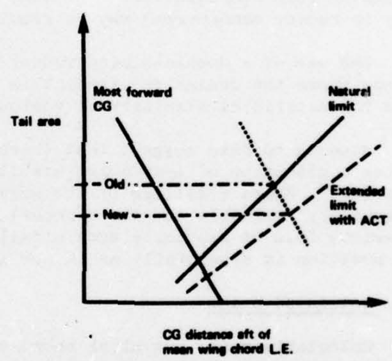
2.1. Longitudinal Stability

For most subsonic transport aircraft the range of centre of gravity travel which must be accommodated is primarily a function of the passenger layout. This c.g. range can be moved relative to the wing, but the extent to which this is possible depends on the ability to hold the nose up at the most extreme forward c.g. position, and on the amount of natural (longitudinal) stability required at the most extreme aft limit. The larger the tailplane size the more easily the nose is raised and the greater the natural stability. This produces a relationship illustrated in the figure.



If the longitudinal control surface (elevator or tailplane) can be operated in such a way as to provide stabilising pitching moments proportional to angular displacements and/or velocities in pitch then the effective stability is increased. This can be done by for example sensing angular velocity in pitch and by sending the signal to the elevator operating jacks. Since the effective stability has been increased a further aft c.g. position can be tolerated, so relieving the trim moments at the forward c.g. for a given c.g. range.

As shewn the c.g. range is thus moved aft overall, so reducing the tailplane size required. This gives a direct reduction in structure weight and a fuel weight reduction consequent upon the drag reduction. At the same time the downward tailplane load to trim with flaps down is reduced, so that at take-off with forward c.g. the 'trim loss' is reduced, giving higher $C_{L\ max}$ and lower trim drag. Consequently take-off performance is improved. Trim drag on cruise may also be improved, though this effect is commonly smaller, and if the c.g. range is moved very far aft the trim drag will eventually increase again.



The weight and drag reductions can be used to provide either improved performance (for a given aircraft) or more payload (for a given engine size) or a reduced aircraft weight (for a given payload requirement). With the second and third of these options a reduction in direct operating cost is obtained.

Wherever natural stability is diminished and artificial stability sbustituted there is likely to be a problem of maintaining reliability for the stability augmenting devices. Total failure of the device will obviously cause some degree of degradation in stability. If the loss is not too severe a minimum acceptable level of stability allowing the pilot to maintain control and complete the flight may be maintained. In such a case the reliability of the stability augmenting device must be such as to reduce the chance of failure to a low level, but the possibility of in-service failure is recognised and accepted. Where the loss of stability is very severe, however, the effect of a failure is likely to be catastrophic. In such cases the level of reliability demanded is such that a total failure is unlikely to occur during the life of the aircraft. With system multiplication any single failure may be contained, but there remains the problem of estimating overall failure probability due to combinations of local system failures. It may be argued that a single catastrophic system failure (e.g. lightning strike causing total electrical system failure) must still be allowed for, and on this basis multiplication must include variation in system type (e.g. manual reversion of some kind, or use of parallel fluidic and electronic systems).

One problem that tends to occur with conventional aircraft arrangements arises from the use of wing-mounted undercarriage. With such an undercarriage the attachment points are normally on or near the rear wing spar (or the rear of the wing structural box) and space available for stowage in the wing root and lower fuselage is usually such that little freedom exists to move the undercarriage back or forward relative to the wing. When, therefore, artificial stability is used to move the c.g. range aft relative to the wing there may be some difficulty in meeting the 'minimum nose wheel load' requirement which arises from steering considerations - the most aft c.g. is too near to the main undercarriage which cannot be moved aft relative to the wing. In such cases the limit on stability augmentation may occur when the minimum acceptable nose wheel load is reached. Use of a fuselage mounted undercarriage eliminates this problem, but the weight (and possibly drag) penalty usually associated with this type of undercarriage offsets some at least of the gain from stability augmentation. Clearly where a fuselage mounted undercarriage is used for other reasons there may be some advantage to be gained.

Other design cases (flaps up, aft c.g., trim limit; flaps down max. speed control angle limit; etc.) may intrude where undercarriage minimum loads are not limiting, though improvements in maximum lift capability of the tailplane and increase in travel may assist here. Ultimately reductions in tailplane size (and therefore elevator size) will be set by loss of control power - artificial stability is obtained by moving the aerodynamic control, and in general the moment of inertia in pitch will not decrease in proportion to any tailplane size reduction. Studies done to date do not suggest any serious difficulties in providing appropriate feedback signals.

2.2. Lateral Stability

Just as stability augmentation in pitch produces in the first instance a reduction in tailplane size, so stability augmentation in the lateral plane gives firstly a fin size reduction. The position is, however, more complicated in the lateral case because the requirements to be met are more various, and because the critical trim cases commonly involve the ability to 'hold' an engine failure - for a given engine size and aircraft layout the yawing moment generated by an engine failure cannot be alleviated significantly by moving the c.g. position as in the longitudinal case.

For an aircraft having a small thrust moment arm about the c.g. (e.g. a rear-engined aircraft) an increase in rudder effectiveness (e.g. use of a double-hinged rudder) may reduce the fin area for the worst 'trim' requirement to the point at which the lateral stability and control requirements become critical. In such a case the use of lateral stability augmentation (yaw damping, artificial n_v) and other devices to improve control may allow a reduction in fin area, with corresponding weight and drag reductions (there is, of course, no effect on $C_{L\ max}$ or trim drag in normal flight, though the 'engine out' trim drag may be affected). The stability augmentation is obtained in a somewhat similar way to that employed in the longitudinal case, though the angular displacements and velocities concerned are now about the yaw and roll axes and the stabilising moments correspondingly. For an aircraft having a large thrust moment arm (underwing layout) the engine-out trim requirement may be so severe as to prevent any fin size reduction from the use of stability augmentation, even though increased rudder effectiveness may be achievable.

There is likely in general to be a problem in reducing engine-out yawing moments sufficiently to allow the use of artificial stability for reducing fin area. Use of a yaw damper is, however, commonly required in order to meet normal stability requirements; in this sense artificial stability is in common use. Where rudder power can be increased (double-hinged rudder, for example) or the thrust offset from the aircraft centre-line reduced the need for stability augmentation beyond the conventional yaw damper may appear. In such cases the yaw damper becomes more powerful and may be accompanied by automatic turn compensation. At some stage automatic thrust compensation (to assist in limiting sideslip in rudder manoeuvres) may be required, and finally 'artificial n_v' must be supplied.

The use of a double-hinged rudder may increase drag in the asymmetric trim (engine out) case at low speeds where the design requirement is such that a low minimum control speed is needed there may be a need for artificial stability of various kinds merely to maintain acceptable stability.

Studies to date suggest that there are some specifications and aircraft layouts whose use tends to favour application of artificial stability, but that there may be limits to the level of augmentation possible. Where a failure of the augmentation system would not be catastrophic the system must be 'adequately' reliable and the aircraft controllable after the failure. Where system failure leads to a serious loss of stability such a failure must be inherently improbable within the life of the aircraft - the position is essentially as in the longitudinal case.

2.3. Estimated Gains

Calculations for a typical short/medium range transport aircraft suggest that, within the restrictions noted above, use of longitudinal stability augmentation might produce drag and weight reductions such that direct operating cost was reduced by 1 - 2% where system failure would be non-catastrophic. With system multiplication etc. to meet the potentially catastrophic failure case the gains may be increased to give 2 - 3% reduction.

For the lateral case there could be no gain unless the engine-out yawing moment could be contained with a reduction in fin size. Assuming this to be the case, a gain of 1 - 2½% D.O.C. might ensue. These gains assume a constant payload so that the aircraft becomes smaller as a result of using the ACT technology.

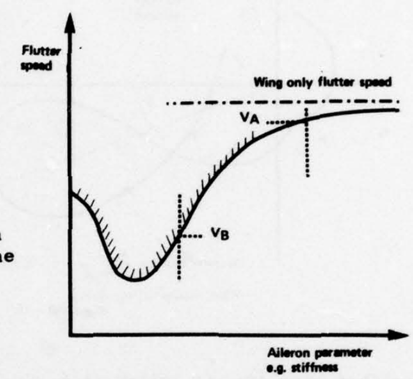
In the longitudinal case typically a reduction in tail volume ratio of about 22% can be expected if longitudinal stability is relaxed to the level of neutral stability in the ACT failed inoperative case and this is equivalent to about 25% in tail plane area after allowing for the reduction in aircraft size. The gain in $C_{L\ max}$ which can be expected to result from reduced trim (less down tail load) is about 3% which equates to a 3% reduction in thrust/weight ratio at take-off.

In the lateral case in the absence of the engine-out constraint a 50% fin area reduction can be expected if the stability is relaxed to the neutral level and this would put great demands on the rudder - calling perhaps for a double hinged rudder. The usual engine out constraint is such however that very little of the potential gain can be used in any practical design.

3. FLUTTER SUPPRESSION

For the typical civil transport aircraft the material provided to achieve adequate strength normally also provides adequate flutter margins. These margins are set, in most regulations, as 20% above the diving speed. There are exceptions and for example extra weight may have to be provided on fin torsional material on T-tail designs and for another example extra material may have to be provided to increase the rotational stiffness of control surfaces. In most cases however the weight penalties provided to achieve satisfactory flutter margins are small. It follows that there is little to be gained on weight savings by introducing a flutter suppression system and correspondingly little on Direct Operating Cost.

However in cases where for one reason or another flutter margins are inadequate, a flutter suppression system can be very valuable in restoring the margins. A particularly sensitive example is a wing with underslung engines, for the flutter speed of this system is sensitive to small changes in engine location and it is not easy to re-engineer a new location if the flutter margins turn out to be inadequate. But, if the natural flutter speed was low, below V_c say, an especially high reliability against failure would need to be provided and demonstrated convincingly, for in such a circumstance failure of the system at cruise or beyond would imply disaster. For this reason a flutter suppression system for a civil transport is unlikely to be used unless the natural flutter speed is above V_c by a satisfactory margin. In any case there will usually be alternative methods which can be employed e.g. deploying fixed weights, changing the geometry, changing the stiffness, and the relative merits of the various methods including use of an A.C.T. system will need to be compared. In many cases an existing trailing edge control will prove an adequate "motivator", if suitably used, to provide the extra margins. Its suitability can always be tested by exploring the sensitivity of the flutter speed to variations in control surface parameters. For example typically a wing flutter speed will normally vary with aileron rotational stiffness as shown in the figure. If the mode shape of flutter at points A and B are compared, the difference in aileron motion - amplitude and phase - will be observable. It follows in a very loose sense, that if the opposite aileron motion to that which 'caused' the drop in flutter speed from V_A to V_B could be achieved, that a gain in flutter speed beyond V_A would result.



Before considering how to establish the control laws in any case it is instructive to examine briefly the weight savings which follow from purposely designing for a low wing flutter speed. The wing flutter speed is dominated by the torsional stiffness which is provided largely by the skin thickness. Typically 50% of the wing weight is the 'torsion box' and of this, 70% is the cover weight. Skin to stringer areas vary quite considerably but an average figure of 50% of the cover weight being skin and 50% stringer is reasonable hence the skin weight which provides the torsional stiffness is .5 x .7 x .5 = 17.5% of the wing weight or about 2% of the aircraft take-off weight. If a natural flutter speed of 0.6 V_D for example was chosen, as against 1.2 V_D for a normal design, then the skin weight could reduce to $(\cdot 6/1.2)^2 = \frac{1}{4}$ of the normal value. To a first order therefore 75% of the skin weight could be saved - bending strength considerations apart - implying a 13% reduction in wing weight. This reduction would imply about 2% D.O.C. benefit but would be eroded by the problems of repairing the bending strength and providing the necessary systems reliability.

A reduction in skin thickness by 75% when the original skin area and stringer areas were equal implies a reduction by some 60% of the bending strength. This is the same order as the target value already discussed and could be achieved using a load alleviation system using the same motivator. But however the torsional stiffness being so low i.e. $\frac{1}{4}$ of the normal value, it would approximately halve the reversal speed and at 0.6 V_D the aileron effectiveness would be zero and the aileron would then be totally ineffective. Clearly therefore either a leading edge control would have to be used - whose effectiveness increases with reduced torsional stiffness but whose basic force-making characteristics are about $1/10$ those of a trailing edge control of the same size - or more subtle means for increasing the effectiveness - moving the effective flexural axis for example - would have to be used or a lesser target accepted.

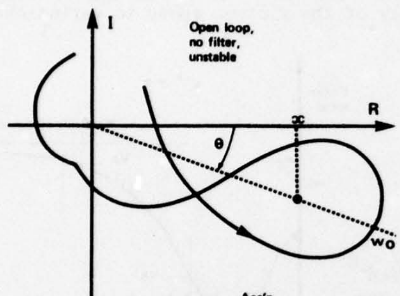
Reverting to the consideration of a case where there is a need to introduce A.C.T. for repairing flutter margins by the use of a flutter suppression system and where a trailing edge control surface (or any other) is deemed appropriate as the controlled motivator, the problem is to establish a control system and control laws. The essential function of any such system will be to increase the damping in the flutter root to a satisfactory level over the speed range without unduly impairing the other roots or introducing other instabilities. This may not always be possible and generalisation is difficult. The simple and easy situation will be when the flutter mode is at a frequency sufficiently different from any other (non-flutter) mode of similar character and when the control forces are able to do work in the mode i.e. the control is situated at a position of significant movement of the one mode only. The difficult, if not impossible, situation will be when the fluttering mode is at a frequency close to another similar (non fluttering) mode and the control is situated at a position of significant movement of both.

In either case the same techniques of establishing control laws can be tried. Two extremes can be instanced. The first uses modern control theory e.g. the pole-placement technique in which the desired roots are specified for a given flight condition in advance: for example all stable roots to be the same numerically but the unstable roots to have the sign of their real parts changed. These techniques require an arbitrarily chosen assignation of enough sensors and lead to a formal mathematical solution for the K in the form:-

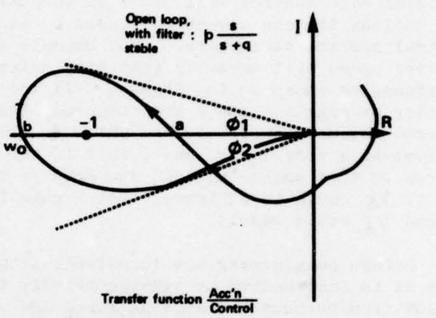
$$\eta = K_1 q + K_2 \dot{q}$$

where η is the control angle
q are the generalised co-ordinates
k are coefficients.

All the K will exist in general as a result of such a process. For practical implementation of the K there will need to be as many sensors as there are q's. Several choices will need to be explored and it will often be found by computational experiment in any case that many of the coefficients in K_1, and K_2 can be set to zero without much loss of effectiveness, but that there will still be several remaining. The number of sensors can be further reduced by the use of one or other of the techniques involving "observers", which substitute some sampled measurements over a time span for some sensors, for values from others, but the technique usually yields a need for several sensors. The practical implementation of a system needing several sensors requires high reliability from each sensor and is often impracticable especially bearing in mind the need for the system to be cost effective. The second technique assumes that only one sensor, and of a given type, will be deployed, assigns it an arbitrary location (one of many trial locations) and establishes the transfer function between motivator force and sensor output. Examination of the Nyquist plot for all the trial locations establishes whether there is a location for which the gain and phase margins are likely to be adequate. A (complex) gain can then be assigned easily interpreted in terms of filter characteristics and check calculations run including, as is always prudent, sensitivity calculations. A typical pair of results is as in the following figures.



Transfer function $\frac{\text{Acc'n}}{\text{Control}}$
Filter required at w_0, Gain $\frac{1}{x}$
Phase $180 - \theta$



Transfer function $\frac{\text{Acc'n}}{\text{Control}}$
$\phi_2 \geq$ Phase margin $\geq \phi_1$
$a \geq$ Gain margin $\geq \frac{1}{b}$

In neither case does it follow that stability will be achieved over the whole speed range and this must be the subject of a check. Such techniques clearly depend upon the accurate modelling of the aircraft dynamics. Current accuracies are sufficient provided there is adequate margin on the filter characteristics. In practice experimental data is highly desirable to monitor the modelling and flight tests will be used to provide a final adjustment to the system laws.

At its simplest the equations for a given flight condition have the form

$$Aq + B\eta = 0$$

q = generalised co-ordinate
η = control co-ordinate

which, with control law $\eta = Kq$ included, this becomes

$$\bar{A}q = 0 \qquad \bar{A} = A + \Delta A \qquad \Delta A = BK$$

It is always instructive to examine $\bar{A}$ in relation to A when the mechanism by which the flutter speed is improved can usually be detected. This may be by reducing the important coupling term, or by changing a frequency ratio or even by introducing direct damping into a coordinate. But, it should be noted, this mechanism results from the analyses and is not pre ordained.

4. RIDE QUALITY IMPROVEMENT

Although there are considerable insurance claims per year in the U.S. relating to injuries sustained on normal passenger flights, most of them due to suddenly encountered turbulence by passengers away from or not strapped into their seats, the ride quality offered by the normal passenger transport aircraft is very good. The case for doing anything special therefore relates only to improving some feature of a particular type which leads to a sub-standard ride. This is in contrast to those military aircraft which are required to operate at very high speed low down and whose crews have a difficult job to do the while, the success of which may depend upon a ride quality improvement system.

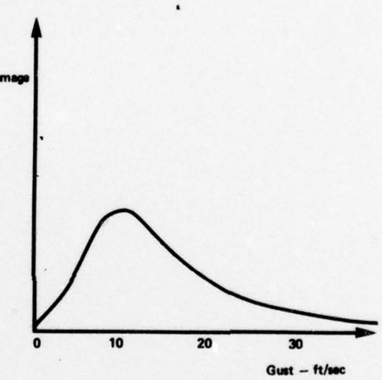
The most likely shortfall in ride quality of a civil transport is in lateral motion involving uncomfortable yawing oscillation at Dutch-roll frequencies around ½ c.p.s. The human seated body is more sensitive to lateral motions than to vertical motions and there are occasional difficulties with lateral fuselage bending modes around the 2 c.p.s. frequency. A feature of some modern designs with rear-fuselage-mounted engines is that the part of the fuselage projecting in front of the wing is unusually large and, as such an aircraft is developed, usually gets larger. The crew and front passengers may then be subjected to unwelcome lateral (and vertical) motions. There is a question as to how long such a fuselage can be made before it becomes unacceptable from a ride quality point of view. Studies (unpublished) seem to indicate that although the amplitude of the motion at the crew station, for example, increases with increasing front fuselage length, the frequency reduces and the human tolerance to the motion keeps pace with the motion. No limit seems to exist therefore on these grounds. Had there been a limit, use of active controls, would have been a way of overcoming it.

In any of these possible cases, however, a system designed for gust force, reduction may be expected to reduce the motion everywhere and the same techniques may be used. In the lateral case use of rudder is of course substituted for use of elevator to improve the stability and the system is essentially an increased authority yaw damper. Reducing the front fuselage lateral motion by using a rudder control actively at the rear of the rear fuselage may involve difficulties since there will usually exist modes at various frequencies close to another having broadly similar mode shape. There will then be difficulty in ensuring that instabilities are not introduced or that in attenuating the mode of original interest the system does not augment another mode close by and nullify the nett effect. In such cases, too, since this motivator and motion to be controlled are separated by considerable structural length, imperfection in the modelling will be more important. It was for these reasons that the I.L.A.F. system of the early years was suggested in which the sensor and the motivator were arranged to be at the same place. As in the gust alleviation systems the accuracy of the modelling is unlikely to be better than 10-15% on any quantity and consequently it can be expected that flight measurements will be necessary to tune the parametric values available.

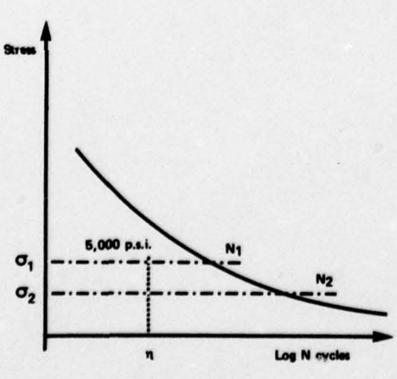
As discussed under the heading of fatigue reduction, a system which reduces the wing stress by, say, 25% will also reduce the motion at any crew or passenger station by approximately the same amount. Of course, as a result of reducing the scantling sizes of a wing due to the deployment of an A.C.T. system, the vibration frequencies will tend to become lower and move into that part of the turbulence spectrum having higher energy but the same A.C.T. mechanism which allows this, will more than compensate for any reduction in ride quality on this account.

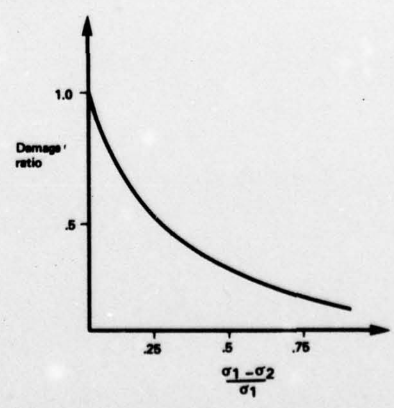
5. WING FATIGUE LIFE IMPROVEMENT

In essence the procedures and systems adopted for gust load alleviation are the same as those needed for wing fatigue life improvement. But for the typical civil transport designed to the usual regulations and operated in the normal way the bulk of the fatigue damage arising from atmospheric turbulence emanates from the 'moderate' gusts of about 10 f/sec in magnitude. This is because the larger gusts which invoke more stress are progressively rarer whilst the smaller gusts, although more frequent, produce progressively less stress. Accordingly any system aiming to improve wing fatigue life, only needs to operate on the moderate gusts as shewn. This is in contrast to the design gust load system which must operate on the maximum gust. Thus a design gust load system will inevitably improve fatigue life but a fatigue life improving system will not necessarily alleviate (isolated gust) design loads. The dominant quantities in assessing fatigue of structures made from the aluminium alloys are firstly the magnitude of the alternating stress and secondly the mean stress about which the alternating stress takes place. Typically, for a civil transport using conventional materials, the 1G stress is the mean stress and is about 16,000 p.s.i., and the alternating stress for the 10 ft/sec gust is about 5,000 p.s.i. If in unit time there are n cycles at stress σ_1 which would give failure at N_1 cycles, then the damage is n/N_1.



Correspondingly if the alternating stress is reduced to σ_2 then the damage is n/N_2. The ratio of damage due to the change from σ_1 to σ_2 is $n/N_1 \div n/N_2 = N_2/N_1$. From a typical S-N curve for 16.000 p.s.i. mean stress level and with σ = 5000 p.s.i. the damage ratio at 25% stress alleviation i.e. σ_2/σ_1 = 0.75 is 0.4 and for σ_2/σ_1 = 0.5 is 0.08. The attenuation in stress from σ_1 to σ_2 can be brought about by an A.C.T. system and will reduce the damage from gusts. Typically the total damage to a transport aircraft operated normally will be about ½ from the ground-to-air cycle and ½ from gusts. If a 25% stress reduction is assumed then the damage from gusts is only 40% of the full-stress level and the life is improved by 1 ÷ (.5 + .1 x .5) = 80%. The fatigue life improvement clearly depends on the proportion of the damage arising from gusts to ground-to-air cycles which may vary between 30 : 70 and 70 : 30 depending on aircraft weight, flight plan and many other conditions.





Since it is the bottom wing surface, the tension surface, which is critical for fatigue and since this represents some 20% of the wing weight, a 25% stress reduction as from the deployment of an A.C.T system would allow a 5% wing weight reduction for the same life. This is a .05% reduction in A.U.W. typically and is not attractive enough as a target. However, if a design has experienced fatigue difficulties the deployment of, say, a 25% stress reduction system designed after the manner described in section 1 will increase the life by some 40% and may well be a most cost-effective system.

FUEL CONSERVATIVE SUBSONIC TRANSPORT

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ABSTRACT

This paper describes a fuel saving active control system being developed for commercial application of the L-1011* airplane in the early 1980s. Highlighted are features of the TriStar* that permit an effective yet simple load relieving system to be adopted. A description of the active control system, which involves integrated movement of both the aileron and horizontal tail, is given. The load relieving benefits obtained and the ability to increase wing span without major structural change are discussed. The potential fuel savings offered by this system is indicated. The presentation includes comments on the structural design criteria established for the system, the analytic models employed in the active controls analysis, and the initial breadboard control system hardware defined for ground and flight test purposes. Also described are ground simulation and flight test plans and results, and thoughts on further application of active controls for future consideration. The program is sponsored in part by the Energy Efficient Transport (EET) element of the NASA Aircraft Energy Efficiency (ACEE) program.

INTRODUCTION

Recent increases in fuel prices have placed a very large financial burden on the airlines of the world. The result has been a modification of operating practices and a turn to new technology to reduce fuel usage. Near-term modifications must be addressed as well as downstream new-technology aircraft.

The impact of rising fuel prices is shown in Figure 1. The dollars paid by U.S. domestic operators to purchase the fuel needed for 1976 operations was $2.14 billion. The cost of this same amount of fuel if purchased in 1973 would have been $850 million. Fuel costs have risen almost 250%. The difference in costs is greater than the total profits made by all the airlines involved totalled over the past ten years! Clearly one very important economic consideration involves fuel cost and the need for better fuel economy. How can aircraft be made more fuel efficient?

In the United States, NASA and industry have begun a program to define means for reducing the energy demands for air transportation. A large portion of this effort is being sponsored by NASA's Aircraft Energy Efficiency Program (ACEE), reference 1. The studies are examining near term solutions that involve revised operating procedures, simple modifications to existing airplanes and power plants, and technology advances that can be applied to existing aircraft either as modifications or design derivatives. Far term technology studies are also being conducted to examine new concepts that hold promise of large reductions in fuel consumption - laminar flow boundary layer control and advanced turboprop propulsion systems. Both of these new ideas will require practical solutions to some difficult obstacles before their benefits can be realized. This paper will concern itself with the near term technology for achieving better fuel efficiency.

Several fuel conservation refinements provide attractive answers in small measure. Modified operating procedures, aerodynamic clean up and improved sealing to reduce leakage, and minor airframe or engine modifications have been and are being implemented quickly and at low cost.

Typical flight developed hardware improvements being applied to the L-1011 are shown in Figures 2 and 3. Small contour changes that influence local flow behavior in sensitive pressure gradient regions of the airplane can produce small but worthwhile reductions in airplane cruise drag. Such refinements are difficult if not impossible to find in wind tunnel tests. Illustrated in Figure 4 is a change in engine nozzle geometry that reduces core exhaust leakage and improves exhaust performance - this refinement was initially developed by Rolls-Royce and is now a production modification for all on-line L-1011 aircraft and is retrofitable to all airplanes already in service. Each of these changes offers cost effective means for improving fuel economy.

The next steps to be taken involve application of new emerging technologies combined with airplane redesign. These steps include new wing shapes, application of composite materials, and use of active controls. The next several paragraphs will discuss each of these schemes, review their potential in light of the questions at hand, and establish why active controls technology has the greatest prospect for offering near term, low cost, effective relief to the airline economic dilemma.

*Trademarks of Lockheed Corporation

1976 FUEL COST FOR U.S. DOMESTIC TRUNKS	$2,142,265,000
PRE-CRISIS COST OF SAME FUEL	$850,000,000
ΔCOST	$1,292,265,000

THIS ONE YEAR HIGHER FUEL COST EQUALS THE AFTER-TAX EARNINGS OF ALL U.S. DOMESTIC AIRLINES FOR THE PAST 10 YEARS

Figure 1. Impact of Fuel Prices

Figure 2. Diverter Fairing

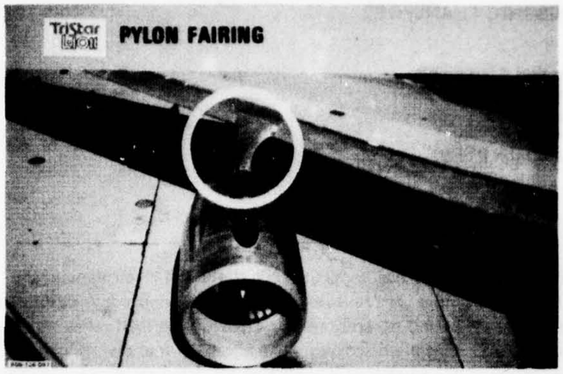

Figure 3. Pylon Fairing

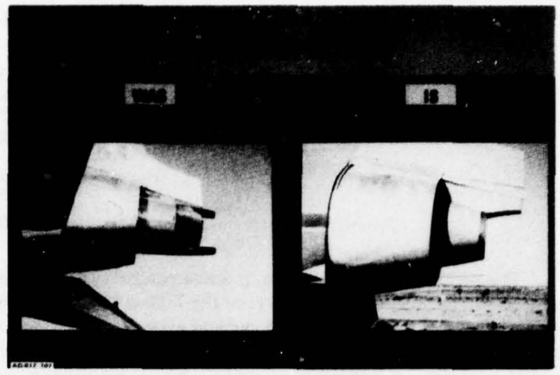

Figure 4. Afterbody Fairing

New Wing Benefits

New wing designs will adopt refined supercritical airfoils, Figure 5, that feature more rounded leading edge radii and cambered trailing edges. These revisions reduce shock wave strength, provide better chordwise loadings, and permit use of reduced sweep and thicker spars. In addition to airfoil shape alterations, today's fuel costs will warrant increased wing span to more favorably balance the new trades between added structural weight and better fuel efficiency, Figure 6.

The combined effect of the new supercritical airfoil and planform changes will provide approximately eight percent fuel savings. This figure is based on a comparison with the L-1011, and relates to the profile shape changes of Figure 5, a five degree reduction in wing sweep, a 0.02 increase in thickness ratio, and an increase in aspect ratio from 7 to 10. Such a new wing for an existing airplane would need a new high lift system, new structural design, new fuel and landing gear systems, a revised fuselage center section, and a complete re-certification flight test program. Exact cost to the airlines has not been established, but the noted changes suggest that the price tag would be considerable.

In summary, a new wing on an existing airplane design could be made available in 2 to 3 years, and offer eight percent fuel savings, but would be costly because it means revision of the basic backbone of the airplane design. New technology wing designs will more logically appear on new aircraft and new engine combinations of the future that will enter service in the late 1980's, after established airline purchasing power and market needs warrant these "all - new" transports.

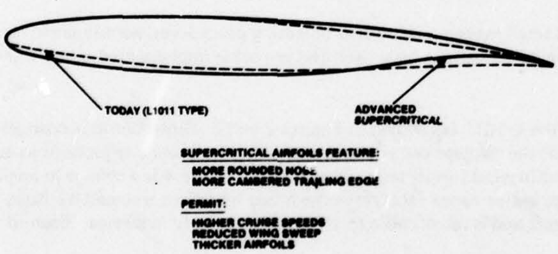

Figure 5. Advanced Airfoils

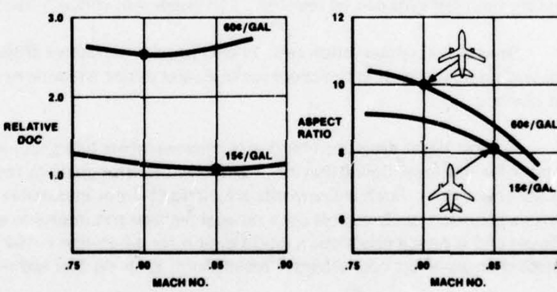

Figure 6. Wing Geometry Trades

Composite Materials Potential

Use of composite materials for aircraft structure portends empty weight savings of 5 to 10 percent and attendant fuel reductions of 2 to 4 percent. Parts count is reduced and fabrication is simplified using bonded joints. Structural properties can be tailored to meet desired strength characteristics, thereby improving structural efficiency. Corrosion resistance is enhanced. Composite material and labor costs are projected to be significantly reduced when the technology becomes established. These are the benefits of composite structures.

All these advantages justify the concerted effort to develop these new materials for aircraft application, Figure 7. The NASA ACEE program is dedicating a sizeable portion of its funds towards composite technology research and development.

These programs are disclosing two findings - the fuel savings potential gained from composites usage is small when applied as a substitute for a metal component, and secondly, much more time and effort is required before extensive use of this new material will be commonplace. The full benefits of composite application will be realized when the weight savings are employed to scale down a new airplane design. When this is done, performance, fuel economy, and economics will all benefit because of the smaller wing, engine, wheels, brakes, and empennage that result from the weight savings. The "all-new design composite airplane" is many years away, however. Many detailed considerations relating to design practice and procedures, tooling requirements, manufacturing and quality assurance procedures, and maintenance practices need to be established. The most optimistic projections would suggest the late 1980's as the beginning of the era of composite aircraft. Use of composite materials for near term relief of the airline's current economic dilemma does not hold forth as granting large benefits.

Active Controls Impact

Active controls application appears as the most likely candidate for providing cost effective near term means for gaining fuel conservation, Figure 8. This new technology has much in its favor. It can be available in the early 1980's, it can be applied at relatively low cost, and can be treated as a modification to existing aircraft. It may, in fact, for some applications be retrofitted to airplanes, already in service.

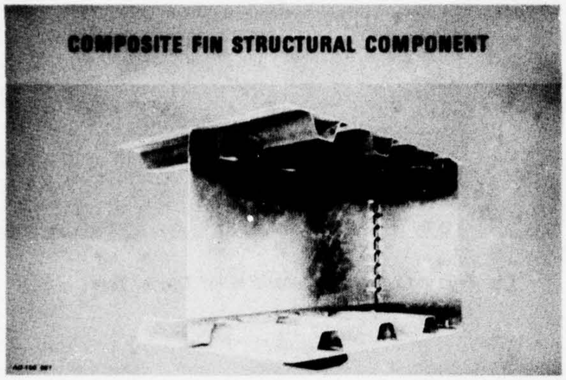

Figure 7. Composite Fin Structural Component

WHAT ARE THEY?
- ☐ CONTROL SURFACES ACTIVATED BY COMPUTERS USING INPUTS FROM SENSORS NOT PILOT COMMANDED

WHAT CAN THEY PROVIDE?
- ☐ EXPANDED OPERATING ENVELOPE
- ☐ WEIGHT SAVINGS
- ☐ REDUCED FUEL

Figure 8. Active Controls Definition

Systems like these exist today on commercial transports, Figure 9. The Mach trim compensator senses Mach number changes, and when commanded by computer, moves an actuator to reposition the stabilizer, without any signal from the pilot. Flap load relievers sense airspeed and flap deflection, and a computer-actuator system regulates flap angle following a programmed schedule. The L-1011 yaw damper and autoland systems are other examples of present day, certified, active control systems.

What applications of new technology active controls will help to conserve fuel? There are two - load relief systems and those that allow relaxed static stability.

Load Relief

Figure 10 illustrates the manner in which control surfaces can be used to relieve wing bending loads. Shown is the normal design load distribution for a wing, and the distribution which could be produced using symmetrical deflection of outboard ailerons as load relievers. Both load diagrams represent the same structural design wing lift, but the distribution with ailerons deflected has moved the center of pressure inboard, thereby lowering the wing root bending moments. For normal cruise flight, the loading conditions are not critical, and the ailerons move back to a faired position.

CRUISE

- MACH TRIM
- RUDDER LIMITER
- YAW SAS
- AUTOPILOT

T.O. & LDG.

- FLAP LOAD RELIEF
- DIRECT LIFT CONTROL
- AUTO LAND

Figure 9. L-1011 Active Control Applications

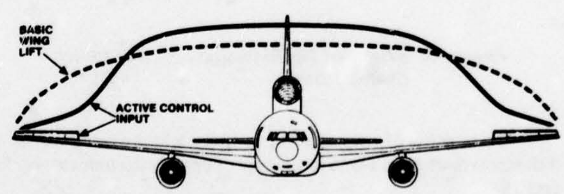

Figure 10. Load Redistribution with Active Controls

Implementation of this scheme suggests several alternatives. The reduced bending moments could permit structural redesign of the wing to save weight. Or the benefits of lower root bending moments could permit increases in aircraft operating weights without need for structural redesign to strengthen the wing. However, neither of these possibilities directly benefits airplane efficiency in terms of fuel conservation.

The energy-efficient way to exploit the load relief benefits of active controls is to modify the wing as shown in Figure 11. Wing span is increased, by a modest amount. For the L-1011, the tip extension planned is four and one half feet per side. These additions have a direct favorable effect on flight efficiency, since induced drag is measurably reduced. For the L-1011, the benefit is a 3 percent fuel saving.

The added span generates an additional wing lift increment at the tip region. Under critical flight conditions, this added tip lift would be unacceptable. However, aileron deflections programmed by an active controls system can be used to relieve this undesired lift increment, and allow for tip extension without wing structural redesign.

Wind tunnel tests have shown that this 4.5 foot extension can be installed without need for stall protection in the high lift configuration, Figures 12 and 13. The slats do not have to extend outward. Therefore the tip extension proposed is a simple structural change accomplished by replacing the original removable tip with the new version.

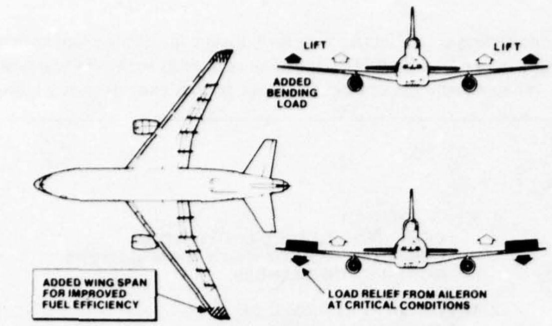

Figure 11. Active Controls for Fuel Conservation

Figure 12. Active Control Wind Tunnel Test

Relaxed Static Stability

The functions of a horizontal tail surface are to provide longitudinal pitch control and ensure natural longitudinal stability characteristics. Both are obviously necessary for satisfactory commercial transport handling qualities. However, it is possible to substitute artificial for natural longitudinal stability without compromising flight characteristics - such is the proposed active controls application suggested by the concept of relaxed static stability.

The impact of relaxing the apparent stability requirement which allows for a reduction in static margin is to permit adoption of a smaller tail size, Figure 14. With proper care, control power needed for nose wheel lift off and stall recovery can be retained with the new smaller tail - greater incidence travel, reduced sweep, longer chord elevator, etc. This same tail, properly integrated into an active control system, can provide the necessary stability contribution that will ensure good flying qualities.

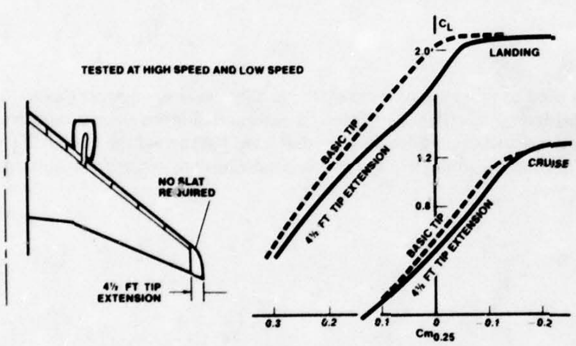

Figure 13. Effects of Tip Extension on L-1011 Pitch Characteristics

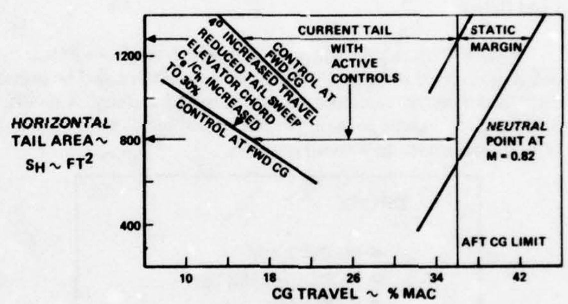

Figure 14. Benefits of Relaxed Static Stability on Tail Size

The benefits of relaxed static stability and the resultant tail size reduction offers lower levels of airplane drag and lower tail weight. Tail size reduction is significant - approximately 25 percent; performance is improved by 3-1/2 percent in range or fuel flow, Figures 15 and 16.

CURRENT HORIZONTAL TAIL AREA	1282 FT^2	
PROPOSED HORIZONTAL TAIL AREA	800 FT^2	
FUEL FLOW OR DRAG REDUCTION	2.6%	3-1/2% RANGE
WEIGHT REDUCTION	1700 LB	

Figure 15. Relaxed Static Stability Impact on Fuel Savings

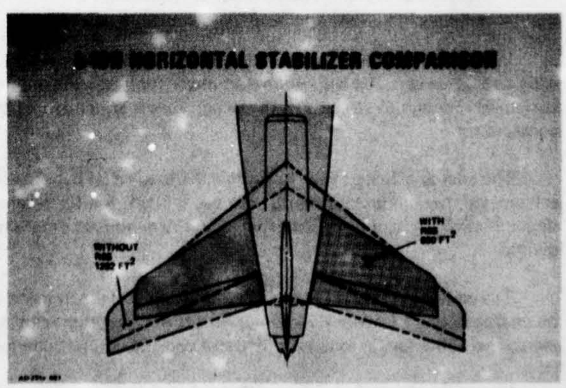

Figure 16. L-1011 Horizontal Stabilizer Comparison

Fuel Savings Potential

The combined impact of load relief and relaxed stability applications of active controls results in a 6-1/2 percent fuel savings for the L-1011 airplane. Development cost will be less than one tenth that of a new wing design, yet the benefit is almost as good. Realization of the benefits can be experienced by 1985; the appearance of a new wing design for a new technology aircraft is not likely until the 1990's.

Figure 17 integrates the active control fuel savings potential for an airline fleet of thirty L-1011 airplanes operating for 10 years. Assuming a representative fuel cost of 40 cents per gallon, the fuel reduction creates a monetary savings of 45 million dollars.

The incremental cost due to maintenance has been investigated and found to be insignificant compared to the projected fuel cost saving. This assessment was based on the maintenance cost of similar systems, particularly the present yaw stability augmentation system on the L-1011. The maintenance cost increment, including both material and labor, is estimated to be less than one percent of the projected fuel cost saving.

Certification Approach – Guidelines and procedures for certificating current L-1011 load relief systems (yaw damper, rudder limiter, flap load relief) were developed during the basic L-1011 type certification program and within the framework of the current civil air regulations. It is expected that the certification basis for the presently contemplated wing load relief functions will follow a similar pattern.

The basic premise in this method is that the safety criterion can be directly related to existing design criteria known to have demonstrated a satisfactory level of safety in service. With this premise, a system reliability requirement can be specified as a function of the effectiveness of the active controls in reducing the design loads and a design margin above the full potential of the active control system. A design margin is necessary because without it, an infinite reliability would be required to comply with the basic premise. An example of the application of this procedure is discussed below.

Probability techniques for analyzing load relief systems are illustrated by Figure 18 taken from Reference 3. Here the design vertical tail load is related to the vehicle response to lateral gusts, for which the probability distribution is known. The design load, associated with one occurrence in a 50,000-hr. aircraft life, could be reduced from the level F in the figure to the level G - a one-third reduction - by installing a totally reliable yaw damper. If an extremely conservative assumption were made that the yaw damper was inoperative 3 percent of the time, then the design load would be at the level H, representing the combined probabilities E associated with a 97 percent operative, 3 percent inoperative yaw damper. This conservative design load level H is only slightly higher than the best level G, and represents an attractive (approximately one-quarter) reduction from the no-active-control (no yaw damper) value F.

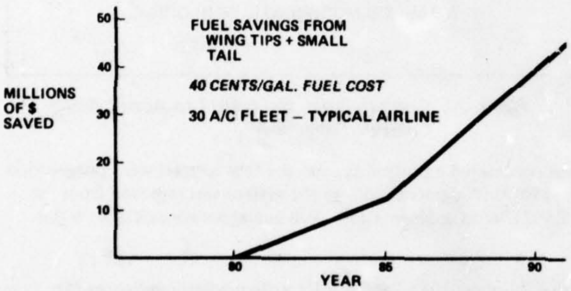

Figure 17. Impact of Active Controls

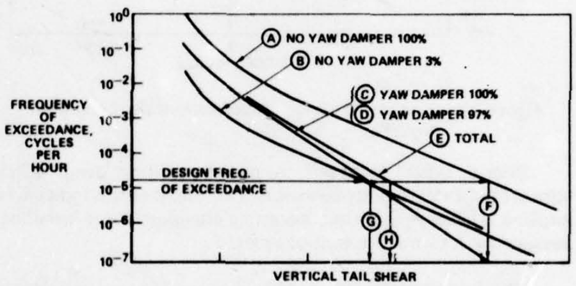

Figure 18. Frequency of Exceedance of Vertical Tail Shear With and Without Yaw Damper

From this illustration it may be seen that significant reduction in design loads and structure weight may be obtained with "state-of-the-art" active controls; that is, controls that may be inactive part of the time. The amount of reduction is usually limited by other considerations; in this case, vertical tail loads required for asymmetric power conditions.

It is also of note from service experience that the L-1011 yaw damper has been inactive only about one hour for every 100,000 flight hours, although the "dual-dual" channel design permits scheduled dispatch with one channel inoperative. The record is three orders of magnitude better than the original design assumptions. It suggests that future designs will assume a much lower fraction of inoperative time. However, in the typical case given in Reference 3, the design load for the totally reliable system would be only 7 percent lower than that of the assumed 97 percent-reliable system.

The above example relates to the treatment of gust-induced design loads. Here the probability of exceedance of various gust velocities can be related to known atmospheric distributions and to previous experience with airplanes having known safety records relative to flight in turbulence.

The approach for active control of maneuver loads will be on a similar probabilistic basis, relating the probabilities of exceedance of various acceleration levels to the design loads. Data from Reference 4 provide an example of maneuver load factor probabilities. Figure 19 from Reference 4 indicates for the 4-engine turbojet that load factor exceedances due to operational maneuvering have lower probabilities than those for gusts. Check-flight maneuver load factors tended to be somewhat higher. However, because of the low exposure time to check-flight maneuvers, it is believed that this will not contribute significantly to the overall probability of maneuver load exceedance which will be dominated by the operational maneuver load experience appropriately adjusted to provide for the in-flight availability of the active system.

NEAR TERM IMPLEMENTATION

Benefits of Derivative Approach

There are numerous benefits gained by applying active controls first to an existing airplane design, rather than to an all new technology airplane. Foremost is the significantly reduced level of risk. By starting with an already developed certified transport, most of the aerodynamic and structural properties are identified by virtue of wind tunnel tests, analytic modeling, and ground and flight evaluations. Aeroelastic and flutter characteristics are established, and control effectiveness/response behavior is in hand. Knowledge is full-scale real-world.

Steps to be taken in applying active controls to an existing airplane design are simplified because test facilities and hardware are immediately available. Knowledgeable technicians are at hand. Development time is reduced. All these considerations suggest reduced development cost.

The derivative approach to active controls development logically permits evolutionary progress, with small steps being taken first before the most ambitious control configured airplane design is undertaken. This philosophy offers gradual exposure to reliability concerns and redundancy and fault tolerant design requirement definitions. Certification considerations and airworthiness demonstrations are taken on gradually, a step at a time. A fuller understanding of the process will naturally emerge.

L-1011 Design Features

A number of design features inherent in the L-1011 enhance the ability to incorporate active controls, Figure 20. The fact that the outboard aileron is used for all flight regimes is beneficial, since the basic flight control system and wing stiffness characteristics need not be altered in order to adopt this surface to provide load relief functions. The L-1011 stabilizer is an all-moving fully powered control surface, that can readily be modified to provide fast movement and needed redundancy for active control application.

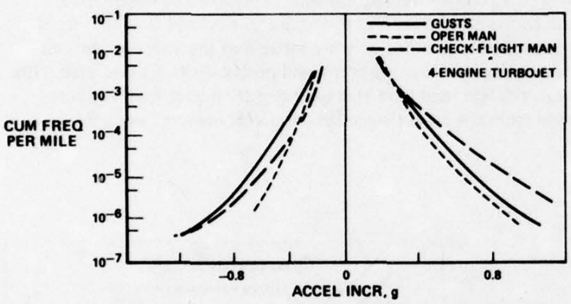

Figure 19. Gust and Maneuver Accelerations for Long-Haul Airplanes

- MULTIPLE REDUNDANT SYSTEMS
- IRREVERSIBLE CONTROLS
- ALL MOVING FAST STABILIZER
- *PROVISIONS FOR OUTBOARD AILERON AUGMENTATION SERVO-ACTUATORS*
- OUTBOARD AILERON USED IN ALL FLIGHT REGIMES

⇩

NO MAJOR MODIFICATIONS REQUIRED

Figure 20. Adaptability of the L-1011 to Accept Active Controls Technology

Prior to L-1011 flight test, the need for artificial damping in roll was considered a possibility, and the first aircraft were provisioned with artificial roll stability augmentation. Flight results indicated no need for this augmentation, so the system was removed from the airplane. However, space and mounting provisions for re-installing outboard aileron augmentation servo-actuators are existent on the airplane for future active control needs.

The development of the L-1011 auto-land system into a fail operative, fully certified Category III system greatly enhances the ability of the basic control system to adopt active control technology. Much of the system already possesses redundancy and control system design logic that precludes any flight safety degradation when load relieving active control system additions are installed. Much of the design approach and fail operative criteria developed during the autoland system design are applicable to the extensions in design that are to be adopted.

L-1011 PHASE 1 ACTIVE CONTROLS PROGRAM

Lockheed initiated an L-1011 independent research and development program in early 1974 with the primary intent of developing improved TriStar derivatives through the application of active controls. The work startup was undertaken shortly after early studies of load relief applications on the C-5 airplane showed the favorable benefits that were attainable. These gains plus the virtues of the L-1011 design led to the implementation of a dedicated active controls effort.

Throughout, the program has been structured to first develop maneuver load control, elastic mode suppression, and gust alleviation systems. Later, a phase two program would develop the relaxed static stability active controls system.

Structural Characteristics

Three active control applications are considered as most likely to benefit the L-1011. These are:

1. Maneuver Load Control (MLC) in which symmetric aileron deflection is used to redistribute wing airloads during maneuver.

2. Elastic Mode Suppression (EMS) in which symmetric aileron is used to damp the first wing bending elastic response in order to reduce gust loads. The horizontal stabilizer could conceivably also be used to damp elastic modes to reduce gust loads. The EMS concept could also be used to enhance flutter margins.

3. Gust Alleviation (GA) in which the horizontal stabilizer is used to reduce airplane response (at the center of gravity) to gust, i.e., reduce the short period response to gust.

Three existing control surfaces (the inboard and outboard ailerons and the all-movable horizontal stabilizer) were considered as being logical active control surface candidates, Figure 21. The outboard aileron was an obvious candidate for providing load alleviation. Need for an active horizontal stabilizer was anticipated to provide gust alleviation and to produce trim moments that would compensate for the trim shift created by symmetric aileron deflections. Early analyses indicated that the contribution from the inboard aileron was small and further consideration of this surface was halted. The final test system uses the outboard ailerons and stabilizer.

Fifteen degrees of up aileron deflection provide the bending load relief levels indicated in Figure 22 for 2.5 g limit design flight maneuvers. Figures 23 and 24 show typical wing design conditions and their zones of influence on the design of the wing spars and surfaces. Those portions of the wing spars that can be favorably affected by the proposed active control system are indicated. Load relief benefits are realized along most of the wing span. Reductions in gust loads, significant for both limit and fatigue design, are provided by utilizing the outboard aileron and the horizontal stabilizer.

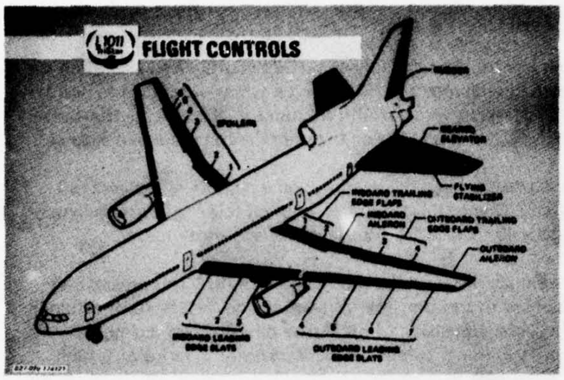

Figure 21. Flight Controls

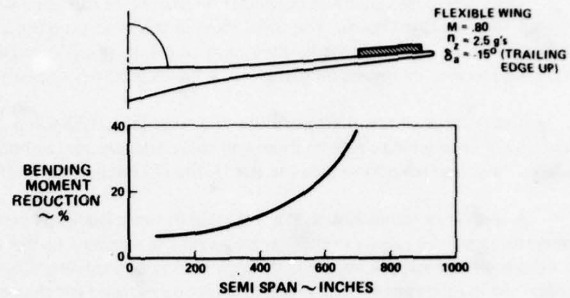

Figure 22. Bending Relief Produced by Ailerons

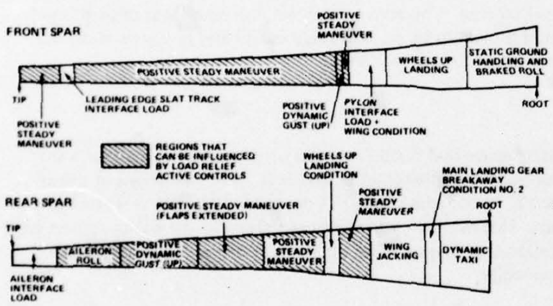

Figure 23. Typical Wing Spar Design Conditions

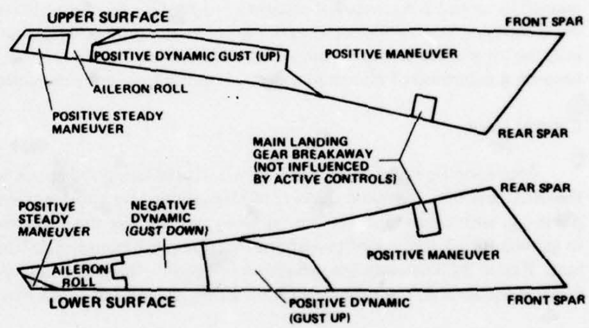

Figure 24. Typical Wing Surface Design Conditions

This load reduction capability allows for the span extension indicated in Figure 25. By proper accelerometer sensory input data feeding the active control system, it will be possible to accommodate this wing geometry change with essentially no wing structural change.

Combining the desired changes in structural loads with the anticipated capability of an active outboard aileron and horizontal stabilizer results in the establishment of specific performance objectives for the active system. These include use of ailerons to reduce loads in one-g flight (2 degrees passive) and maneuvering flight (15 degrees at 2.5g) and reduction of wing gust loads in both the flaps extended and clean configuration. Also included are objectives to avoid increases in severity of the horizontal stabilizer design loads and repeated loads spectra, and objectives to maintain or improve handling qualities and flutter margins.

Analytical Models

For the purpose of active control system synthesis, it was necessary to reduce the highly sophisticated mathematical models used in production loads and flutter analysis to moderate size and still provide sufficient accuracy to insure valid results.

A number of airplane mathematical models were utilized in the analysis. One of these, developed for use in conjunction with the quadratic optimization synthesis procedure, represented the airplane and control characteristics as first order differential equations in the time domain (state-space) in the form denoted as

$$\dot{\underline{x}} = F\underline{x} + G\underline{u}$$

where $\underline{x}$ is the vector of state-space variables and $\underline{u}$ is the vector of control variables. The model expressed in this form included three controllers (inboard aileron, outboard aileron and horizontal stabilizer), two rigid body modes (pitch and plunge), six structural modes, free-stabilizer pitch, gust-input, and actuator variables. The free-stabilizer pitch coordinate was introduced so that the stabilizer control system dynamics could be represented.

The equations were written first in terms of absolute displacements at 256 lumped mass points. These equations were then modalized in terms of the generalized coordinates indicated. The basic state-space equations were supplemented by matrices by which the state-space variables and their derivatives could be multiplied to give desired shears, bending moments, torsions, and local accelerations. The representation of the unsteady aerodynamics in the time domain involved approximating the aerodynamic forces in terms of the airplane-motion variables and their first three derivatives (and also the gust velocity and its first three derivatives). As a result, the second order equations became third order equations and the number of state variables became three times the number of variables indicated above, yielding a 40 by 40 "F" matrix.

Another analytical model used heavily in the analysis was the full scale production flutter model formulated in the frequency domain. This model was used in a parallel effort to explore other optimization techniques (particularly for the EMS concept). In addition, there was concern that the state-space model might not be sufficiently accurate to predict flutter characteristics. The flutter model includes forty elastic modes and two rigid body modes. It was correlated with flight and ground vibration data.

Control Law Synthesis Methods

The state-space quadratic optimization procedure utilizes a synthesis algorithm which defines a direct matrix algebra solution for the optimal feedback gains. The main steps of the procedure are presented in reference 5. The computer was programmed to perform the basic linear algebraic matrix operations to compute eignenvalues and eigenvectors, to perform solutions to algebraic matrix equations and to display results on computer graphics modules to permit a high degree of interaction between the computer and engineer.

The number of resultant gains was too large (e.g., 120 gains) to allow practical implementation, and a number of separate routines were developed to reduce these and still achieve a near optimal system. One of these routines, referred to as "spectral decomposition," was also used to reduce the size of the state-space model (from 40 x 40 to 12 x 12) for use in the laboratory testing.

A separate synthesis using the full scale flutter model was performed in parallel with the state-space analysis. This procedure operates in the frequency domain and utilizes the standard flutter equations with additional terms that include the effects of actuator forces and the feedback coupling through the various sensors. The method is an extension of the flutter optimization technique described in reference 6. The procedure involves solving for the phase and gain that provides a prescribed result in terms of system roots, such as the damping of a particular elastic mode. It is then relatively simple to establish a control law that will approximate this phase and gain. Additional program modules were developed that allow the above approach to be used to calculate required transfer function phase-gain relationships over the entire frequency range such that prescribed gust load reduction objectives will be achieved. These prescribed gust loads reductions are expressed in the form of power spectral density (PSD) plots, of key load and acceleration quantities, versus frequency. A separate program module was also developed to calculate transfer functions, using a least squares fairing technique, that will approximate the above determined phase-gain requirements. The above synthesis procedure was programmed into the interactive graphics computing system described in reference 7, and proved to be an extremely useful and powerful tool. It became the method of choice in determining the frequency-dependent portion of the control laws.

Control Laws

Four specific flight conditions were analyzed using the above two (state-space and flutter models) procedures. It was found that the MLC and EMS system criteria could be satisfied by a single transfer function that blended the signals from accelerations at the airplane c.g. with accelerations from the wing tips to drive the outboard aileron. In the case of the GA concept, difficulty was encountered in gaining substantial benefits without increasing horizontal stabilizer loads. However, the specified objectives for the entire system were met. Figure 26 illustrates the reduction in bending loads at a key wing station due to gust. The square root of the area under this curve is equal to $\bar{A}$, the root-mean square of gust load for a unit rms gust velocity.

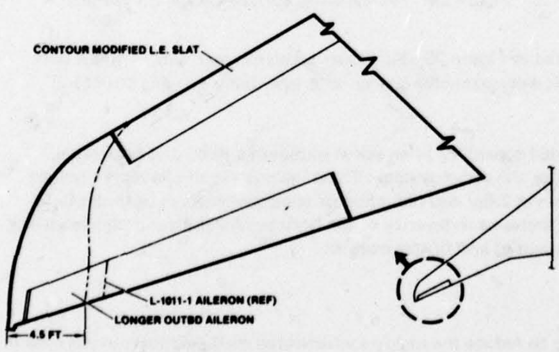

Figure 25. Extended Span Wing

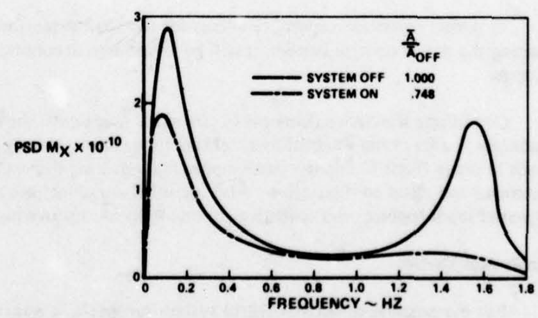

Figure 26. Power Spectral Density of Wing Bending Moment at Gust-Critical Wing Station

Performance Confirmation (Large Models)

The control laws derived from the above procedures were fed back into the large L-1011 production maneuver loads, gust loads, and flutter analysis programs. As pointed out earlier, these programs reflected flight test ground vibration, and wind tunnel results. Reference 8 includes a more detailed description of the loads models. The production gust model includes the rigid-body modes of plunge and pitch and 20 elastic modes. The gust and maneuver loads benefits were verified. The flutter analysis was conducted over sufficient speed and altitude ranges to establish stability for the control laws defined.

Hardware Identification for MLC/EMS/GA Systems

A primary objective of this program was to identify and develop a MLC/EMS/GA system for flight testing in 1977. The resulting system includes flightworthy "breadboard" components, sensors, and all interface hardware necessary to complete the installation on the flight test airplane. The system has been designed and fabricated, and has successfully completed extensive ground simulation testing and flight testing. The major elements of the system are briefly described below.

Computer – The computer receives signals from the sensors and computes command signals to the three servos which drive the horizontal stabilizer and outboard ailerons. The command signals are computed in accordance with the MLC/EMS/GA transfer functions developed from the analytical programs. The computer consists of twelve cards - two power supply cards, redundant sets of three cards each to implement the transfer functions, three cards to provide dual channel command signals to the series servos, and one card that contains all the necessary circuitry and logic to compare the dual channel computations and outputs.

Sensors – The sensors used to provide aircraft state signals to the computer consist of dual channel accelerometers at each wing tip, dual channel c.g. accelerometers, dual channel pitch rate gyros, dual channel stabilizer position Linear Variable Displacement Transducers (LVDT's) and single channel column force sensors at each column. The computer also requires dual true airspeed signals for gain scheduling; for the flight test program, these were input manually by a potentiometer. All the sensors are new hardware except for the force sensors and the stabilizer position LVDT's, which are part of the autopilot.

Outboard Aileron Series Servos – The original SAS computer box has been programmed to drive the outboard ailerons symmetrically by command signals from the active control system cards in the computer. These signals are superimposed on the roll command tracking from the inboard aileron surface positions. The authority of these servos is limited, compared to the eventual requirement, but is adequate for the flight test program. The limited authority assures safety of the airplane in the event of a servo malfunction.

Pitch Series Servo – An extensible link pitch series servo has been installed in the series trim input linkage to the four power servos driving the horizontal stabilizer. A special hydraulic interface was designed and fabricated to provide 1000 psi hydraulic power to the pitch series servo. This servo superimposes signals from the active control system computer on pilot commands and is limited in authority to assure aircraft safety in the event of malfunction.

Interface Hardware – This includes all the mechanical linkages, mounting brackets, electrical harnesses, hydraulic plumbing and various minor components necessary to complete the installation of the system for flight testing.

MLC/EMS/GA System –The hardware is integrated into the system as shown in the functional block diagram Figure 27.

Ground Simulation Test Program

A comprehensive ground simulation test program was completed prior to installing the system in the flight test airplane. This test program was conducted in the Vehicle Systems Laboratory at Lockheed's Rye Canyon Research Facility (Figure 28). A layout of the laboratory is shown in Figure 29. The test program had four broad objectives:

1. Development of component hardware
2. Evaluation of system performance
3. Verification of compatibility with other aircraft systems
4. Evaluation of failure modes

These tests were conducted either on the Vehicle Systems Simulator (VSS), the Visual Flight Simulator (VFS), or both simulators coupled together, as follows:

VSS Tests – The VSS consists of a mockup of the L-1011 cab and the framework to provide the exact geometric arrangement of the system hardware from cab controls to control surfaces including all mechanical, electrical and hydraulic interfaces as they exist on the airplane as shown in Figure 30. In general, the VSS is used to develop and verify individual components and evaluate the function of all hardware integrated into the system.

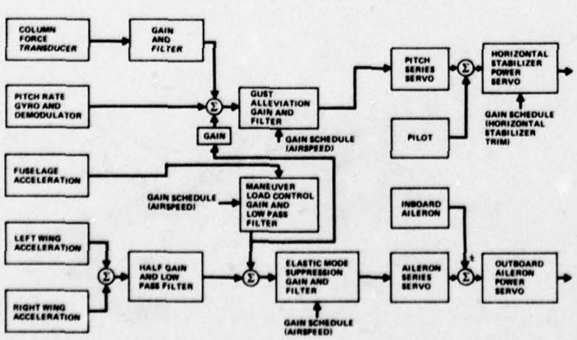

Figure 27. L-1011 MLC/EMS/GA System Block Diagram

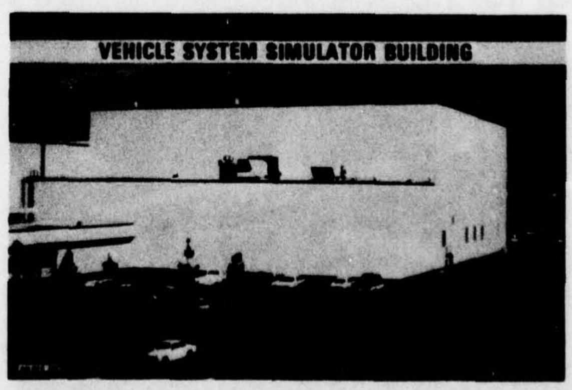

Figure 28. Vehicle System Simulator Building

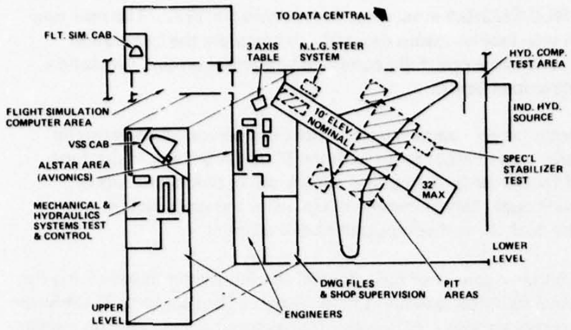

Figure 29. VSS Test Facilities-Rye Canyon

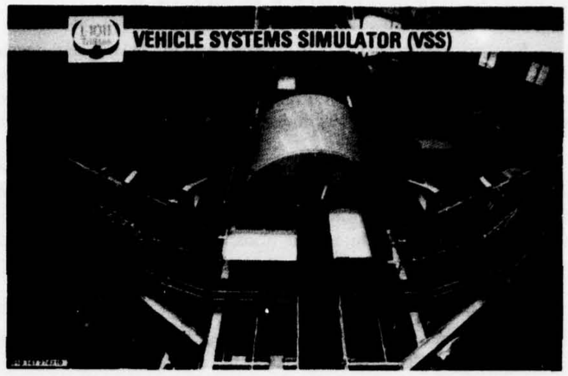

Figure 30. Vehicle Systems Simulator (VSS)

VFS Tests – The VFS consists of a mockup of the L-1011 cab (Figure 31) coupled to an analog simulation of the aeroelastic response of the aircraft. Pilot command signals are fed to the computer, and aircraft response signals are returned to the flight instruments. The aircraft model simulates the longitudinal characteristics (lift, pitch and airspeed), three elastic structural modes and turbulence. The VFS cab can be placed either on a fixed base or on a moving base that responds to the simulated aircraft motions.

VSS/VFS Coupled Tests – A position servo provides VSS control column tracking commanded from the VFS cab. This permits the pilot to fly the simulated airplane through the actual hardware system of the VSS and the actual surface position signals fed back to the VFS computer. This arrangement also allowed the active control system series servos to be commanded by VFS simulation of the active control computer until its fabrication was complete. Figure 32 illustrates the transient response of key airplane parameters due to an aileron step forcing function. The case illustrated is a low speed flight condition and the response is shown for both the active system operative and inoperative. It can be seen that the active system computer, driven by signals from the wing tip and c.g. accelerometers and pitch rate sensors, places additional commands on the outboard aileron and horizontal stabilizer in the direction to improve damping of the wing tip acceleration. These simulator predicted responses were used to compare with flight measured responses (to similar aileron step commands) as part of the many checks to insure that the system was performing in flight as measured by ground tests.

Figure 31. Flight Simulator

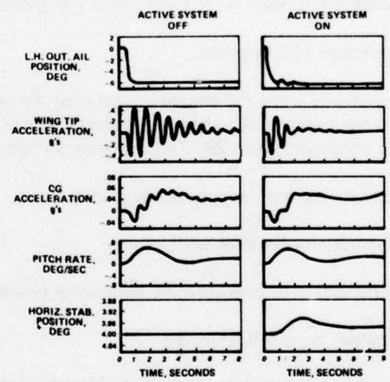

Figure 32. Simulator Model Response to Aileron Step Command

FLIGHT TEST PROGRAM

The flight test program is being conducted using L-1011-1 serial number 1001, Figure 33. This is one of several airplanes used in the basic L-1011-1 development and certification flight testing and has been retained by Lockheed for continuing flight test use. It was the primary airplane for structural flight testing, including loads measurements and flight flutter testing. Its instrumentation includes extensive strain-gage instrumentation calibrated to read shears, bending moments, and torsions at various locations in the wing, fuselage, and empennage, as well as instrumentation giving control surface positions, accelerations at various locations, airspeed, altitude, etc. (see Figure 34).

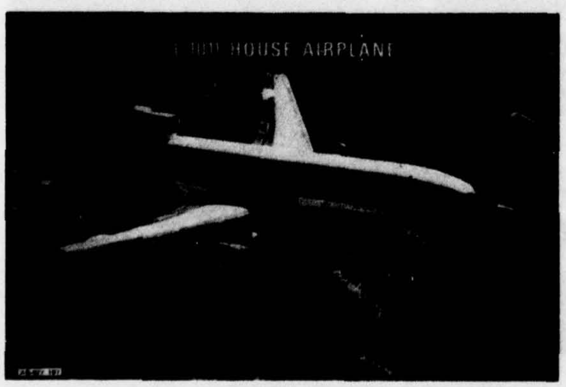

Figure 33. S/N 1001 Flight Test

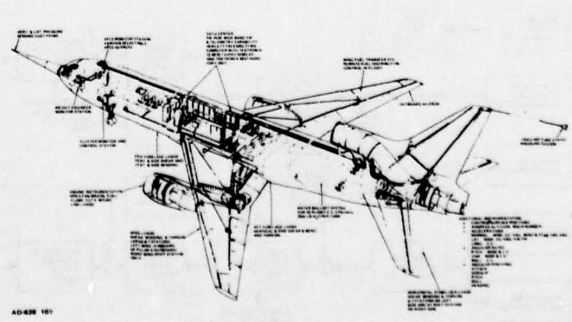

Figure 34. S/N 1001 General Arrangement

A gust boom and probe, previously used in conjunction with gust-load measurements made in 1971, are being used on the present program to measure the effects on gust loads of the MLC/EMS/GA system.

The flight test program and a part of the ground test program is sponsored in part by the Energy Efficient Transport (EET) element of the NASA ACEE program.

Available results from the baseline flight tests without extended tips are discussed in the following paragraphs. These tests were to verify the control functions, the airplane mathematical modelling, and the control effectiveness prior to flight with extended span.

Control Laws – Baseline Flight Tests

The MLC/EMS controls to the outboard aileron call for an MLC gain of 8.67^{o}/g at 180^{o} phase (18.8 dB, trailing edge up for up acceleration) at low frequency. The EMS function to damp wing bending calls for relatively pure damping, an aileron phase of $90^{o} \pm 20^{o}$ relative to tip acceleration, in the wing fundamental bending frequency range of 1.3 Hz to 2 + Hz. Gain requirements for damping were considerably less than for MLC. Figure 35 comparing the specified gain/phase relations with those measured in the airplane ground tests shows that these requirements were substantially satisfied. Special shaping circuits were used to minimize the phase change in the range from 1.3 to 2 + Hz.

Control laws for the stabilizer motion were limited for this baseline case to using the stabilizer to increase the pitch damping at low frequency, Figure 36. The gains were rolled off with increasing frequency to minimize effects on the elastic modes.

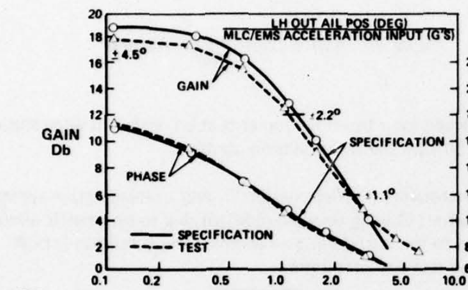

Figure 35. Aileron Control Laws

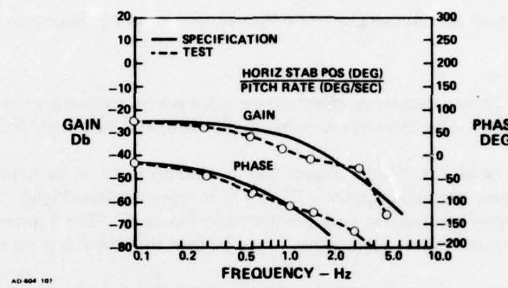

Figure 36. Stabilizer Control Laws

The control laws for both outboard aileron and stabilizer were selected to have substantial gain, phase and frequency margins to assure stability of the elastic modes. Flight envelope expansions were made at 1/2X, nominal, and 2X nominal gains. These showed that the gain margins were well over 2X throughout the flight envelope.

Airplane Modelling Effectiveness

Successful use of active controls for affecting elastic modes depends on the ability to calculate the elastic airplane characteristics with reasonable accuracy. In the case of the L-1011, the structural characteristics are well known from wing and tail stiffness tests, from ground vibration tests and from flight flutter testing. The predictions were verified during the baseline active controls flight tests by open-loop transfer function tests covering the range from low to high weights and from low to high speeds. Separate tests were made for response to stabilizer drive and for response to symmetric aileron drive for six flight conditions. Representative cases are shown in Figure 37 for aileron drive, and in Figure 38 for stabilizer drive. They show the predicted and measured wing-tip acceleration response per degree of control surface motion from low frequency to about 3 Hz. It is seen that the amplitudes are reasonably predicted, and that the phase predictions are excellent. This excellent phase accuracy has verified the interactive graphics flutter computing system as the method of choice in synthesizing control laws for elastic modes.

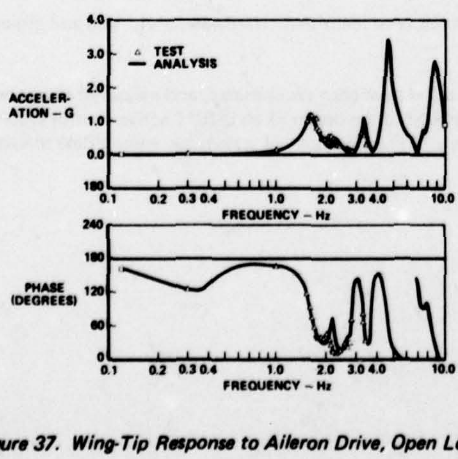

Figure 37. Wing-Tip Response to Aileron Drive, Open Loop

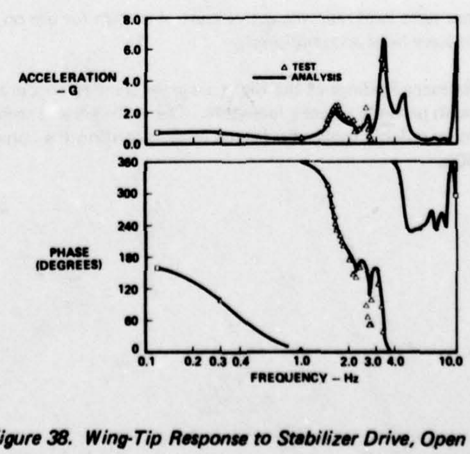

Figure 38. Wing-Tip Response to Stabilizer Drive, Open Loop

Active Control Effectiveness

The active controls were verified by turning the system ON (closing the loop) while performing the open-loop transfer function tests. Figures 39 and 40 show open- and closed-loop wing-tip acceleration response to aileron and stabilizer drives. It is seen that the active controls reduce the peak wing bending response by 45% for the aileron drive, and by about 35% for the stabilizer drive. The lower effectiveness in the second case is attributed in part to a low level of excitation, requiring less than 0.3 degrees amplitude of active aileron motion. The low amplitude resulted in non-linear effects, consisting of an extra phase lag that reduced the damping effectiveness at frequencies above the peak response frequency.

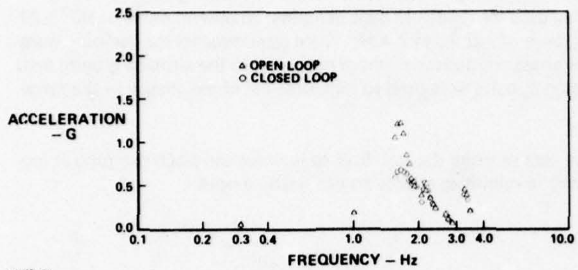

Figure 39. Active Control Effectiveness; Wing-Tip Response

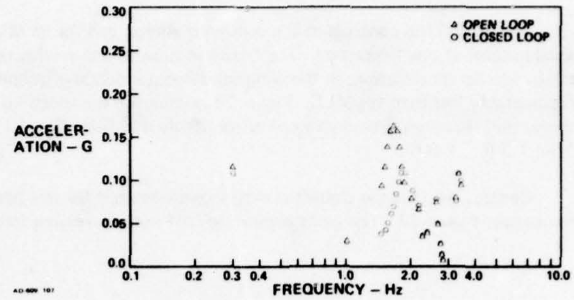

Figure 40. Active Control Effectiveness

The low-frequency effectiveness is shown by comparing the open-loop and closed-loop bending moments at an outboard wing station, Figure 41. The bending moment amplitude at low frequency (0.3 Hz) is reduced 50% by the active aileron control.

The MLC function was verified in wind-up turns, in push-down and pull-up maneuvers ("roller-coasters"), and in steady-state symmetric aileron deflections from -7^{o} to $+7^{o}$ in trimmed level flight. The spanwise variation of wing bending moment due to symmetric aileron deflection is compared with prediction in Figure 42. The moments are normalized to the 1-g bending moments. The condition is high cruise speed flight. Here again it may be seen that there is good agreement between test and prediction.

In summary, the available data from the baseline flight tests indicate that the systems perform substantially as designed. The systems and their mathematical description are equal to the task of applying active controls to the L-1011.

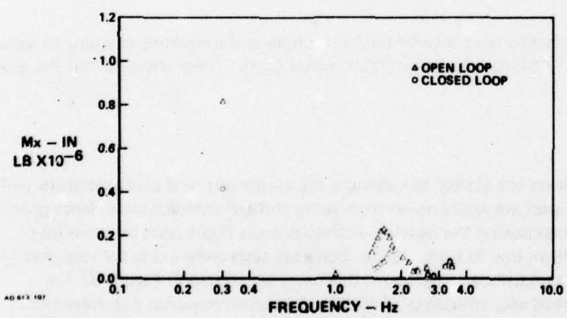

Figure 41. Active Control Effectiveness

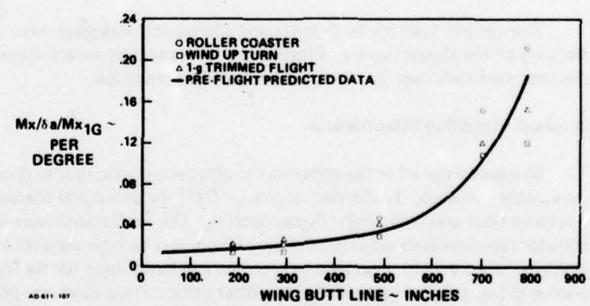

Figure 42. Comparison of Measured to Predicted Relative Wing Bending Moments

SUMMARY

A near term load relieving active control system for use on L-1011 aircraft has been identified. Hardware fabrication and ground and flight tests have been accomplished.

Preliminary findings of the flight program are very encouraging. No difficulties have been encountered, and measured responses are agreeing with predictions very favorably. The methods and techniques being applied to the design of an L-1011 active control system are proving to be reliable tools; and results to date confirm the conviction that this L-1011 active control system can enter airline service in the early 1980's.

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C-5A LOAD ALLEVIATION

by

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SUMMARY

The work on load alleviation systems for the C-5A at the Lockheed-Georgia Company began in 1967 and has progressed through several system variations to the present Active Lift Distribution Control System (ALDCS).

This article reviews the evolution of the present load alleviation system and presents a brief description of the system mechanization and a simplified functional block diagram. Comparisons of analytical and flight test measured maneuver and continuous turbulence loads are shown. Comparisons are also shown for ALDCS ON and OFF airplane response and wing stress measurements obtained during the C-5A Service Loads Recording Program. The effects of loads changes on fatigue damage rate predictions are discussed, with particular emphasis on the implications of multiple component load changes, i.e., reduced bending moments and increased torsional moments.

INTRODUCTION

The various load alleviation studies and the several design, hardware-development, and flight test programs, conducted at the Lockheed-Georgia Company since 1967, have been accomplished with different objectives in mind: 1) development of analytical methods and technology substantiation, 2) "static" load reduction for gross weight/cargo capability increases, and 3) fatigue load reductions for increased airframe fatigue endurance. Figure 1 shows the chronological evolution of these efforts. More detailed descriptions of these studies and programs, including design objective statements and typical analytical results, are contained in other publications.[1, 2]

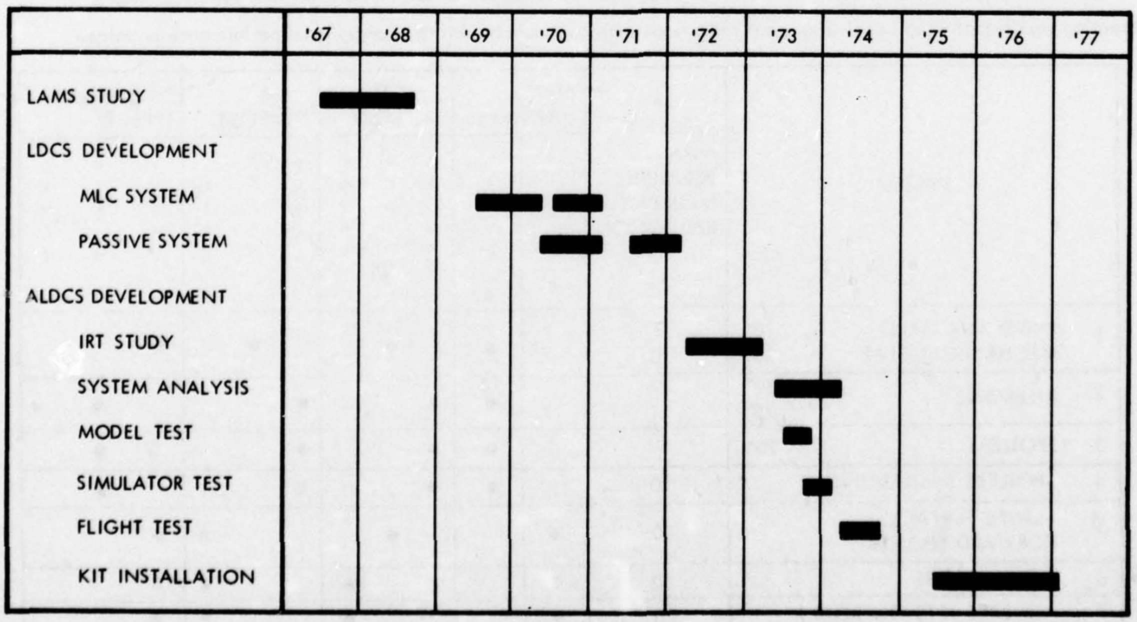


Figure 1 - C-5A Load Alleviation System Evolution

The present active control system for the C-5A is identified as an Active Lift Distribution Control System (ALDCS), the name having been derived from the fact that symmetric aileron deflections are employed as a primary means of providing reduced wing bending moments through modification of the wing spanwise additional lift distribution. The ALDCS program has progressed through a design/development and test phase and a production system design/fabrication phase, and hardware is presently installed on all force aircraft. This system was developed for the specific purpose of providing a significant wing fatigue life improvement through reduction of maneuver and gust incremental wing bending moments.

The sections that follow provide a review of the various load-alleviation systems that finally evolved into the present ALDCS, definition of the basic mechanization features of the system, comparisons of analytical and flight-test-measured loads and system response parameters, and discussions of the analytical problem of assessing the effects of changes in multiple-load components on fatigue endurance. Analytical wing fatigue life improvement factors (ALDCS-on/ALDCS-off), as derived by a load transfer analysis method, are shown for a specific mission profile mix.

LOAD ALLEVIATION CONCEPTS

Various load alleviation concepts have been considered and/or used during the C-5A program. The Aircraft Load Alleviation and Mode Stabilization (LAMS) Program conducted by Boeing Wichita and Honeywell under contract to the Air Force Flight Dynamics Lab involved the C-5A to a small degree. The Lockheed-Georgia Company participated by providing C-5A data to demonstrate the applicability of the analysis methods and techniques to another large flexible airframe. Although the LAMS C-5A System Analysis and Synthesis was based on a single flight condition, the study results concluded that a LAMS type control system could reduce structural fatigue damage rates during flight through turbulence without significant degradation of basic aircraft stability and handling qualities.[3]

During the conduct of the C-5A static test program in mid 1969, it became apparent that some form of wing maneuver load reduction system was highly desirable for the purpose of reducing maximum wing upbending loads - a "strength design" load reduction rather than a fatigue load reduction system. Figure 2 illustrates the various load reduction techniques evaluated and provides summary type trade-off information relative to load reduction magnitudes, hardware changes, development complexity, etc. The uprigged aileron concept was selected as the most practical means of obtaining significant wing bending moment reductions with minimum hardware change/least performance penalty.

	SYSTEM		PERCENT MAX BENDING MOMENT REDUCTION	NEW HARDWARE		MOD. EFFORT			R & D EFFORT			PERFORMANCE EFFECTS		
				SURFACE	CONTROLS	MINOR	MEDIUM	MAJOR	SMALL	MEDIUM	LARGE	NEG.	MINOR	LARGE
1	INB'D AUX. SPLIT MECHANICAL FLAP	10° 25°	2 10		●		●			●		●		
2	AILERONS	10° 25°	7 11		●	●			●				●	
3	SPOILERS	25°	7		●	●			●				●	
4	SPOILERS & AILERONS		10+		●	●			●				●	
5	LOWER SURFACE FORWARD SPOILER		10	●			●				●	●		
6	BLOWN FLAPS		10	●				●			●	●		
7	ADVERSE JETS OUTBOARD		10	●				●			●	●		
8	RESERVE FUEL/DISTRIBUTION		2						●			●		
9	CLIPPED WING TIP		5			●			●					●
10	AERODYNAMIC FENCE		1			●				●		●		

Figure 2 - C-5A Load Alleviation Concepts

Maneuver Load Control - MLDCS - A development program was initiated to design, develop and flight test an active load reduction system. The primary objectives of the system were:

- o Reduce positive maneuver maximum wing root bending moments by 10%
- o Minimize effects on handling qualities
- o Minimize effects on aircraft performance
- o Utilize existing hardware with minimum new components
- o Provide "full time - fail operative" system.

Since it was desirable to reduce the maximum unbending moments for "static strength" purposes only, the concept evolved into a system having a dead band below 1.5g with the system becoming active at higher load factors. This resulted in no drag penalty during takeoff, climb, cruise, etc., except during infrequent maneuvering to load factors above 1.5.

System implementation utilized existing, modified, and new hardware as shown by Figure 3. Normal accelerometers located at the wing first bending node line provided "rigid body" motion intelligence with minimum gain and phase effects for higher frequency responses. The existing pitch and yaw/lateral Stability Augmentation System (SAS) computers provided the means of introducing desired commands to the ailerons and pitch compensation inputs to the inboard elevators. The breadboard MLDCS computer was designed to accept inputs from the accelerometers, a Mach signal from the Central Air Data Computer (CADC) for gain scheduling purposes, a flap position signal to deactivate the system in flaps extended configurations and a touchdown signal to deactivate the system during landing impact and ground operations. Outputs were provided to the yaw/lateral and pitch SAS computers, through which aileron and inboard elevator deflections are commanded, and to flight crew monitoring and control hardware. Triple channel redundancies and fail safe features were incorporated in the system to fulfill the full time operative requirement.

Structural load improvement attained with this system is illustrated by Figure 4. The MLDCS affects only maneuver loads at load factors above 1.5 thus there is no significant effect on fatigue loads resulting from the maneuver source. Gust loads are likewise not significantly affected due to both the rather high "g" onset level and the limited frequency response range of the system. During the development program, a compromise was made on aileron deflection magnitude due to the undesirable increase in positive wing torsion along with the desirable reduction in wing bending moment. Desirable bending moment reductions which reduced wing lower surface axial stress levels had to be limited since wing front beam web shear flow increased significantly due to the increased torsion loads. The final scheduled maximum aileron deflection was set at ten degrees.

The development program included simulator testing and flight testing in addition to the analytical investigations. The flight test program evaluated handling qualities and provided substantiating data for structural load reductions.

The effects of this system on aircraft performance and handling qualities are negligible. During flight testing it was difficult, if not impossible, to determine when this active system was operating. A more detailed discussion of this system is contained in reference 2.

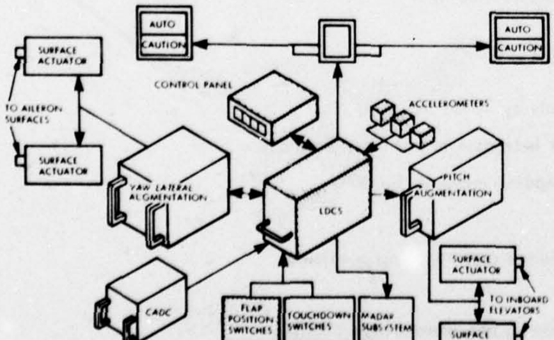


Figure 3 - MLDCS System Components

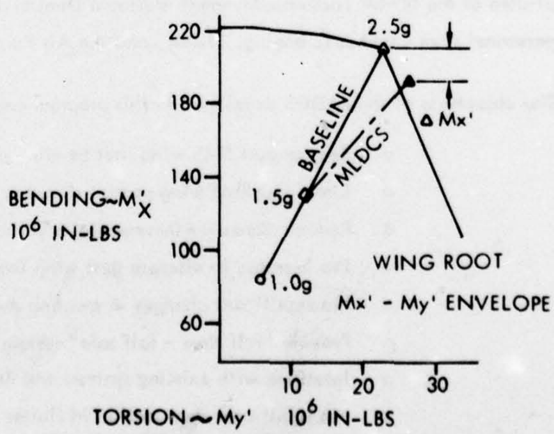


Figure 4 - MLDCS Loads Improvement

Passive Lift Distribution Control System - PLDCS - During the MLDCS development program, it became clear that some form of fatigue loads reduction was highly desirable. Moreover, it was desired to simplify the MLDCS from the standpoint of reduced new hardware in order to obtain early fleet incorporation of a load reduction system - thus the passive LDCS program was instituted.

The primary objectives of this system were:

- Reduce positive maneuver maximum wing root bending moments by 10%
- Provide service life improvement by reduced 1.0g mean bending moments
- Minimize effects on aircraft performance
- Utilize existing hardware with minimum new components.

The PLDCS concept evolved into 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 5 degrees of uprig above 20,000 feet and 10 degrees below 20,000 feet. The objective to attain a service life improvement required that the 5 degree setting be utilized in the takeoff and landing configuration in order to provide the reduced mean load benefit throughout the flight profile. System implementation then became a rather simple matter of using the existing individual aileron trim capability as an interim measure until the equally simple production changes could be incorporated by field level kit installation. The C-5 force has been using the PLDCS, interim and/or production systems, since November 1971.

Active Lift Distribution Control System - ALDCS - In late 1972, the C-5A Independent Structural Review Team (IRT) included the development of an active LDCS in the list of options available to the Air Force as a means of extending the service life of the C-5A primary wing structure. Air Force review of the IRT options resulted in a decision to proceed with an ALDCS development program in mid 1973. This program involved the Lockheed-Georgia Company as prime contractor with participation of The Boeing Company (Wichita Division) and Honeywell as subcontractors. The C-5 System Project Office was the contracting authority having technical and management control of the program with the Air Force Flight Dynamics Lab providing technical assistance and program review functions.

The ALDCS development program has progressed through a design/development/test phase followed by production kit manufacture and installation on the 77 C-5 force aircraft. To date, approximately sixty thousand flight hours have been accumulated on the ALDCS equipped aircraft in normal Military Airlift Command training and logistics operations.

A unique aspect of this development effort was the use of a dynamically and elastically scaled model having an onboard hydraulic system to provide power for activation of the ailerons and horizontal stabilizer. The control system was operated by a console mounted analog computer simulation of the ALDCS computer using inputs from the onboard ALDCS sensors. This model provided an experimental dynamic loads/flutter data acquisition tool with which to gain confidence in the analytical methods used in development of the ALDCS mechanization. The model wind tunnel test program was accomplished at the NASA Transonic Dynamic Variable Density Tunnel at Langley AFB and involved a test team consisting of personnel from Lockheed, Boeing, NASA, and the Air Force. (Reference 4)

The objectives of the ALDCS developed in this program are as follows:

- Reduce gust RMS wing root bending moments by 30%
- Limit gust RMS wing root torsional moment increases to not more than 5%
- Reduce maneuver incremental wing root bending moments by 30%
- No increase in discrete gust wing loads
- No significant changes in existing performance and handling qualities
- Provide "full time - fail safe" system
- Interface with existing systems and use existing hardware where possible
- No significant degradation in flutter margins.

System mechanization was derived using the proposed IRT schematic as a baseline system. This system in itself had its beginnings in the C-5A LAMS pitch axis mechanization. System implementation includes PLDCS and involves use of existing control surfaces, actuators and servos, modified SAS and CADC computers and new hardware. This system, as was the MLDCS, was designed to interface with existing SAS and autopilot systems. System design requirements and test results are discussed in detail in references 5 thru 10.

COMPARISON OF C-5A LDCS SYSTEMS

The three systems which have been developed and flight tested are compared in Figure 5 relative to major objectives, means of implementation, loads improvement magnitudes and aircraft performance/handling qualities effects.

It should be emphasized that the paramount objective in each of these systems was some form of wing bending moment reduction - either strength or fatigue related - with secondary objectives of system simplicity and minimum effects on aircraft performance/handling qualities. No attempt was made to provide a "mode stabilization/control" function for purposes of flutter boundary extension or ride control improvement.

Some of the trade-offs or compromises between conflicting objectives are apparent from the comparison chart. Note specifically that the price of obtaining reduced mean bending moments, as provided by the Passive System, is an aircraft performance penalty. An offsetting benefit of this system was the ability to attain an almost immediate incorporation with a minimum hardware impact.

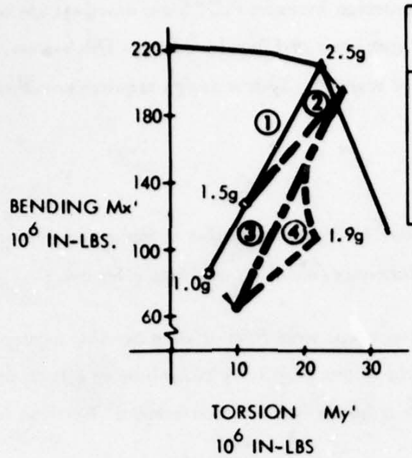
The next variation - to provide reductions in maneuver and gust incremental bending moments while retaining the reduced mean loads generated a significantly larger hardware design/development problem than that of the original maneuver load control MLDCS and in addition retained the performance penalties of the passive system.

SYSTEM	MAJOR OBJECTIVES	MEANS OF IMPLEMENTATION	LOADS IMPROVEMENT MAX STATIC Mx'	FATIGUE Mx MEAN	FATIGUE Mx INC.	PERF. & HAND. QUAL. EFFECTS
MLDCS 1969 - 70	REDUCE MAX UPBENDING (Mx') @ DESIGN LOAD FACTOR	EXISTING SAS & CONTROL SYS. PLUS NEW COMPUTER & ACCEL'S	≈ -9%	-	-	NONE
PLDCS 1970 - 72	SAME AS MLDCS PLUS REDUCED FATIGUE MEAN Mx'	EXISTING TRIM SYSTEM - INTERIM NEW CONTROL BOX & FOLLOW UP LINK AILERONS OPEN UP $+i_T$ STOP	≈ -9%	-10 TO -30%	–	T.O. CLIMB & CRUISE DRAG PENALTY NO F.Q. EFFECTS
ALDCS 1973 - 74	SAME AS PLDCS PLUS REDUCED FATIGUE INCREMENTAL	EXISTING SAS A/P & CONTROLS NEW COMPUTER & ACCEL'S	≈ -9%	-10% TO -30%	-30 TO -50%	SAME AS PLDCS

Figure 5 - Comparison of C-5A LDCS Systems

A comparison of the effects of each of the three systems on wing root loads is shown by Figure 6. The flight condition selected for this illustration was chosen to depict the initial objective of reducing maximum upbending moment by approximately 10% (actually attained about 9% due to bending torsion trade-off effects). The reduction in the 1.0g bending moment is about 25% for the PLDCS and ALDCS while the incremental bending moment is reduced approximately 40% by ALDCS for this condition. Similar load reductions exist for other flight conditions.

WING ROOT Mx' - My' ENVELOPE



LINE	SYSTEM	MAX. Mx' 10^6 IN-LBS	MEAN Mx' 10^6 IN-LBS	VARIABLE Mx'/nz 10^6 IN-LBS
1	BASELINE	212	85.2	84.4
2	MLDCS	194	85.2	84.4/67.8
3	PLDCS	194	64.2	84.4
4	ALDCS	194	64.2	50.9

FLIGHT CONDITION

CLEAN CONFIG.
265 KNOTS @ S.L.
728000 LBS. GW.

Figure 6 - Comparison of LDCS Loads

ACTIVE LIFT DISTRIBUTION CONTROL SYSTEM DEVELOPMENT

The ALDCS development process involved a number of engineering disciplines to integrate the design requirements of loads and structural dynamics, stability and handling qualities and existing flight control systems. A flow chart of the design tasks is shown in Figure 7. Figure 8 provides a simplified interface diagram indicating the integration of the ALDCS computer with the existing C-5 flight control subsystems.

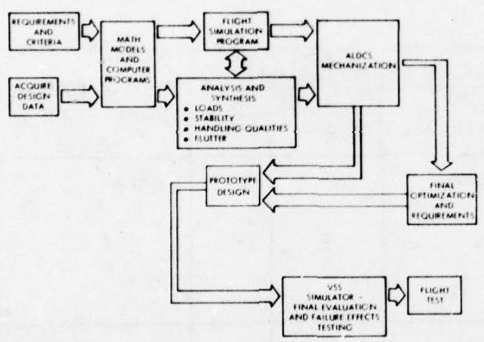


Figure 7 - ALDCS Development Tasks

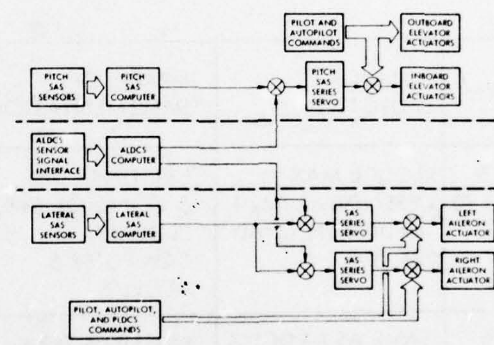


Figure 8 - ALDCS Systems Interfaces

The task of acquiring the necessary design data was simplified by the existence of airplane math model data, flight control subsystem mechanizations, and flight test response correlation data from the basic C-5A design program. The major void in design data was the lack of aileron and elevator characteristics information at frequencies beyond their design points. The actuators were designed and tested primarily for handling qualities and automatic stabilization of aircraft low frequency short period and dutch roll modes, whereas the ALDCS would require the sensing and active control of higher frequency aeroelastic mode dynamics. These missing actuator data not only included frequency response but hysterisis, surface rates and tolerance bands in unloaded and loaded conditions. These data were obtained by tests on the C-5 Vehicle Systems Simulator using both new and worn servo actuators, by tests performed by the servo actuator manufacturer, and by frequency response flight tests.

System Mechanization - The dual channel redundancy design ALDCS computer provides signals to both the lateral augmentation series servo to control the aileron actuators symmetrically and the pitch augmentation series servo to actuate the inboard elevator control surfaces. Aileron actuators also receive commands from the pilots, autopilot, and passive LDCS. The pilots and autopilot command inboard as well as outboard elevators. Figure 9 shows the locations of the ALDCS sensors and interfacing computers and affected control surfaces. The wing mounted accelerometers are the only additional sensors required for ALDCS integration.

The ALDCS mechanization consists of an array of sensors, gains, and filters. Figure 10 is a block diagram of the ALDCS simplified mechanization to be used as a roadmap during the ensuing discussion. The aileron and elevator channels will be discussed separately.

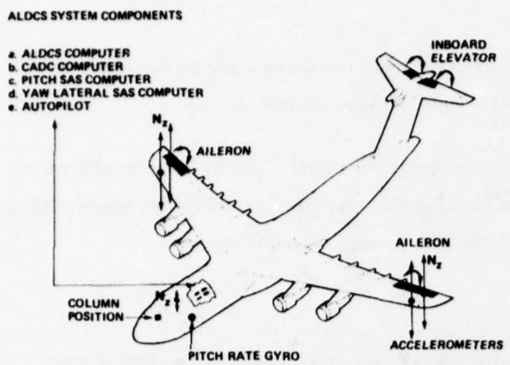


Figure 9 - ALDCS System Components

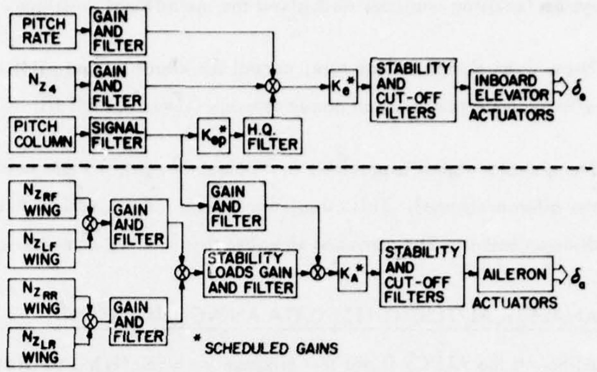


Figure 10 - ALDCS System Block Diagram

Aileron Channel - The aileron control channel commands the right and left ailerons symmetrically to accomplish the maneuver load relief function. The feedback sensors utilized for the aileron channel are provided by two vertical accelerometer locations per wing, one located on the forward main beam (W.S. 1186) and the other on the rear beam (W.S. 1152) both at an outer wing location. The signals from these accelerometers are averaged and compensated by smoothing filters that attenuate sensor noise and aid in the elimination of higher frequency wing vibration modes beyond the ALDCS control bandwidth.

The Stability and Load Control Gain and Filtering portion of the aileron channel provides the necessary compensation to adequately phase the feedback accelerometer signals for control of the inner wing bending moments and to attain the design goal stability margins.

A pilot's feedforward command, acquired from the existing C-5 elevator cable position (ECP) transducer, is summed with the compensated acceleration control signal to provide abrupt maneuver load control. The feedforward signal is filtered for proper abrupt load alleviation aileron command phase.

These control signals are then gain scheduled by aircraft dynamic pressure from the Central Air Data Computer (CADC) to provide proper stability and load relief schedules and to minimize handling qualities degradations throughout the aircraft speed envelope. Cut-off filters are provided to preclude adverse coupling with higher frequency uncontrolled modes. The ALDCS aileron command signal is controlled by boundary control logic which contains the circuitry to disengage the signal when exceeding flight boundaries where the ALDCS is not required. These operational boundary conditions are when Stallimiter subsystem is activated, the airplane exceeds maximum horizontal airspeed/Mach (350 KCAS/M = 0.825), and when the airplane load factor exceeds 1.9 g's. These logic control signals are obtained from existing aircraft subsystems with the exception of load factor. This signal is derived from ALDCS wing and fuselage accelerometers to closely represent aircraft C.G. acceleration. The system is automatically re-engaged as the aircraft re-enters the ALDCS operational envelope. The aileron command signal is then limited and interfaced with the lateral SAS aileron series servoactuators.

Elevator Channel - The elevator channel contains three sensors, two feedback parameters and one feedforward command. Airplane pitch rate, as provided by the pitch SAS rate gyro, is utilized to augment the airplane short period damping and thereby alleviate the excitation of short period induced gust loads and to restore the handling qualities degraded by the aileron pitching moment effects.

An existing C-5 autopilot subsystem vertical accelerometer mounted in the forward fuselage provides additional gust load control and compensates the airplane pitch response characteristics.

A feedforward signal, pilot's elevator input demand, is required to restore the airplane maneuverability and accelerated stability (stick force per 'g') characteristics that are significantly degraded by the load control signals. This signal is scheduled as a function of airplane dynamic pressure and compensated by a command model filter to provide the proper system handling qualities throughout the operational envelope.

These three signals, pitch rate, normal acceleration and pilot elevator command input are summed and again scheduled with dynamic pressure and passed through system cut-off filters for stability and gust load control phasing.

The elevator signal is provided to a boundary control logic network that disengages the signal under the same conditions as the aileron channel. This circuit includes a fade-out filter to minimize acceleration transients resulting from abrupt surface disengagement. The command signal is then limited and interfaced with the pitch augmentation subsystem.

ANALYTICAL/FLIGHT TEST DATA AND COMPARISONS

Although the ALDCS flight test program included flying qualities, stability and control, failure effects, and other non-structural testing; this paper will discuss only the structural loads aspects of the program. The following sub-sections provide representative samples of typical analytical/test comparisons.

Structural Response to Force Inputs - Symmetric aileron and elevator frequency sweeps were performed during the test program to establish system stability and frequency response characteristics. Figure 11 illustrates the dramatic reduction which the ALDCS makes in the wing first bending frequency response. Test runs were made for both frequency sweeps and for constant frequency inputs primarily in order to establish well defined resonance curves. The ALDCS was designed for minimum response at frequencies above about 1 Hz. This was due to: 1. The desire to use existing servos and actuators, which have a rapid roll-off at frequencies above 1 Hz. and 2. Analytical results which show little to be gained, in the way of gust load alleviation, at higher frequencies.

Incremental Maneuver Loads - Significant reductions in wing incremental maneuver bending moments result from the aileron modification of the spanwise additional airload distribution. Figure 12 shows both analytical and flight test measured incremental wing root bending moment ($\Delta M'x/g$) for ALDCS OFF and ALDCS ON conditions. These data illustrate the attainment of a design goal of at least a 30% reduction in wing root incremental maneuver bending moments. The test points cover a Mach number range of 0.54 to 0.78 and represent normal operating airspeeds for the test altitudes.

The use of ailerons to provide maneuver incremental bending moment reductions results in significant increases in positive torsion. No criteria were established for torsion load magnitudes since attainment of the design goal of a 30% reduction in bending moment was considered of prime importance and the aileron required to produce that reduction automatically produces an increased torsion moment. Figure 13 shows incremental wing root torsion ($\Delta M'y/g$) for ALDCS OFF and ON. The test measured torsion loads show generally less torsion increase due to ALDCS then the analytical conditions. In general, wing root incremental torsion increases due to ALDCS are approximately 30 percent for maneuvering flight conditions.

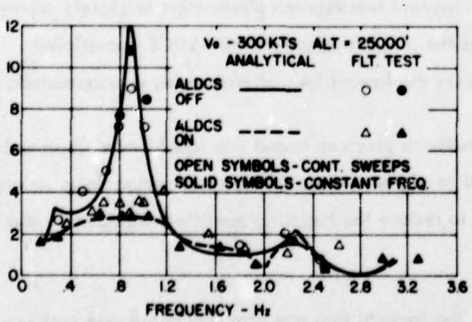


Figure 11 - Wing Root Bending Response

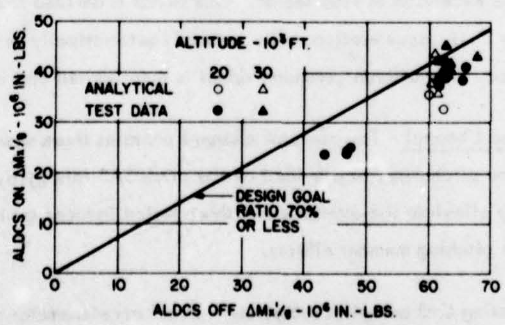


Figure 12 - Maneuver Loads - W.S. 198 Bending

The effect of ALDCS on the spanwise distribution of incremental maneuver bending moment is illustrated by Figure 14. This condition is chosen to illustrate spanwise loading effects since the Mach number of 0.78 is typical of normal cruise conditions although a test altitude of 30,000 feet is slightly below cruise altitude for the test condition gross weight. The other test condition spanwise incremental bending moment comparisons are similar and show equally good correlation.

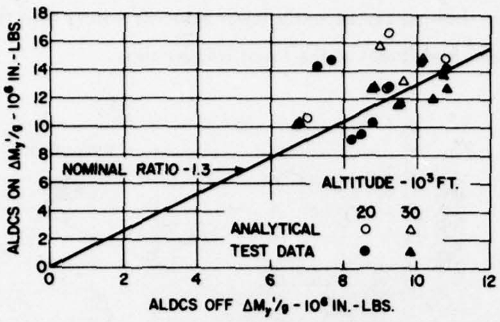


Figure 13 - Maneuver Loads - W.S. 198 Torsion

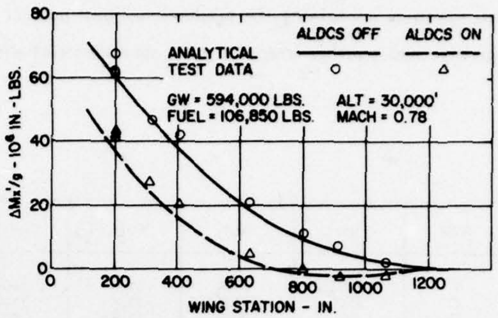


Figure 14 - Maneuver Loads - Spanwise Bending

Gust Loads - Load reductions during continuous turbulence are achieved both from the ALDCS aileron response and from the reduction in rigid body and short period airplane response due to the inboard elevator pitch damping action. The design goals for gust load alleviation, simply stated, are at least a 30 percent reduction in RMS root bending moment response with no more than a 5 percent increase in RMS root torsion response. The design of the ALDCS aileron and inboard elevator input signals and gains/filtering provided the desired balance between aileron generated direct lift modifications and elevator controlled angle of attack response reductions.

Summary plots of wing root bending moment and torsion response are shown in Figures 15 and 16. Normalized responses σRMS loads divided by RMS gust velocity - for several test conditions of varying airspeed, altitude, and mass configuration are included in these summaries. Table 1 identifies the various parameters for each of the test conditions. Multiple symbols on the summary figures denote multiple test runs.

These comparisons are based on symmetric response only since any unsymmetrical responses are common to both ALDCS OFF and ALDCS ON configurations. The analytical $\bar{A}v$ values result when the analytical model is subjected to the test derived vertical gust velocity profiles and normalized by the RMS vertical gust velocities. The test $\bar{A}_{HV}$ values are the correlated test responses rather than the total test responses, i.e., the responses which are coherent with the measured normalized vertical gust inputs.

Figure 15 illustrates good analytical/test agreement and clearly shows the significant reductions (30 to 50 percent) in wing root bending moment response with ALDCS ON. Figure 16 shows generally good analytical/test agreement for root torsion and illustrates the relatively small change in gust induced torsion loads with the ALDCS ON.

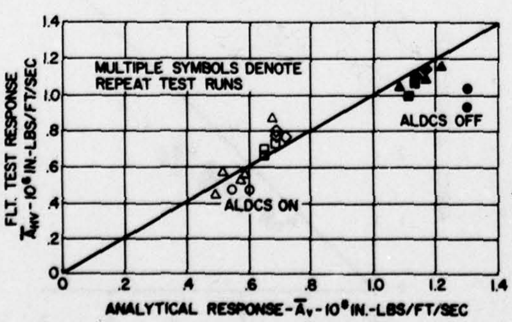


Figure 15 - Normalized Gust Response - W.S. 198 Bending

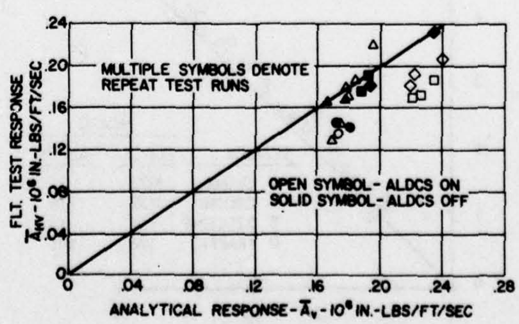


Figure 16 - Normalized Gust Response - W.S. 198 Torsion

A spanwise bending moment analytical/test comparison with ALDCS ON is shown in Figure 17. The results of five runs for one set of test parameters are shown and illustrate the typical scatter inherent in dynamic response test program results. Three of the five runs show excellent agreement between test derived $\bar{A}_{HV}$ and analytical $\bar{A}v$ bending moments at each of the seven instrumented wing stations. The one run showing significantly greater test response and poorer agreement had a measured input gust spectrum which was more irregular than the other runs, particularly at low frequencies - the region of the gust spectrum containing the predominant gust power. The run showing significantly lower response values than the average also had a rather irregular gust input spectrum with reduced amplitudes at the lower frequencies.

SYM	Ve	ALT	FUEL	CARGO
○	235	5000	110K	100K
△	250	5000	230K	110K
□	325	7000	100K	90K
◇	325	7000	250K	110K
◇	325	15000	80K	90K

Table 1 - Gust Response Data Test Conditions

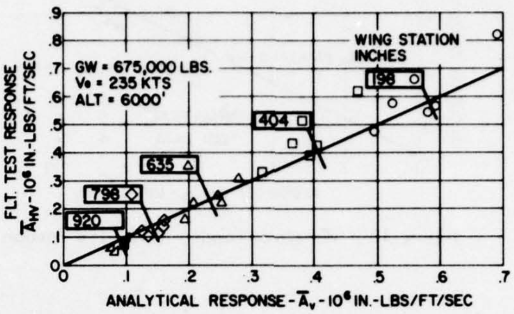


Figure 17 - Normalized Gust Response - Spanwise Bending

SERVICE LOADS RECORDING PROGRAM DATA

The C-5A Aircraft Structural Integrity Program included a Service Loads Recording Program (SLRP) whose basic purpose was to obtain recorded operational data to define the loads environment encountered in normal Military Airlift Command operations. Thirteen of the aircraft included wing strain gage instrumentation to measure axial stresses in addition to the normal aircraft motion and control surface deflection parameters. Data were obtained in both the baseline and the ALDCS aircraft configurations, since the SLRP began prior to ALDCS installation, thus average operational statistics are available for comparison purposes.

The flight recorded aircraft response time history data were separated into maneuver and gust sources using a response rate of change and time duration criteria. The maneuver source was sub-divided into climb, cruise, descent and traffic segments while the gust data were categorized by altitude bands.

The following SLRP data comparisons illustrate the effectiveness of the ALDCS in producing reduced incremental wing responses during maneuvering flight, continuous turbulence, and aerial refueling operations.

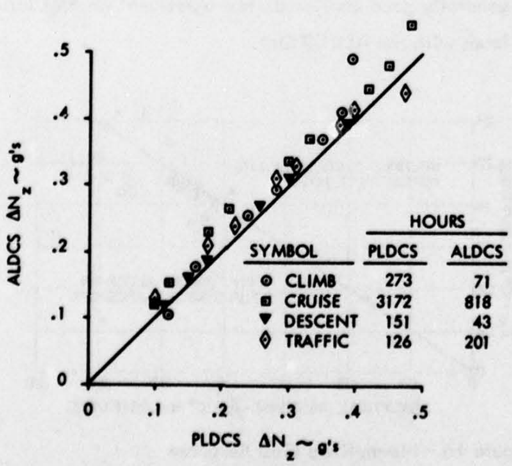


Figure 18 - Maneuver Load Factor Comparison

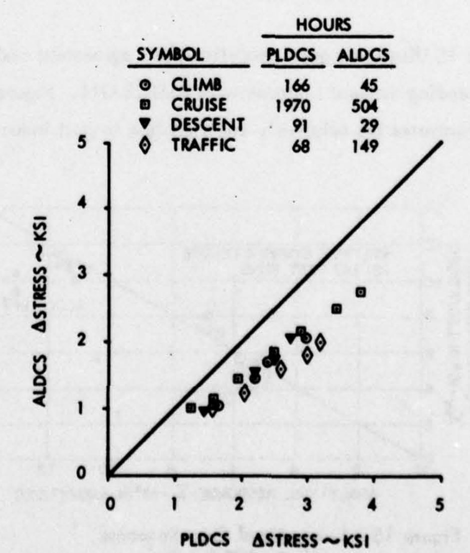


Figure 19 - Maneuver Axial Stress Comparison

Maneuvering - Figure 18 shows the correlation of incremental maneuver load factors by flight segment. This figure was constructed by plotting incremental load factors at equal exceedance values from the individual load factor occurrence curves. Figure 19 presents the same type of information for wing stress at one of the inner wing strain gage stations. The significant reduction in incremental stress (approximately 30 to 40 percent) for approximately the same load factor exceedances is apparent. Similar results were obtained at the other wing stress instrumentation stations.

Gust - Figures 20 and 21 present data in the same general format as the maneuver data except by altitude band rather than flight segment. Here again the incremental load factors show generally good correlation whereas the incremental stresses are reduced on the order of 30 percent by the ALDCS. As with the maneuver data, similar results exists at the other wing stress instrumentation stations.

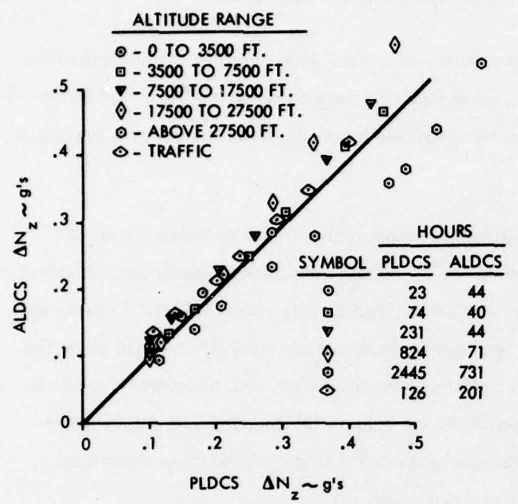


Figure 20 - Gust Load Factor Comparison

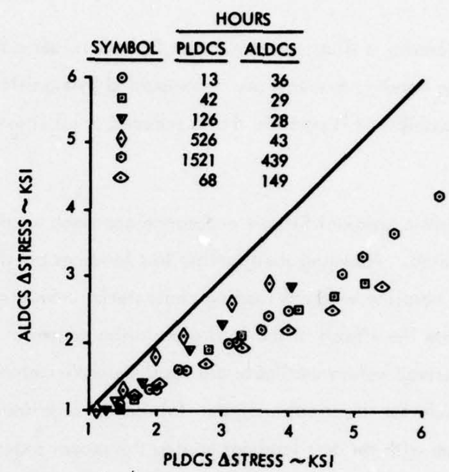


Figure 21 - Gust Axial Stress Comparison

Aerial Refueling - The increased emphasis on aerial refueling operations resulted in a special test program, using two of the SLRP aircraft with additional wing stress measuring instrumentation, to obtain a statistically stable sample of operational aerial refueling aircraft response data. Figure 22 shows that aerial refueling with ALDCS ON results in an approximate 20 percent reduction in the load factor spectra. Figure 23 shows that similar reductions in inner wing incremental axial stresses occur however the outer wing stress spectra are not significantly affected by ALDCS. Since the principal forcing function affecting the outer wing is the asymmetric aileron associated with station keeping, the symmetric only ALDCS provides no significant load alleviation effect on the outer wing.

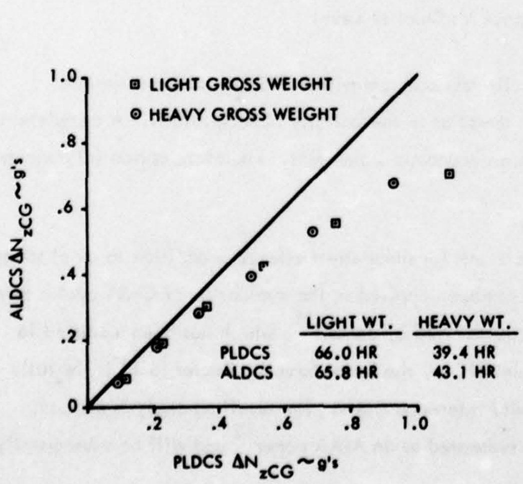


Figure 22 - Aerial Refueling Load Factor Comparison

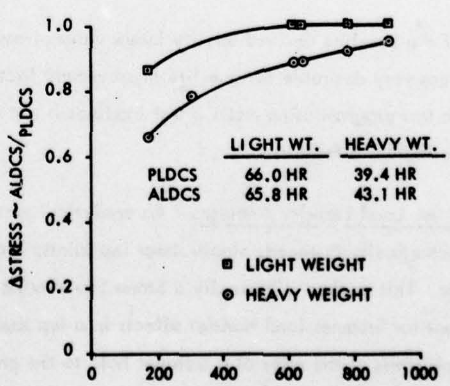


Figure 23 - Aerial Refueling Axial Stress Comparison

FATIGUE ENDURANCE CONSIDERATIONS

The assessment of the effectiveness of any load alleviation system, with respect to increased fatigue endurance, requires an analysis methodology which addresses the effects of the total loading spectrum as affected by the load alleviation system. Since the C-5A load alleviation system was developed as an "add on" to an already existing structure which has well defined loading characteristics, test substantiated stress to load relationships, and cyclic test derived quality index values (K); the question of what to do about changes in shear stress as well as axial stress must be addressed. This is particularly true when using the classic approach to fatigue analysis, i.e., use of Miner's rule and constant amplitude S-N data in a system which deals with a singularity stress state.

External Loading Characteristics - A significant change in external loading characteristics with ALDCS results in combined stress spectra which are significantly different than that of the baseline analyses and test data base.

The dilemma is illustrated by Figure 24. Consider a load alleviation system that is configured such that large reductions in wing bending moments are accompanied by significant increases in wing torsion. Depending on the bending/torsion relationships, it is possible that a reduced axial stress spectra will be accompanied by an increased shear stress spectra as shown.

The classic uniaxial fatigue endurance approach would produce two different endurance vs. quality index curves as illustrated. Assuming the baseline test loadings consisted of appropriate shear, bending and torsion loads; the net effect of the baseline external loading characteristics would be included in the test derived quality index values. In order to evaluate the effects of the load alleviation system in this example, new quality index values must be obtained since the test derived values available are for different external loading characteristics. At this point, one of several choices is available for structural analysis: (a) use the baseline test derived quality index values, (b) conduct a new cyclic test program with the test loadings having the proper external loading characteristics or (c) find or develop an analytical method which accounts for the changes in the shear as well as the axial stress spectra.

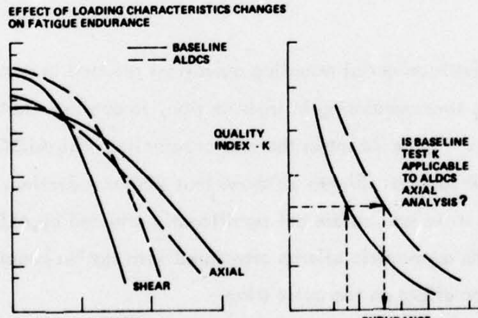


Figure 24 - Analytical/Test Endurance Vs Quality Level

Use of the baseline derived quality index values from the C-5A cyclic test program with ALDCS axial stress spectra produces very desirable fatigue life improvement factors but leaves doubt as to the validity of such results. A complete new cyclic test program using ALDCS test loadings is not practical from an economic standpoint. Therefore option (c) appears to be the logical alternative.

Axial vs. Load Transfer Analyses - An analytical method; which accounts for shear stress effects in addition to axial stresses in mechanically fastened, single shear lap joints; is available and has been applied in the evaluation of C-5A cyclic test results. This method is basically a Stress Severity Factor Method, as reported by Jarfall[11], which has been modified to account for fastener load transfer effects in a lap shear joint. By definition, the Stress Severity Factor (SSF) is the ratio of peak stress at the edge of a fastener hole to the gross area uniaxial reference stress. The resulting analysis method, including correlation of analyses to test data, has been previously presented as an AIAA paper[12] and will be subsequently referred to as a load transfer analysis.

Application of the load transfer analysis to a beam cap to web splice and to a lower surface spanwise splice on the C-5A wing illustrates the significance of changes in external load characteristics on apparent peak stresses at the edge of fastener holes.

Figures 25 and 26 show the variation of axial and peak stresses with maneuver load factor for the aforementioned representative structural locations. External loads for the baseline airplane configuration and the PLDCS and ALDCS configurations for a representative speed/altitude - mass configuration maneuver load condition were analyzed using axial and load transfer methods. These analyses used test derived stress to load ratios, baseline configuration quality index values (K's) and necessary analytical structural parameters such as fastener tilt factors, hole bearing stress concentration factors, etc. The analysis results for the beam cap to web location (Figure 25) provide a graphic illustration of the potential inadequacy of relying on an axial stress analysis using test derived data from one aerodynamic configuration to evaluate the effects of changes in aerodynamic configuration (external load characteristics). This particular structural location shows no apparent improvement (reduction) in stress level due to the ALDCS when using the load transfer analysis for this particular load source (maneuver). The implication here is simply that the axial stress reduction due to reduced bending moment is obviated by the increase in shear stress due to amplified torsion loads. This apparent equal trade-off between reduced bending moment and increased torsion will vary with other structural locations (Figure 26) and with specific mission segments (variations in cargo weight, fuel weight, airspeed and altitude) and load sources (maneuver, gust, etc.).

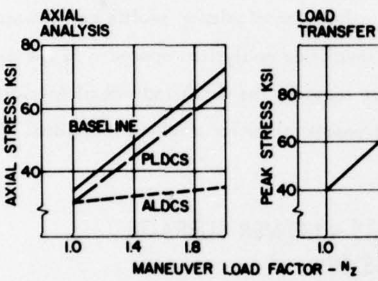


Figure 25 - Analytical Maneuver Stresses - Cap/Web Splice

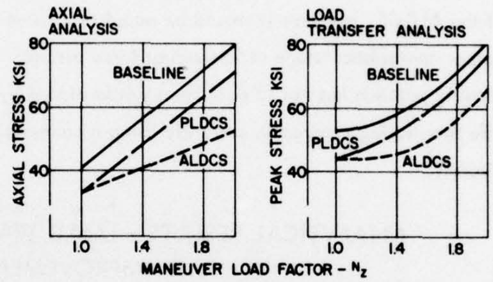


Figure 26 - Analytical Maneuver Stresses - Spanwise Splice

Gust Loads/Stress Phasing Problem - Since the load transfer method requires discrete relationships between axial, shear and bearing stresses, the external loads definition also requires discrete relationships. This poses no problem for maneuvering flight, landing impact, etc., however; those load sources which are dependent on Power Spectral Density (PSD) methods for describing the statistical variations of loads or stresses present a problem in the use of the load transfer analysis method.

The continuous turbulence (gust load source) PSD analysis results in independent axial stress and shear stress spectra due to the very nature of the statistical basis for their derivation - the individual spectra being a function of the RMS responses (σ) and the characteristic frequencies (N_o). The PSD methods provide the means of calculating correlation coefficients between various load components, however, discrete phasing relationships do not exist within the methodology. In short, the present PSD loads analyses and the Load Transfer Stress Analysis Methods are not compatible.

Gust Loads/Stress Time History Analysis - In an attempt to define the possible net analytical effects of gust loads/stress phasing relationships on peak stresses, the load transfer procedure is coupled with a loads/stress time history solution, as illustrated by Figure 27. Analytical and test measured shear, bending moment and torsion load time histories are used to generate axial and shear stress time histories at selected structural locations. Then, discrete time points are analyzed using the load transfer equation to generate peak stress time histories. Phasing factors are then determined by assessing the axial and shear phase relationships that exist at peak stress maximum and minimum amplitudes. These phasing factors are simple percent in phase or out of phase relationships - phase lag not being considered in further analysis. The axial and shear stress spectra are then simply related by these phasing factors. A detailed discussion of this procedure is contained in reference 13.

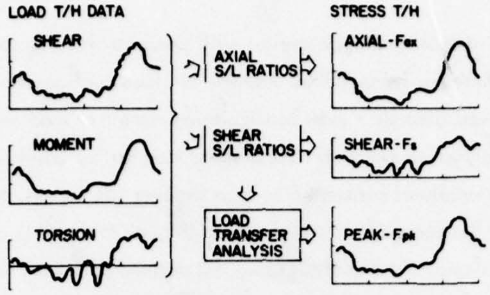


Figure 27 - Stress Time History Generation

ALDCS LIFE IMPROVEMENT FACTORS

The previously discussed analysis methods have been utilized to calculate fatigue endurances for selected structural locations on the C-5A wing for both PLDCS and ALDCS aircraft configurations. A discrete set of mission profiles, based on actual force usage, was utilized in the analysis. Figure 28 presents the ratios of these analytical fatigue endurances in the form of life improvement factors. These results clearly indicate a significant improvement in endurance as a result of the ALDCS, however it should be noted that these factors will vary as a function of mission profile definitions and/or actual operational usage of individual force aircraft. Additionally, the simplified analytical approach to the axial/shear phasing problem has not to date been substantiated by test data. For these reasons, the C-5A Individual Aircraft Service Life Monitoring Program is currently using a conservative 1.25 life improvement factor for tracking individual force aircraft.

ANALYTICAL RESULTS - LOAD TRANSFER ANALYSIS - LOWER SURFACE
LIFE IMPROVEMENT FACTORS - SS/WR

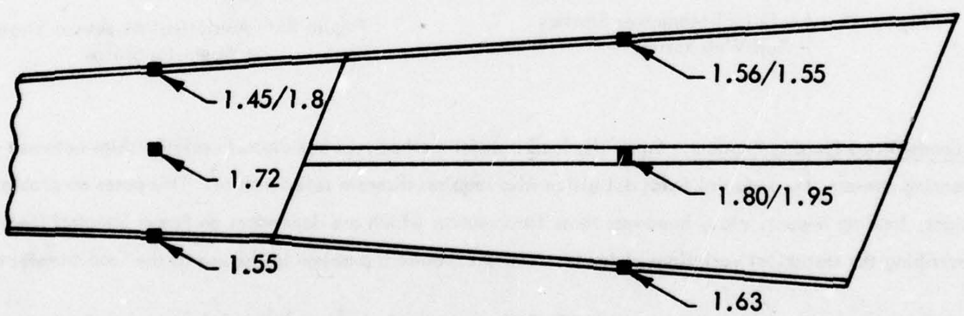


Figure 28 - ALDCS Life Improvement Factors

OPERATIONAL HISTORY

The C-5 ALDCS was introduced into operational service in July of 1975. In the course of the next 14 months the entire force of C-5's had ALDCS incorporated. Although the system was used extensively for the first year it was not operationally required for dispatch until after the entire force was updated. Suggestions were submitted to provide operational briefings on the system at each C-5 base, however, since flight test had indicated that there were no discernable differences in aircraft handling qualities with ALDCS "on" or "off", no such briefings were provided.

As subsequent events indicated, it was a mistake not to provide these operational briefings. Since most flight crews did not understand the system and did not know how it operated, there was some apprehension about its operation and in many cases they were afraid to use it, particularly in critical operations such as inflight refueling. As a result, early operational reports from the system were mixed and ranged from pilot comments that the system provided improved ride during turbulence with less wing flexing all the way to generally degraded handling qualities. It is evident in retrospect that even though flight test revealed no evidence of handling quality degradation and even though the flight test was quite extensive, it was not all encompassing, such that differences between aircraft control systems, system tolerances, operational utilization, and pilot characteristics did result in some change in pilot feel of the aircraft. In particular, there were complaints of speed stability, difficulty in holding bank angle, tendency toward uninitiated pitch inputs during a precision maneuver and a number of other adverse comments. These criticisms were difficult to quantify and assess in that they were different from airplane to airplane, from pilot to pilot and from base to base. However, in the summer of 1976 special limited instrumentation, utilizing the C-5 MADAR systems, was installed on four aircraft. This was accomplished at Travis Air Force Base and a controlled test program was conducted in which specified precision maneuvers such as inflight refueling were conducted with 14 different pilots. During the test over 90 AR hook-ups were made in which ALDCS was turned on and off without pilot knowledge. The results of these tests were that there was unanimous consent with the entire pilot sample that ALDCS was preferred "on" during all precision maneuvers. Although there were a few actions taken to provide improved maintenance inthe field, there were no design changes made to ALDCS.

Following completion of the controlled tests and the submittal of the test report, there were no further operational complaints from pilots on the use of ALDCS. It is now required for dispatch and used from take-off to landing on all flights.

System reliability was initially predicted to be 3,000 operational hours. There were some early operational reliability problems which were resolved by a minor field update of the computer electronics. Present mean time between unscheduled removals is approximately 1000 hours. In flight system failures have resulted in system deactivation, as designed, with only minor transient responses, as substantiated during the flight failure effects testing. There has been only one failure of a wing accelerometer to date although a minor electrical connector corrosion problem, since resolved, occurred on several aircraft.

CONCLUSIONS

The C-5 ALDCS program has demonstrated the practicality of using existing flight control surfaces and systems to affect specific changes in structural load distributions and magnitudes for load alleviation purposes.

This work illustrates an application of active control technology to the solution of a particular problem on an existing aircraft. Application of the same engineering principles during the design stage of new aircraft could have significant effects on the overall "design compromise".

The analytical and measured data generated during the C-5A ALDCS development/test program and the Service Loads Recording program provides a basis for the following conclusions:

1) The maneuver and gust load test data substantiate the analytical load methods and provide a solid base from which to develop ALDCS life improvement estimates.
2) Uniaxial stress/fatigue analysis methods may result in overoptimistic predictions of fatigue endurance improvement due to load alleviation systems as a result of favorable changes in axial stress, but not accounting for significant changes in shear stresses.
3) The load-transfer method provides an analytical means of accounting for changes in shear as well as axial stresses.
4) An axial, shear peak stress time history analysis provides a basis for establishing axial/shear stress phasing factors.

5) Significant improvements in analytical fatigue endurance can be realized with the application of existing active control load alleviation technology.

6) Incorporation of Active Control systems on operational force aircraft is a practical means of providing maneuver and gust loads alleviation.

7) Flight crew acceptance of active controls would be enhanced by thorough user indoctrination of system functions and operational characteristics.

Acknowledgements

The author wishes to express his appreciation to Mr. William J. Hargrove of the Lockheed-Georgia Stability and Flight Controls Department and Mr. Larry A. Adkins (ALDCS Program Manager) of the Electronics/Controls Department for their contributions to this Chapter.

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B-1 RIDE CONTROL

by

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SUMMARY

The B-1 aircraft is one of the first aircraft to include a control configured vehicle (CCV) concept ride control, in the early design phases. A substantial savings in weight was achieved with this approach as compared to direct material stiffening. This paper discusses the design development, including system requirements and mechanization details. The design implementation is also discussed, including hardware and installation details. Finally, flight test performance evaluations, comparisons of analytical and test data, system improvements, and flight crew evaluations are presented. While the detailed information is provided for a system designed to improve ride quality through control of structural motion, it is concluded that the technology discussed is applicable to load relief and even flutter suppression of flexible vehicles, military or commercial.

LIST OF SYMBOLS

$\bar{A}_{(\ )}$	Root-mean-square acceleration due to 0.3048 m/sec (1 ft/sec) rms gust intensity; subscript Z denotes vertical axis, Y denotes lateral axis
AR	Aspect ratio
CG	Center of gravity
℄	Centerline
$\bar{c}$	Mean aerodynamic chord
δ, δ_{cv}	Vane deflection
EI	Bending stiffness
FS	Fuselage station
g	Gravity constant
$\bar{H}_{(\ )}$	Crew sensitivity index; subscript Z denotes vertical axis, Y denotes lateral axis
h_{AGL}	Altitude above ground level
h_p	Pressure altitude
K	System gain
λ	Taper ratio
Λ_V	Sweepback angle of leading edge of vane
Λ_W	Sweepback angle of leading edge of wing
M	Mach number
M_L	Mold line
MAC	Mean aerodynamic chord
$n_{(\ )}$	Load factor; subscript Z indicates vertical axis, subscript Y denotes lateral axis
ω	Frequency, rad/sec
Ω	Wave number, $\left(\frac{\omega}{V_o}\right)$
S	Laplace operator
SCAS	Stability and control augmentation system
$\sigma_{\alpha V}$	Gust intensity estimated from angle-of-attack vane readings
$\sigma_{\beta V}$	Gust intensity estimated from sideslip angle vane readings
σ_{w_g}, σ_{v_g}	Vertical and lateral gust intensity, respectively
T_D	Human response weighting function
Φ_{gust}	Gust power spectral density
$\Phi_{\bar{H}_{(\ )}}$	Crew sensitivity power spectral density; subscript Z and Y denote vertical and lateral, respectively
Φ_{n_z}, Φ_{n_y}	Power spectral density of vertical and lateral load factors, respectively.
WP	Water plane
Z_F, Y_F	Vertical and lateral fuselage coordinates, respectively

INTRODUCTION

The B-1 is one of the first vehicles to include a control configured vehicle (CCV) concept in the early design phases. The aircraft has a requirement to provide a specified level of ride quality for the crew. This requirement has been met on the B-1 through the use of an automatic control system called a structural mode control system (SMCS) whose main external feature is a set of vanes near the crew station which are canted down 30° from the horizontal, as shown in figure 1. A substantial savings in weight was achieved with this approach as compared to direct material stiffening. The details of system requirements had to be determined from a production (long-life) point of view, which has not been done before for a system of this type. Extensive wind tunnel tests of the vane characteristics were conducted. Analytical models of the flexible aircraft and control systems were developed to analyze requirements and to investigate stability and performance. Component parts were tested to the requirements in the laboratory. Flight test of the SMCS is continuing, and comparisons with analytical predictions are being made.

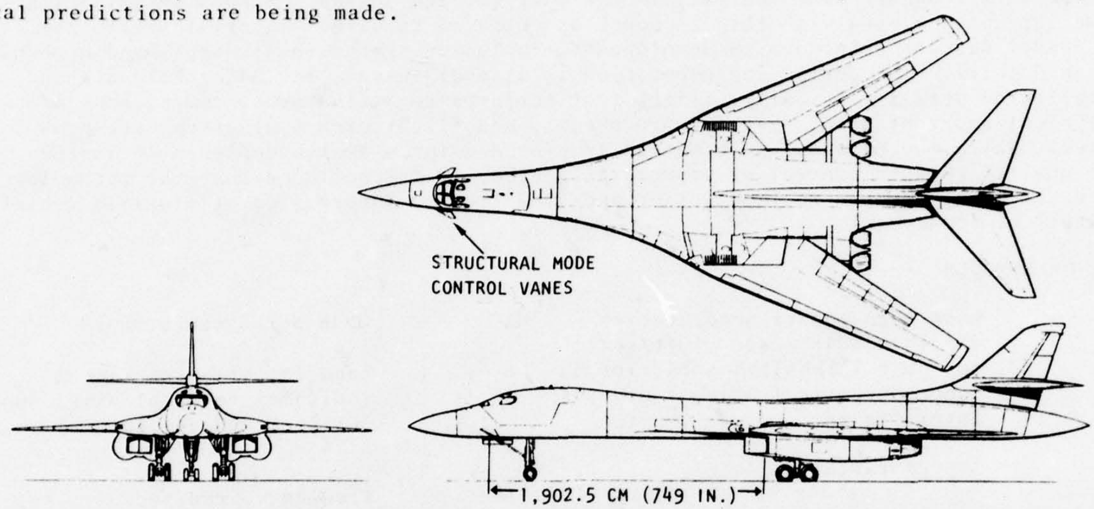


Figure 1. B-1 Aircraft with Wings Swept Aft

The overall objective of this paper is to describe the conceptual design, development, and flight tests of the B-1 SMCS and its impact on ride quality. Since the B-1 is the first aircraft to have a system such as the SMCS designed for production and long service use, it is expected that this information will add to the technology base for the design of future large military or civil aircraft.

SMCS RATIONALE

One of the principal missions of the B-1 involves flying for long periods of time in close proximity to the terrain. B-1 design requirements have produced a relatively flexible aircraft. This vehicle flexibility combined with the ever-present low-altitude atmospheric turbulence can produce an acceleration environment at the crew station which can degrade handling qualities and general crew efficiency with a consequent degradation of mission success. Reference 1 reviews this ride quality problem and offers design criteria which, when complied with, tend to alleviate the problems described. These criteria have been formally included in the B-1 design specifications.

The B-1 (figure 1) employs a variable sweep wing which is swept aft when flying the low-altitude mission. The wing is swept primarily to improve the vehicle drag characteristics; however, this is fortunately favorable to improving the vehicle ride qualities also. The aft-swept wing has a low lift curve slope and thus is less susceptible to turbulence-induced angles of attack and the consequent excitation loads. Despite sweeping the wing, the level of turbulence excitation susceptibility on the flexible B-1 was still too high to meet the ride quality requirements. Two basic design choices remained in order to comply: (1) add material (and weight) to stiffen the structure over that needed for strength and flutter requirements, or (2) use automatic control systems. A choice was made in favor of the latter approach because of a potential savings in weight and because of the existing depth of analytical and flight test experience available (references 2 through 6) on these types of systems.

The SMCS has been designed to the fail-safe rather than the fail-operate, fail-safe concept. This approach has been taken because the system is intended strictly for improving ride quality; thus the B-1 has full structural integrity with or without the SMCS operating. Should the system fail for any reason, the vanes will be centered and held and the mission continued, admittedly at a worsened level of ride quality.

Ride Quality Criteria

The SMCS performance ability to improve ride quality is evaluated against a parameter called the crew sensitivity index, $\bar{H}$. The parameter $\bar{H}_z$ is associated with the vertical motion, and the $\bar{H}_y$ is associated with lateral motion. The development of the $\bar{H}$ concept of ride quality evaluation can be found in reference 1.

In order to illustrate the component considerations entering the definition of $\bar{H}$, the vertical parameter $\bar{H}_z$ will be examined. *The three main components of $\bar{H}_z$ are shown* in figure 2. The gust power spectral density shown is a measure of excitation energy in the atmosphere as a function of the wave number, Ω; once the speed of the aircraft is defined, the wave number can be viewed as a frequency parameter ($\Omega = \omega/V_o$).

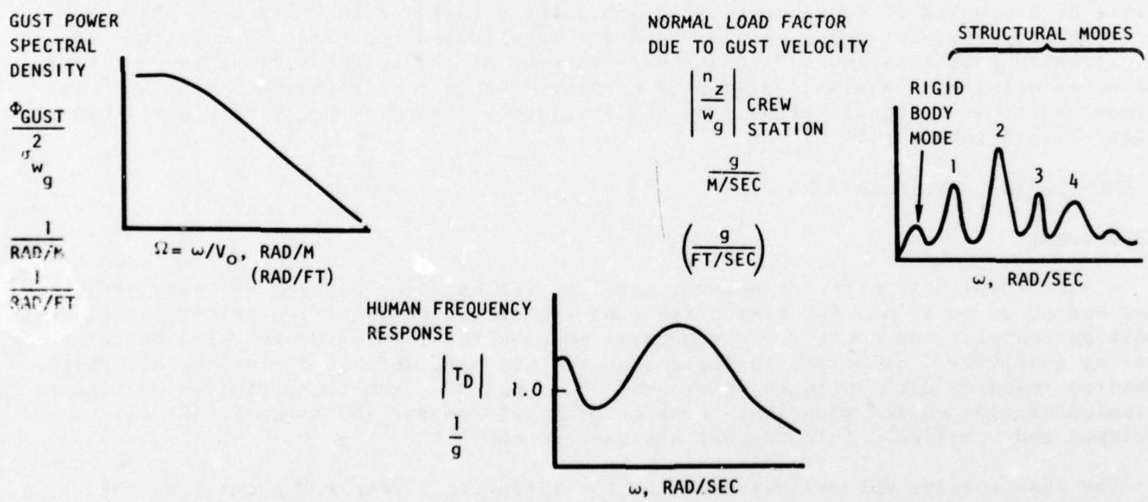
The second curve is a typical flexible airplane normal load factor due to a unit vertical gust velocity frequency response plot for the crew station. Correlating against the gust power spectral density curve, it can be seen that the energy in the atmosphere can excite the rigid-body (whole-vehicle) motion and a number of the lower frequency structural modes.

The third curve may be viewed as a weighting of the response motion at various frequencies, depending on the dynamic response characteristics of the human body and how the human feels about them. As shown at the bottom of the figure, all of these data are brought together in what is a weighted root mean square (rms) normal load factor due to a unit gust intensity. If T_D were left out of the calculation, the rms load factor, $\bar{A}$, would be obtained.

The lateral parameter $\bar{H}_y$ is developed similarly with the gust spectrum remaining the same, but with lateral load factor frequency response and the human response function reflecting different characteristics.

The levels of $\bar{H}_z$ and $\bar{H}_y$ accepted as design criteria are influenced by a number of factors, including mission time; these are discussed and evaluated against response characteristics of a number of typical military aircraft in reference 1.

The only element of the parameter that can be changed by a control system is the contribution of the structural response to the normal and lateral load factors.



$$\bar{H}_z = \left[\int_0^\infty \left|T_D\right|^2 \left|\frac{n_z}{w_g}\right|^2 \frac{\Phi_{GUST}}{\sigma_{w_g}^2}\, d\omega\right]^{\frac{1}{2}} \quad \frac{1}{M/SEC} \quad \left(\frac{1}{FT/SEC}\right)$$

Figure 2. Crew Sensitivity Index for Ride Quality

Weight Savings

The weight savings attributable to use of the SMCS was evaluated by determining the stiffnesses required to meet the ride quality with and without the use of the SMCS. Only the fuselage stiffnesses were affected by the trade-offs made; the lifting surface stiffness levels were set by strength and flutter considerations. Experience from several other design studies showed that the vertical stiffness was the one requiring the main adjustment. Figure 3 shows two levels of stiffnesses determined from the comparative study. As indicated, the lower level curve met all requirements for strength, stability and control, and flutter, but fell short of providing stiffness enough for ride quality compliance. The higher level curve provides the required level of stiffness to meet the ride quality requirement. The initial level of this latter stiffness was set by establishing a frequency requirement for the first fuselage mode of about 2 hertz (Hz); this had been established as a criterion as the result of several earlier design iterations which showed that when used in detailed computations, ride quality criteria could be met. The weight associated with the difference between the two curves was evaluated as the weight savings attributable to the SMCS.

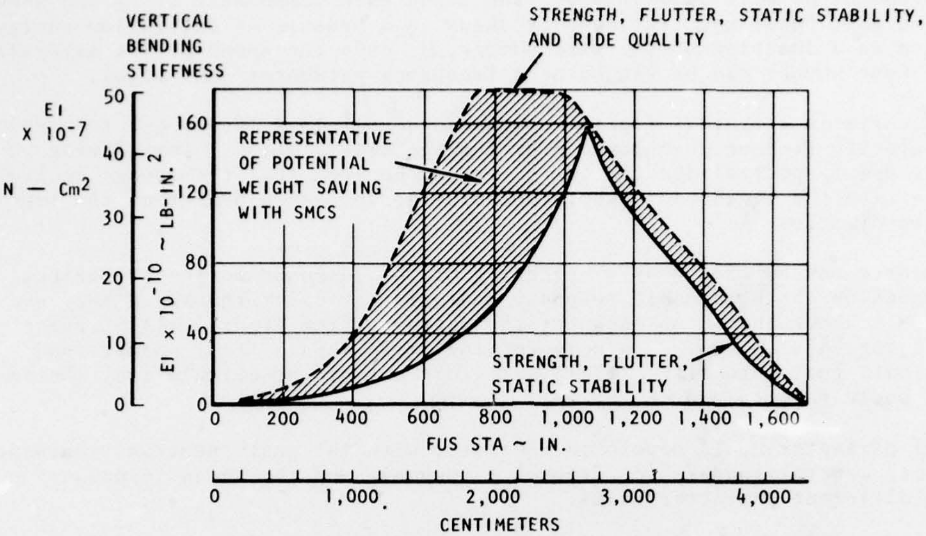


Figure 3. Fuselage Vertical Bending Stiffness Requirements for Vertical Ride Quality

A trade-off study indicated that a total of 4,482 kilograms (kg) (9,880 pounds (lb)) of stiffening weight would have been added to the fuselage to meet ride quality requirements without the SMCS. A total of 182 kg (400 lb) was estimated for the proposed SMCS. Thus, it was estimated that a weight savings of approximately 4,300 kg (9,480 lb) could be realized in meeting the ride quality requirements using an SMCS. As will be discussed in subsequent paragraphs, the details of the SMCS have changed during development, but these changes have not invalidated the order of magnitude of the estimated weight savings. A continual tracking of the weight savings has not been made on as detailed a basis as discussed herein. The fact that the SMCS continued to be required to meet specification ride quality levels served as proof of a continuing weight savings through its use.

KEY SMCS DESIGN CONSIDERATIONS

SMCS Concepts

A successful design for structural mode control embodies control of lower structural modes, up to 10 Hz, for a wide range of vehicle weight, configuration, and flight condition changes, and control of structural modes without interference with basic handling qualities. Solutions to these requirements were defined during the Air Force-sponsored research documented in references 2, 3, and 4. From these studies, a concept of implementation called identical location of accelerometer and force (ILAF) was developed and verified. This concept was used on the B-1.

The ILAF concept was evaluated during the flight test research program on the XB-70. References 5 and 6 document the analyses and tests of an ILAF modal suppression system using a rear-located accelerometer as the sensor and the elevon of the XB-70 as a force generator. A forward-mounted accelerometer and control force generator were also considered and analyzed, but were not tested due to termination of the XB-70 flight test program. The forward control surface was used as a shaker vane, however, and an extremely good match between analytical and measured transfer functions (accelerometer output vs control vane input) was obtained (reference 7).

With the XB-70 experience as background, and with consideration of the primary goal of improving aircraft ride quality at the crew station, it was determined that the force generator for the B-1 SMCS would be a forward-located control vane. After several design iterations, the location shown schematically in figures 1 and 4 was chosen. The 30-degree anhedral permits symmetric deflections of the vane to generate vertical control forces, while antisymmetric deflections provide lateral control forces (figure 5). In actuality, each vane responds separately to inputs from both the vertical and lateral control systems.

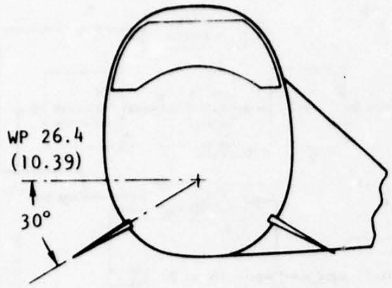


Figure 4. Vane Configuration for Vertical and Lateral Bending Control

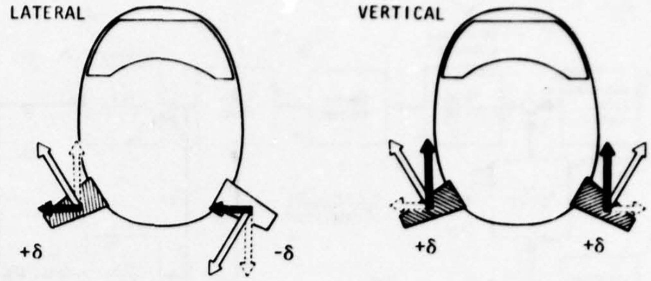


Figure 5. Vane Control Forces

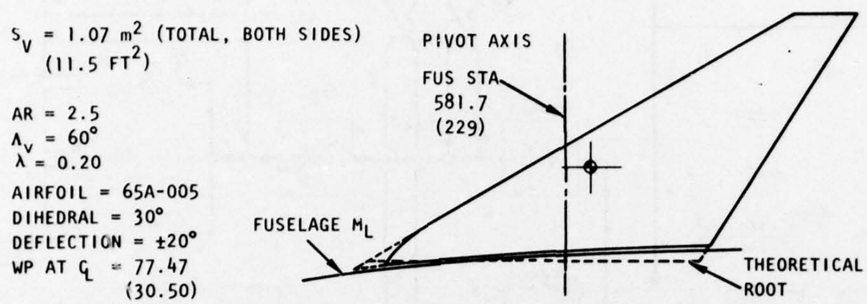


Figure 6. Structural Mode Control Vane Geometric Characteristics

A study of vane normal force characteristics as a function of planform geometry indicated that a vane planform with a 60-degree leading edge sweep, aspect ratio of 2.5, taper ratio of 0.2, and 5-percent thickness would give normal force characteristics which were nearly linear up to a local flow angle of almost 30 degrees and was within the realm of practical construction. The vane planform is shown in figure 6.

A series of wind tunnel tests was run to evaluate vane aerodynamic characteristics and interference effects with the rest of the aircraft. Although it was found that relatively large vane deflections (10 to 20 degrees) could induce significant interference effects on the remainder of the aircraft structure, subsequent flight test showed no significant effects of the interference effects on aircraft ride quality or on the dynamic response to vane excitation. For deflections below 10 degrees, which is where the SMCS operates most of the time, the measured interference terms were very small.

SMCS Vane Effect on Inlet/Engine Characteristics

The placement of aerodynamic surfaces low on the forward fuselage initially gave rise to concerns about the effects of ingested vane-induced vortices on inlet/engine characteristics. Approximately 26 hours of testing in continuous wind tunnel tests of subscale and full-scale models of the B-1 air induction system were dedicated to investigations of these effects. Emphasis was placed on investigations exploring combinations of SMCS vane deflection angles and aircraft maneuvers during operation at M = 0.85. Major destabilizing factors associated with vortex ingestion appear to be taken into account by the distortion methodology. During full-scale, inlet/engine tests, stall-free engine operation was demonstrated during a series of rapid throttle transients. Tests were conducted with vortices being ingested and with off-schedule geometry generating distortion levels approximating design limits. Additionally, no significant differences in engine stall margin were recorded during intentional fuel-pulse stalls with and without vortex ingestion. Initial flight tests to explore SMCS operation during maneuvers were conducted during the phase I flight test program. No indications of vortex ingestion have been evident during the conditions tested, which is consistent with 0.2-scale wind tunnel results.

SMCS Mechanization

The mechanization of the SMCS is presented in figure 7. Some of the numerical values associated with elements of the block diagram shown have changed during the system development, but the mechanization has remained essentially as shown, from proposal through fabrication and test. The system consists of two basic functional parts; one is associated with operating the vane panels in unison to control symmetric structural motion (vertical system), and the other is associated with operating them differentially to control antisymmetric side bending structural motion (lateral system).

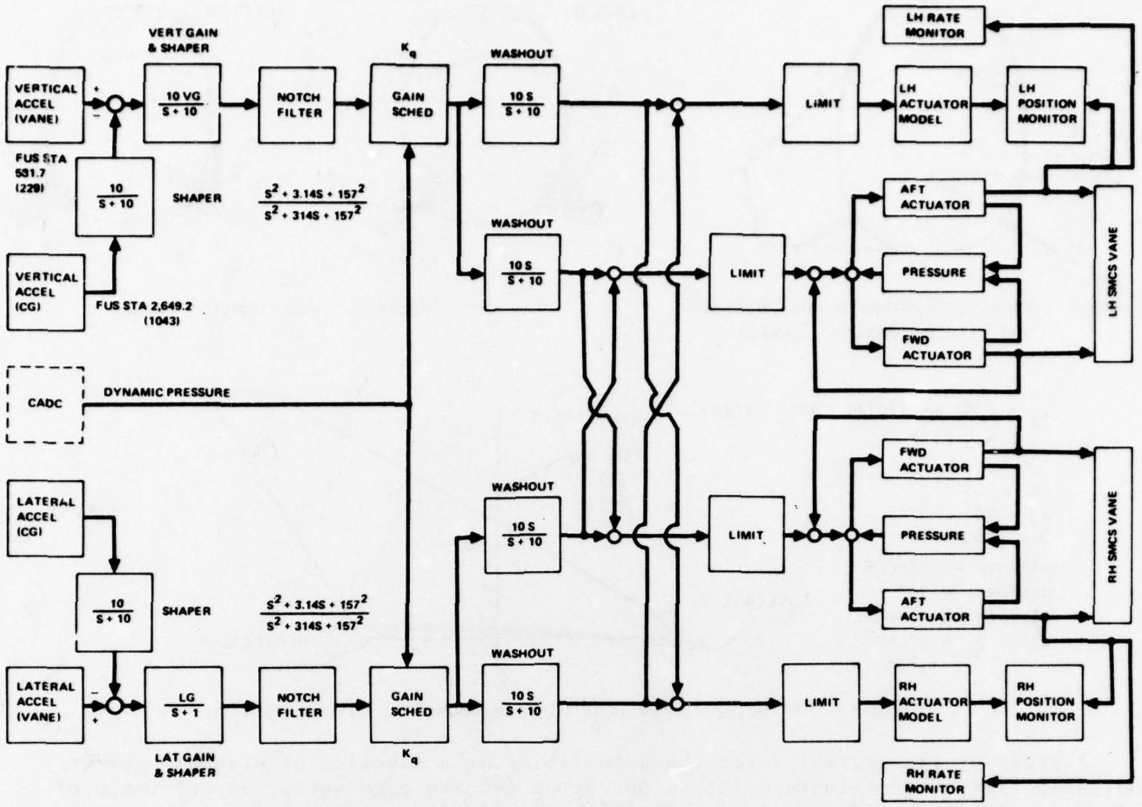


Figure 7. Structural Mode Control System Block Diagram

The implementation of the basic ILAF concept can be seen in the placement of the vertical and lateral accelerometers at the same general location as the control vanes. To augment this principle by eliminating most of the rigid-body motion, a second set of accelerometers is placed near the center of gravity. Because the rigid-body motion content and lower structural modes only are desired from the signal of the center-of-gravity accelerometer, the signal is passed through a simple lag, which eliminates higher frequency structural mode content.

After the difference signal from the accelerometers at the vane and at the center of gravity is obtained, it is passed through shaping and a notch filter designed to eliminate the primary natural frequencies of the vane-actuator installation. The signal then passes through a gain which is scheduled by dynamic pressure from the central air data system. The primary utilization of the SMCS will be during low-altitude high-speed flight. The speed and altitude, however, will vary over a limited range; thus, dynamic pressure gain scheduling was selected to maintain control force effectiveness.

The functional intent of the system is to produce structural damping; therefore, the signal to the force actuation devices must be proportional to structural velocity. This velocity signal is obtained by appropriate gains and shaping networks. Selections of the gains and shaping networks are a function of the structural, aerodynamic, and actuator dynamic characteristics. Basically, simple lags are used to approximate integration of the structural acceleration signals to obtain the required velocity signals.

Washout networks are used to effectively disengage the vertical or lateral functional parts of the system in the event of hardover failures. In addition to isolating hardover failures, the washout networks attenuate rigid-body (whole-vehicle) response acceleration signals that cannot be canceled by the accelerometer signal differencing.

After the washout circuits, the signals are divided and proceed to the independent left and right vane-actuator assemblies. Before reaching the actuators, however, the signals pass through electronic limiters in the circuits. These electronic limiters prevent the vane actuators from making hard contact with the physical actuator throw stops.

Depending upon whether the signals come from the vertical or lateral motion-sensing part of the system, the actuators move the left and right vanes in unison or differentially to produce the required aerodynamic control forces. The system will also respond to mixed signals from the vertical and lateral sensor systems. Pressure sensors coordinate the force output between the forward and aft actuators.

There are two actuators associated with each vane so that a free-floating vane can be avoided in the event of a malfunction. Use of the dual hydromechanical components insures that the vanes can be returned to neutral position and held when disengaged manually by the pilot or automatically by the SMCS monitors. The monitors use vane deflection and maximum vane rate information to detect malfunctions. The part of the monitor that uses vane deflection information consists of a duplicate of the electronics, from the shaping network output to the actuator input and an electronic model of the actuator. Thus, differences between the command vane position and the actual vane position exceeding certain values for a specified time interval are used to automatically disengage the SMCS. The part of the monitor that uses vane maximum rate information disengages the SMCS when maximum rate is sustained for more than an accumulated number of seconds during a specified time interval. This latter monitor is designed to handle dynamic instability possibilities such as limit cycling.

In the early design phases, it was thought prudent to design the SMCS so that it would operate only in conjunction with the SCAS. Thus, any unforseen hardover vane failure effects on rigid-body motion would be attenuated. In retrospect, it appears that this design approach is overly conservative because of the small size of the SMCS vane.

The SMCS is not designed to operate continuously. There is a cockpit switch enabling the crew to turn the system on prior to low-level flight and to turn it off afterwards. Also, while not specifically noted on the block diagram, there is a switch mechanized so that the system is disabled automatically as the landing gear is lowered and enabled as the gear is raised. This feature is necessary to preclude the vane from inducing inertia reaction forces in the absence of aerodynamic forces which will cause instabilities (the so-called "tail wags dog" phenomenon) if the switch is accidently left on or during ground testing.

SMCS Performance

During the development of both the airplane and the SMCS, analyses were made on a continuing basis to monitor the SMCS performance relative to improving ride quality. One such cycle of analyses is discussed in the following paragraphs.

The analytical models of the flexible aircraft used in these design studies employed modal (in contrast to direct structural influence coefficient) techniques. The mass characteristics and stiffness data were continually upgraded to reflect the airplane development; the stiffness and mass reflected in the data presented herein include ground vibration test results. A total of 10 symmetric and 12 antisymmetric structural modes have been included in the analyses.

The aerodynamics associated with rigid aircraft shape reflect wind tunnel test data. The longitudinal-symmetric aerodynamics associated with symmetric structural bending and vertical gusts have been determined using unsteady subsonic doublet-lattice lifting surface theory correlated with wind tunnel test results. The fuselage gust effects were determined using a modified slender-body technique. The horizontal tail control data were obtained using the unsteady doublet-lattice theory. The SMCS vane aerodynamics were quasi-steady and based on theory and correlated wind tunnel tests for both the longitudinal-symmetric and lateral-directional-antisymmetric cases. The lateral-directional rigid-body aerodynamics were from wind tunnel test data, while similar data for the antisymmetric structural bending modes have been determined using doublet-lattice lifting surface quasi-steady aerodynamics; the rudder control effectiveness was determined using doublet-lattice unsteady aerodynamics theory. The side gust loads on the fuselage were obtained using a modified slender-body theory; while the gust loads on the vertical tail were calculated using the unsteady doublet-lattice theory.

The Von Karman gust power spectral density curve was used in calculations of the ride quality (crew sensitivity indexes, $\bar{H}_z$ and $\bar{H}_y$) and the rms accelerations due to turbulence, $\bar{A}_z$ and $\bar{A}_y$. The scale length, L, was 152.4 m (500 ft).

Crew sensitivity index data for the vertical axis, both $\bar{H}_z$ and $\bar{H}_z$ power spectral density curve, are presented in figure 8. Data are shown for the basic aircraft, the SCAS operating, and the SCAS + SMCS operating. The peak at low frequency is the short-period response, and the large structural response at about 18 rad/sec frequency is a mode consisting primarily of first fuselage vertical bending mode motion. As can be seen, the SCAS does its intended job of damping the short-period motion, but slightly excites the primary mode, contributing to vertical motion. The specification level for $\bar{H}_z$ is ≤ 0.028. The data presented show that operation of the SMCS substantially reduces the structural motion (while not interfering with the short period) and do, in fact, show capability for meeting the specification $\bar{H}_z$.

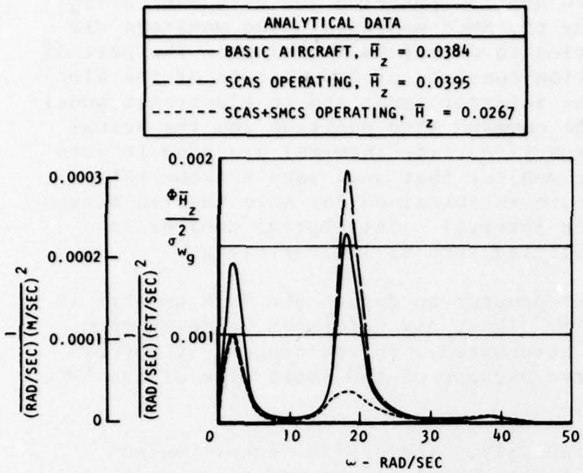


Figure 8. Power Spectral Density Associated With Vertical Crew Sensitivity Index, $\bar{H}_z$, M = 0.85, h_p = 762 m (2,500 FT), Λ_w = 65 , Medium Weight

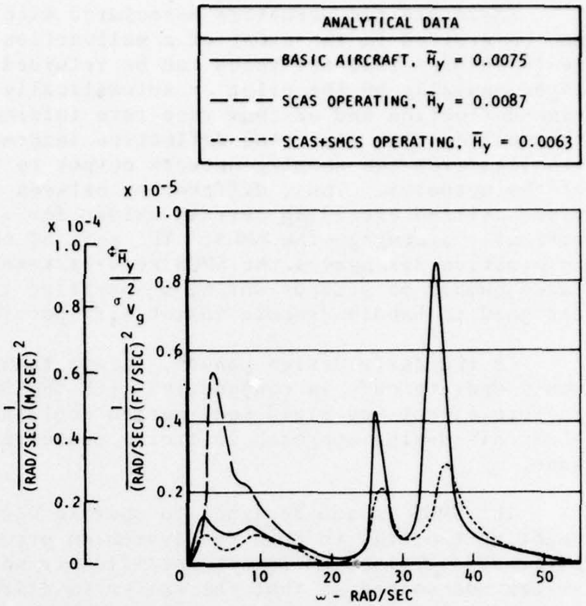


Figure 9. Power Spectral Density Associated With Lateral Crew Sensitivity Index, $\bar{H}_y$, M = 0.85, h_p = 762 m (2,500 Ft), Λ_w = 65°, Medium Weight

The lateral crew sensitivity index, $\bar{H}_y$, and the power spectral density associated with $\bar{H}_y$ are presented in figure 9. Comparable data shown for the vertical case are shown for the lateral; that is, basic aircraft response and the effects of SCAS and SCAS + SMCS operating on that response. The low-frequency responses are related to the Dutch roll mode, and the two responses at 27 and 34 rad/sec are structural responses of aircraft modes which have large first fuselage side bending mode components. The SCAS is shown to modify the Dutch roll response but leaves the structural motion unchanged. Operation of the lateral SMCS does not have as dramatic an impact on the structural mode responses as does the vertical SMCS; however, the specification level of $\bar{H}_y \leq 0.007$ is met.

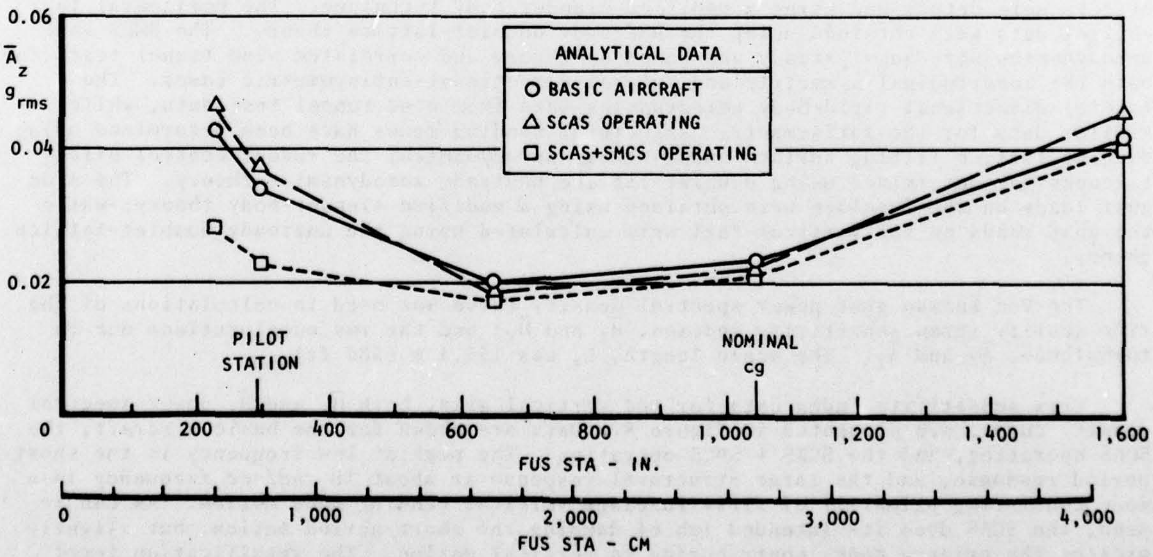


Figure 10. Effect of SMCS on Normal Load Factor Response Along Fuselage Due to Turbulence, M = 0.85, h_p = 762 (2,500 Ft), Λ_w = 65°, Medium Weight

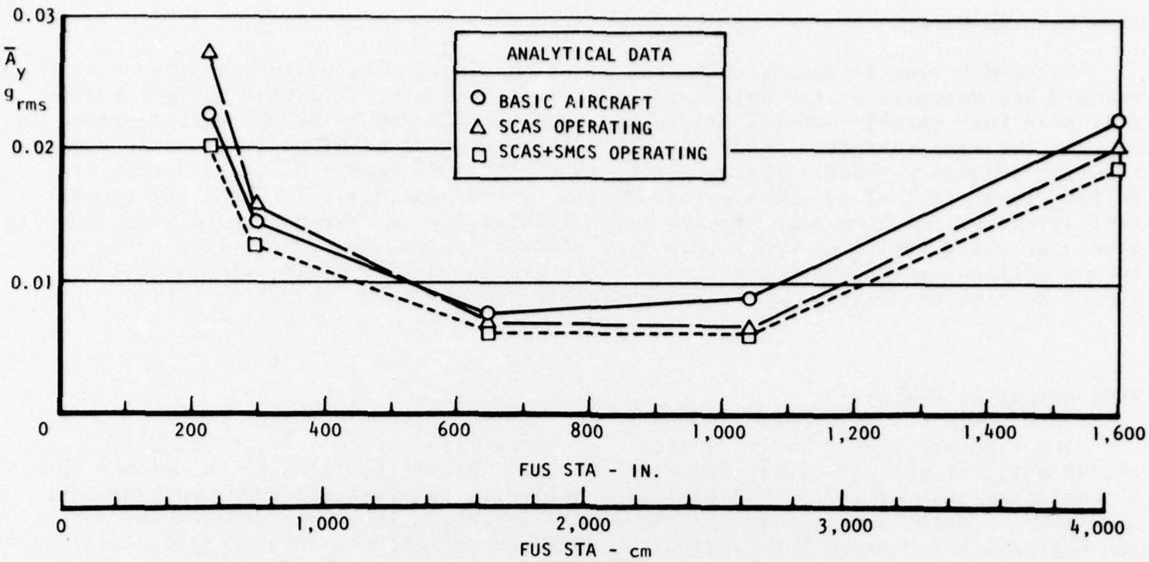


Figure 11. Effect of SMCS on Lateral Load Factor Response Along Fuselage Due to Turbulence, M = 0.85, h_p = 762 m (2,500 Ft), Λ_w = 65°, Medium Weight

While the primary goal of the SMCS is to meet the ride quality requirement at the crew station, it is also of interest to see if the system reduces (or excites) loads at other fuselage locations. Figure 10 shows the effect of SCAS and SCAS + SMCS on the normal rms load factor, $\bar{A}_z$, along the fuselage; figure 11 shows similar data for the lateral rms load factor, $\bar{A}_y$. In both figures it is shown that the SMCS reduces acceleration levels at all fuselage stations below that for either the basic aircraft or SCAS operating.

Stability analysis of the B-1 configuration with the SMCS operating has been performed for both the vertical and lateral axes, using the modified Landahl stability criteria described in Reference 8. The system was demonstrated to be stable in both axes.

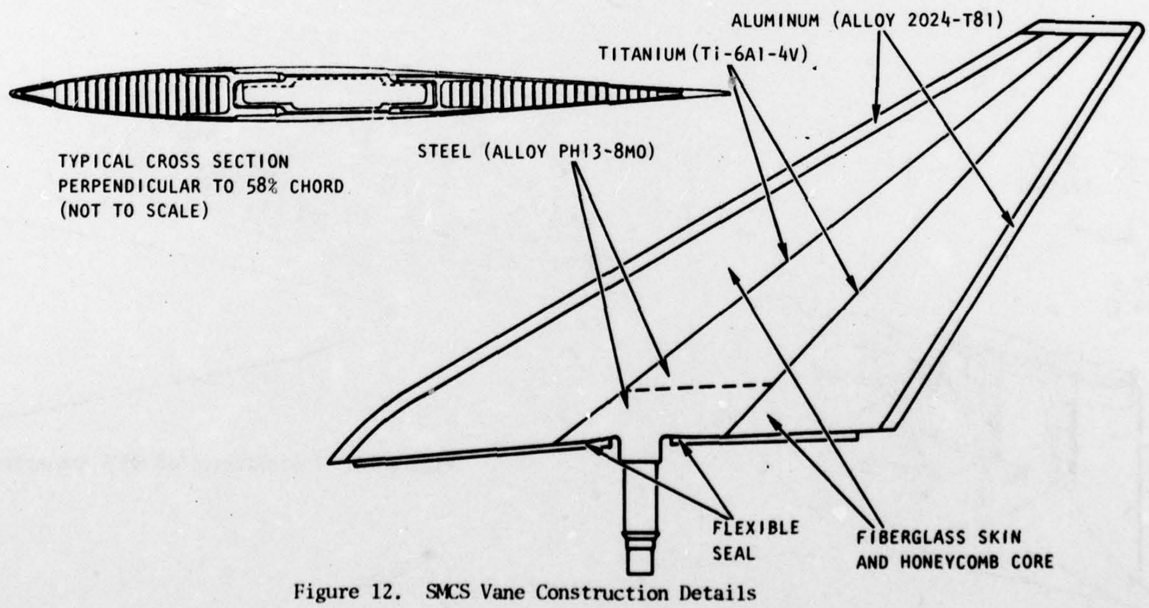


Figure 12. SMCS Vane Construction Details

SMCS DESIGN DETAILS

SMCS Vane Construction

The key features of the SMCS vane construction details and materials used are shown in figure 12. The trunnion and main box skins are made of steel. The box main spars and ribs are titanium. The material forward and aft of the main structural box is fiberglass honeycomb and skins. The leading and trailing edge closeout strips are aluminum.

SMCS Bearing Design

Each SMCS vane is supported by two pivot bearings, mounted in trunnion plates inboard and outboard of the actuator attach fitting (horn). Initial design studies indicated that spindle-mounted needle bearings would probably be the best approach to support the vane surfaces. Concerns as to the wear that might be experienced due to the high-frequency, small-amplitude motions of the SMCS vanes led to selection of Teflon-lined (TFE) plain spherical bearings. These bearings, although they exhibit a coefficient of function more than 10 times that of needle bearings, have exceptionally good wear characteristics under this type of duty cycle. Subsequent life tests of over 1 miliion cycles showed that the SMCS bearings would exhibit acceptable frictional heating characteristics with negligible wear over the required life.

SMCS Actuation Design

The pressure of the basic hydraulic system at its source is 2,757.9 N/cm^2 (4,000 psi); at the end of the lines to the SMCS actuators, it has been assumed that a static pressure of 1,723.7 N/cm^2 (2,500 psi) will be available under maximum-rate conditions. Actuation requirements have been set by maximum vane deflections of ± 20 degrees, and maximum vane rate of 200 degrees per second. The required total hinge moment for the actuation system was 903,878 cm-N (80,000 in.-lb) for each vane. This requirement was met by a dual actuation system, each capable of 35,585 N (8,000 lb) force operating through a 12.6 cm (5 in.) arm.

The fail-safety philosophy for the SMCS was also a significant actuation system design driver. The SMCS was to be a fail-safe system and be free from flutter potential in any failure state. This requirement led to the following implementation. Two servo cylinders actuate each of the two SMCS vane panels; one extends, while the opposite retracts. Each servo cylinder actuating a given vane panel is supplied from one of two separate independent hydraulic systems. The airplane has a total of four separate hydraulic systems numbered 1 through 4. The No. 2 hydraulic system feeds both the right forward actuator and the left aft actuator, while hydraulic system No. 3 feeds the right aft actuator and the left forward actuator. Thus, in the event of a failure in one of the hydraulic systems, sufficient power is available to center and hold both vane panels. Then, in the event of a failure of the second hydraulic system powering the SMCS actuators, a reservoir system holds pressure on the actuators to prevent flutter.

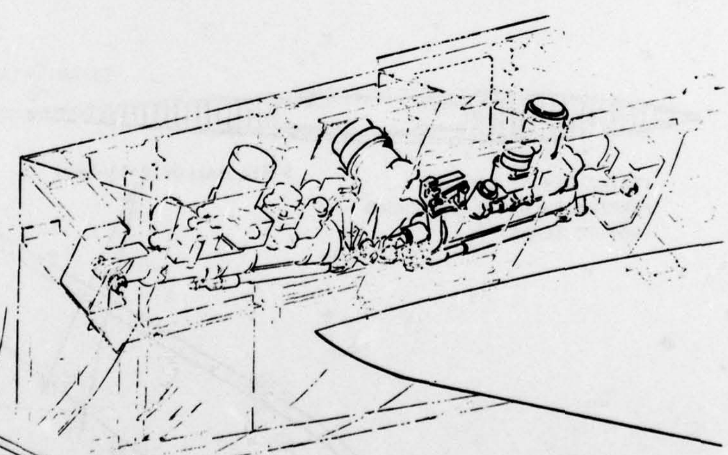


Figure 14. Installation of SMCS Actuators

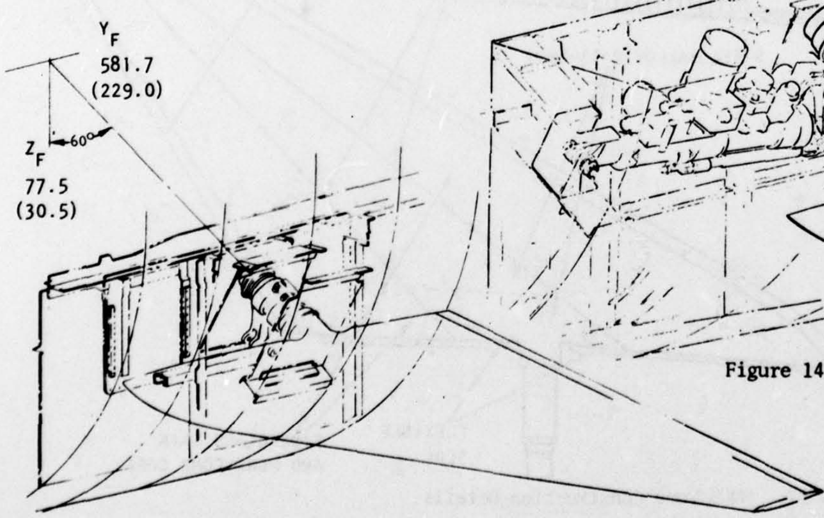


Figure 13. SMCS Vane Pivot Bearing Support Structure

SMCS Vane/Actuation Installation

Figure 13 shows how the pivot bearings are supported and how this support structure ties into other structural elements. Figure 14 presents the details of how the actuators are installed relative to pivot attachment and backup structure.

FLIGHT TESTS

While the analytical work and the hardware development effort gave every indication of a successful SMCS system, it was flight testing that provided the final proof. Herein, highlights of the SMCS flight testing are presented with some comparisons to analytical results, system improvement results, and, finally, flight crew assessments of the system.

SMCS Performance

Perhaps one of the most informative ways of demonstrating the vehicle ride quality characteristics to be modified and the effect of operation of the SMCS is through the viewing of time history plots. A set of these are shown in figure 15. The data were recorded on a flight at M = 0.70 where the B-1 was flying at altitudes of 305 to 610 m (1,000 to 2,000 ft) above the terrain in the local Edwards Air Force Base area. Considerable light-to-moderate turbulence was present nearly continuously. This situation provided an opportunity to test the SMCS in the environment for which it was designed. The SMCS was turned on with both the verticai and lateral gains set at 1.5. To demonstrate comparative aircraft performance with and without SMCS operating, several time periods with the SMCS on and off were recorded, figure 15 being typical of these data.

Vertical and lateral motion at the front end of the aircraft with the SMCS off are shown by the first two time histories of figure 15. As indicated, the primary motion in the vertical axis was the first fuselage bending mode at approximately 3 Hz. The lateral motion was composed of whole vehicle motion near 1 Hz and the first fuselage side bending mode motion of approximately 5 Hz superimposed. When the SMCS was operated, as shown in the next two plots, considerable attenuation of the 3 Hz motion was achieved. Very little motion of the aircraft at lower frequencies appears to be present. The effect of the SMCS on the lateral axis motion was not as dramatic as on the vertical motion, but the 5 Hz motion was partially suppressed. The whole-vehicle lateral motion was not attenuated. It is to be recalled, however, that the SMCS is designed to attenuate structural mode response without adversely affecting whole-vehicle motion (handling qualities). The last two plots in figure 15 show the SMCS right and left vane motion during the time that the SMCS was operating. As shown, the maximum vane deflections seldom exceeded ±6 degrees, whereas ±20 degrees were available. Both the 3 Hz vertical structural motion and 5 Hz lateral structural motion can be seen to drive the vane deflections, the largest component being due to the vertical motion.

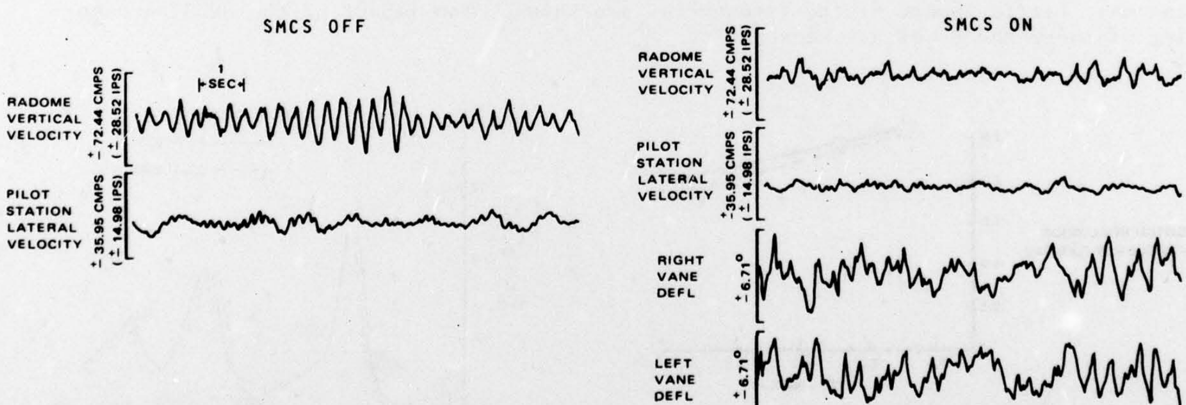


Figure 15. SMCS Performance in Turbulence M = 0.70, h_{AGL} = 305 m (1,000 Ft). Λ_W = 65°

Another conventional way of looking at ride quality performance is in the form of power spectral density plots (PSD) of load factors at the pilot station. Figures 16 and 17 show this type of data for the vertical and lateral load factors, respectively, for SMCS both off and on. Because of other test requirements, the aircraft flown to the date of this writing had no gust boom. Approximate vertical and lateral gust intensities were estimated using nose boom angles of attack and sideslip angles in order to provide normalizing factors for the PSD data. Gust intensities for the data shown were estimated at 1.22 to 1.52 m/sec rms (4 to 5 ft/sec rms). Figure 16 shows that the 3 Hz first fuselage mode, previously shown in the time history plots of figure 15, is the main contributor to vertical motion at the pilot station. As in figure 15, the data of figure 16 demonstrate that the SMCS was very effective in reducing the pilot station load factor. The lateral load factor PSD responses shown in figure 17 are more complex than the vertical responses. The key features of this plot is the whole-vehicle motion at low frequencies, the approximately 5 and 7.5 Hz responses, and a number of higher frequencies. Again, most of these features are recognizable in the time histories or figure 15. The lateral SMCS is seen to have reduced the two main peaks at the expense of exciting some higher frequency modes. The exciting of the higher modes and the solution to this difficulty are discussed in subsequent paragraphs. The net effect of the SMCS at this stage of development was to provide some lateral response suppression, but not as dramatic as in the vertical axis.

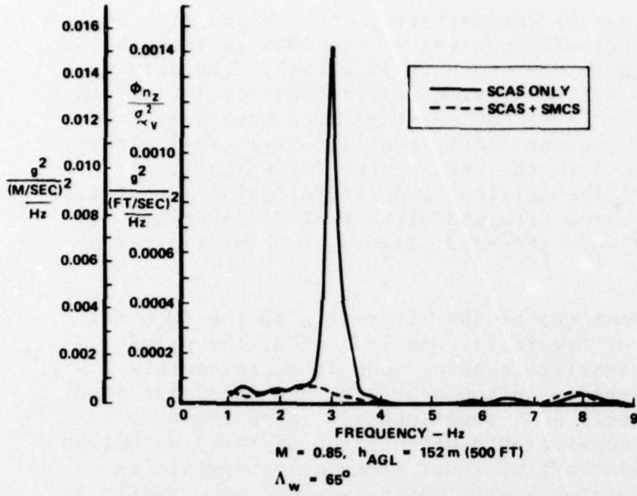


Figure 16. Vertical SMCS Performance in Turbulence as Shown by Power Spectral Density of Vertical Load Factor at Pilot Station

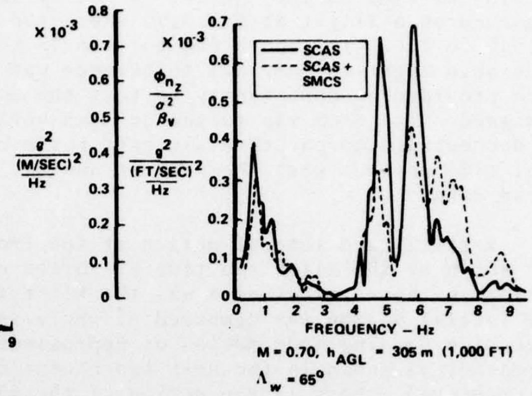


Figure 17. Lateral SMCS Performance in Turbulence as Shown by Power Spectral Density of Lateral Load Factor at Pilot Station

One of the design goals for the SMCS was not to interfere significantly with basic handling qualities. The proof that this goal was achieved is shown in figure 18, where short-period and Dutch roll mode frequencies are displayed for SMCS both off and on. Little impact of the frequencies are shown. The impact of the SMCS on damping of these modes was not measurable.

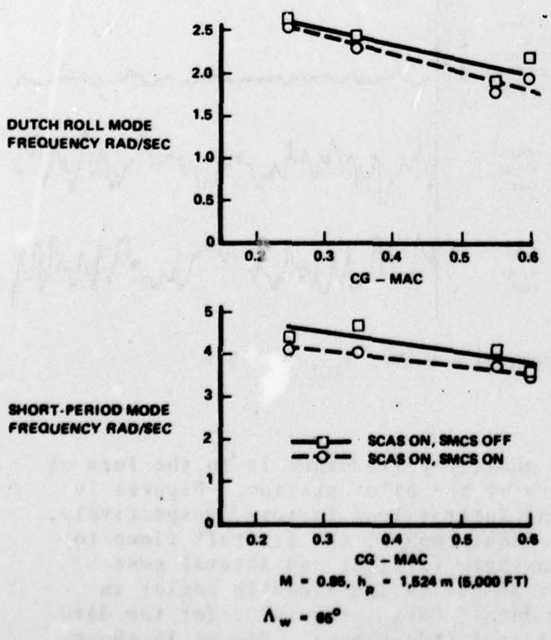


Figure 18. SMCS Impact on Short-Period and Dutch Roll Frequencies

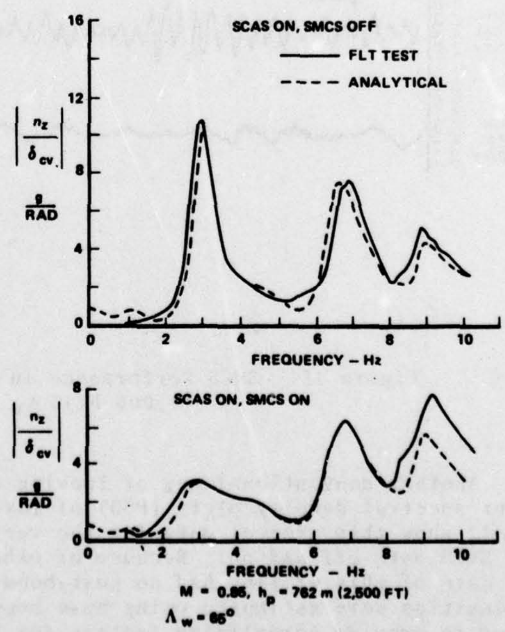


Figure 19. Comparison of Flight Test and Analytical Data, Frequency Response of Vertical Load Factor at FS 582 (229) Due to SMCS Vane Deflection

Analytical Match of Flight Data

Another way of evaluating the SMCS performance was through the use of frequency responses of the vertical and lateral load factors at the vane station due to forcing by the control vanes. These same data provided an excellent chance to check the accuracy of the analytical models of the aircraft and control systems used in the continuing development process.

Figure 19 presents the flight test and analytical frequency responses of the vertical load factor at the vane station due to vane deflection with and without the SMCS operating. From the viewpoint of system performances, these data show the excellent capability in reducing the lower frequency modes. Some excitation of the 9 Hz mode is shown; time history data and pilot evaluations indicate this not to be a significant difficulty. Some slight adjustments in the analytical data in frequency and damping of the highest frequency modes shown were made to obtain the matches shown; the 3 Hz mode required no adjustment.

Figure 20 shows the flight test and analytical frequency responses of the lateral load factor at the vane station due to vane deflection with and without the SMCS operating. Excellent reduction of the lower frequency modes by the SMCS is shown by the flight test data; however, this performance was offset by the excitation of a fuselage torsion mode at 7 Hz. In the early analyses (figure 9), this 7 Hz mode was not revealed. Subsequent investigations showed that the early analyses predicted this mode to be at much higher frequency. When lowered to the observed flight test frequency, the data matches shown were obtained. While not as good a match as for the vertical axis data, the basic phenomena are described.

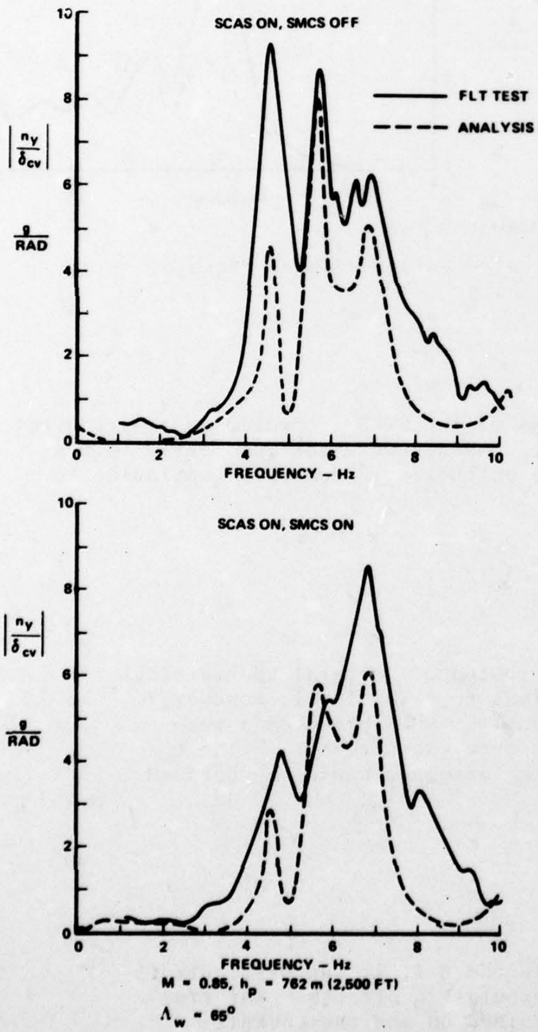


Figure 20. Comparison of Flight Test and Analytical Data, Frequency Response of Lateral Load Factor at FS 582 (229) Due to SMCS Vane Deflection

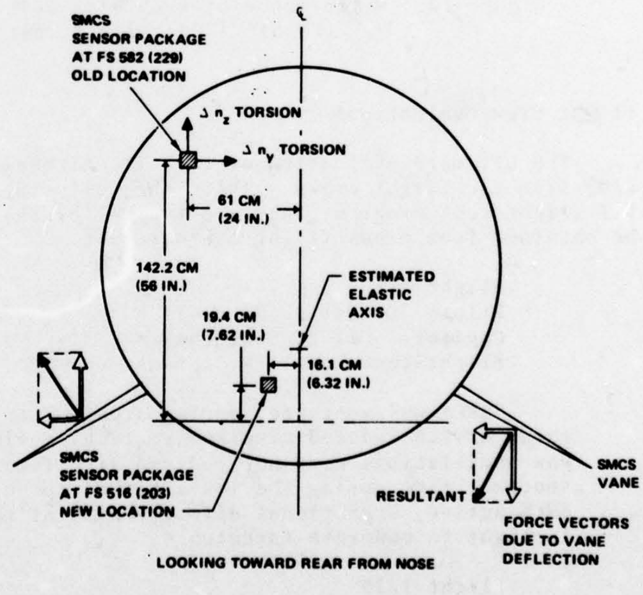


Figure 21. SMCS Sensor Package Locations and Coupling Characteristics

SMCS Improvement

Analytical investigations using the validated analytical model revealed that 7 Hz torsional coupling with the SMCS operating (figure 20) could be eliminated. The source of the coupling can be seen in figure 21. The original forward SMCS sensor package was located high on the bulkhead and some distance off of the centerline, resulting in a substantial moment arm from the elastic axis. This location was dictated by other equipment placement. Early analyses also indicated satisfactory performance (figure 9) with this sensor location. The adjusted analytical model, as described earlier, demonstrated that the sensor would cause coupling into the vertical axis as well as provide adverse feedback to the lateral axis. The solution to this difficulty was to move the sensor to an available location approximately 66 cm (26 in.) forward and close to the elastic axis, as noted in figure 21. Figure 22 shows that vertical axis SMCS with the sensor relocated performs just as well when the sensor was at the original location (figure 8). A significant improvement, however, was obtained in the lateral axis, as demonstrated by the flight test data. The primary 5 Hz sidebending mode was attenuated without excitation of the 7 Hz mode.

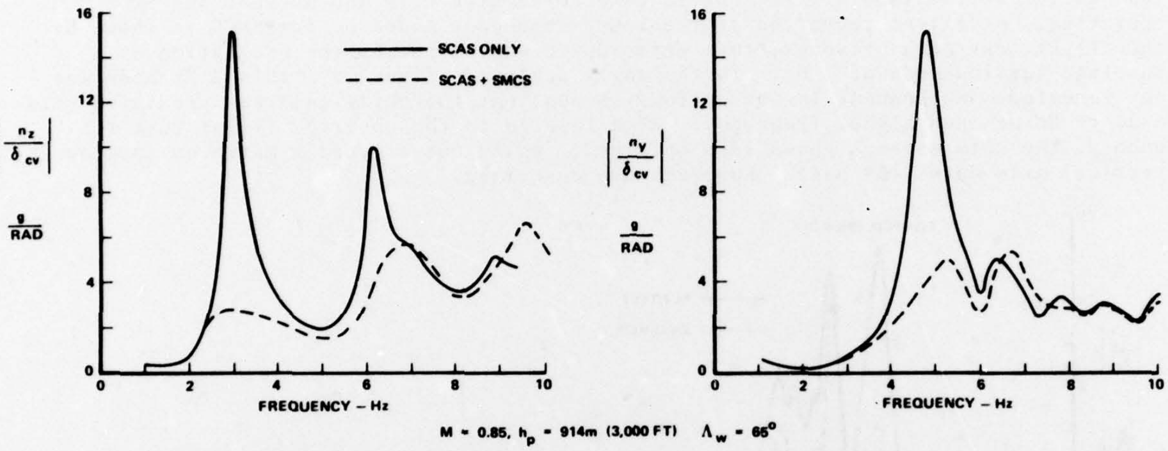


Figure 22. Performance of SMCS With Relocated Forward Sensor Package, FS 516 (203) Flight Test Data

Flight Crew Evaluations

The ultimate evaluation of the effectiveness of the SMCS in providing ride control came from the flight crews. While the following comments were obtained early in the B-1 flight test program, they are typical of the enthusiastic comments continuing to be obtained from crews flying the aircraft.

Flight 1-19
Pilot: Lt Col E. McDowell
Copilot: Col E. Sturmthal
Flight Test Engr: R. Abrams

"SMCS was activated during areas of turbulence. Lateral and vertical gains of 1.5 reduced airplane vertical motions to a low level; however, yaw oscillations were not reduced significantly. SMCS produced a much smoother ride during the low-altitude turbulence encountered. Without SMCS active, operational effectiveness of the airplane would be degraded in light to moderate turbulence."

Flight 1-20
Pilot: C. C. Bock, Jr.
Copilot: T. D. Benefield
Flight Test Engr.: P. S. Sharp

"--The air was fairly turbulent, and in the B-1, it was difficult to judge the level of turbulence because of aeroelastic effects. The ride was very uncomfortable until the SMCS was turned on and the cockpit motion drastically reduced with the vertical and lateral gains at 1.5.--"

CONCLUDING REMARKS

The B-1 SMCS was designed to improve ride quality in the highly turbulent environment at low altitudes. This was done by controlling motion due to structural modes using a relatively simple controller and aerodynamic force generators. This technology, however, is directly usable with any flexible aircraft, military or commercial, whether the objectives be to provide ride quality improvement, load relief, or even flutter suppression. Only system details such as sensor locations, force generator size and location, magnitude of power required, frequency range, and fail-safety requirements would be dependent on these objectives.

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ACKNOWLEDGEMENTS

The authors wish to acknowledge the contributions of Donald T. Bubna to the analytical data used herein.

REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document
	AGARD-AG-234	ISBN 92-835-0225-6	UNCLASSIFIED

5. Originator Advisory Group for Aerospace Research and Development
North Atlantic Treaty Organization
7 rue Ancelle, 92200 Neuilly sur Seine, France

6. Title
ACTIVE CONTROLS IN AIRCRAFT DESIGN

7. Presented at

8. Author(s)/Editor(s)		9. Date
Various	Edited by P.R.Kurzhals	November 1978

10. Author's/Editor's Address		11. Pages
Various	Director, Electronics Division National Aerospace and Space Administration Washington, DC 20546	184

12. Distribution Statement This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.

13. Keywords/Descriptors

- Control configured vehicle
- Fly-by-wire
- Highly manoeuvrable aircraft technology
- Weight reduction
- Aerodynamic efficiency
- Operating cost

14. Abstract

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AGARDograph No.234
Advisory Group for Aerospace Research and Development, NATO
ACTIVE CONTROLS IN AIRCRAFT DESIGN
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AGARDograph No.234
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