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AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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Flying Qualities

(Qualités de Vol)

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.508

Flying Qualities

(Qualités de Vol)

Copies of papers presented at the Flight Mechanics Panel Symposium, held in Quebec City, Quebec, Canada from 15th to 18th October 1990.

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Preface

The validity of constraining the responses of today's control dominant aircraft to conform to the classic flying qualities criteria derived from stability dominant aircraft experience has been an issue for many years. The introduction of full time visual scene enhancement with sensor fusion, and computer generated/interpreted night scenes, also escalates display dynamics into the arena of flying qualities concern.

Integrated flight and propulsion control schemes and direct force controllers have the potential for completely coupling all the sensors with all the controllers to provide any combination of controlled motion from six independently controlled single-degree of freedom systems to a single completely coupled six-degrees-of-freedom system.

These new technologies have expanded flight envelopes, reduced drag, increased manœuvrability, provided the framework for practical gust alleviation and active flutter suppression, and provided flexibility for fault-tolerant, damage-adaptive flight controls.

However, the updating of flying qualities criteria has in general not kept pace with these technological changes. The purpose of this Symposium was to review flying qualities issues today, and to report progress towards their resolution. The following topic areas were covered:

- Flying Qualities Experiences on Contemporary Aircraft
- Application of Flying Qualities Specifications
- Flying Qualities Research
- Flying Qualities at High Incidence.

The concluding "Round Table Discussion" provided the Session Chairmen with an opportunity to share with the Symposium attendees their view of the major issues relevant to their session topic which need to be addressed in the future.

Préface

Depuis de nombreuses années, la communauté de la mécanique du vol s'interroge sur la validité de la méthode qui consiste à moduler les réponses des aéronefs d'aujourd'hui, qui sont caractérisés par la commande, pour qu'elles se conforment aux critères des qualités de vol classiques qui découlent des aéronefs caractérisés par la stabilité.

L'arrivée des systèmes d'enrichissement permanent de l'image combinés avec l'interconnexion des capteurs, ainsi que l'imagerie nocturne créé/analysée par ordinateur, fait passer la dynamique de la visualisation dans le domaine des qualités de vol.

Les systèmes intégrés de commande de vol et de commande de la propulsion et les systèmes de contrôle direct des forces permettent d'envisager le couplage direct de tous les capteurs avec toutes les commandes pour réaliser toute combinaison de mouvement commandé, allant de six systèmes à un seul degré de liberté et à commande individuelle, à un seul système à six degrés de liberté et à couplage intégral.

Ces nouvelles technologies ont eu pour effet d'élargir le domaine de vol, de réduire la trainée, d'accroitre la maniabilité, de fournir l'environnement technologique favorable à l'atténuation des rafales et à la suppression du flottement et d'amener la flexibilité demandée pour la réalisation de commandes de vol insensibles aux défaillances et adaptatives à l'endommagement de l'aéronef.

Cependant, les critères applicables aux qualités de vol n'ont pas suivi ces évolutions technologiques. L'objet de ce symposium a été d'examiner les questions qui se posent dans le domaine des qualités de vol aujourd'hui, et de rendre compte des progrès réalisés en vue de leur résolution. Le symposium a traité des sujets suivants:

- L'expérience acquise dans le domaine des qualités de vol sur les avions modernes.
- La mise en application des spécifications des qualités de vol.
- La recherche en qualités de vol.
- Les qualités de vol à forte incidence.

Le débat "table ronde" qui a clôturé la séance a fourni aux présidents de séance l'occasion d'avoir un échange de vues avec les participants sur les principales questions qui se posent dans ce domaine et qui sont à résoudre à l'avenir.

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Le Panel du Mécanique du Vol tient à remercier les Autorités Nationales du Canada pour leur invitation à tenir cette réunion à Quebec ainsi que des installations et du personnel mis à sa disposition.

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FLYING QUALITIES RESEARCH – QUO VADIS

by

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Thank you Dave for that very kind introduction.

Good morning ladies and gentlemen. It is indeed an honor to present one of the keynote addresses at this symposium. However, I do have a some concern that members of the AGARD flying qualities community now look upon me as a senior citizen for it seems one has to achieve that status in order to give a proper keynote. In retrospect that may be closer to the truth than I care to admit, as my first AGARD presentation was twenty four years ago to the flying qualities symposium in Cambridge England. At that symposium I delivered a paper that presented a controversial criterion on the susceptibility to longitudinal pilot induced oscillation (PIO) which was developed jointly with Mr. A'harrah, current chairman of the flight mechanics panel. My comments today may be equally controversial, but in a broader context.

My remarks this morning are directed toward fixed wing aircraft, principally high performance tactical aircraft. However, as evidenced by the title of several papers on the agenda at this symposium, they will soon be applicable to helicopters as well.

I ask the students of Latin to excuse my very liberal translation of that beautiful language in the title of this talk. However, I believe flying qualities research is at a cross roads and that the community should address the question "What are the future directions in this discipline?"

To place this thesis in proper perspective, I will briefly trace the history of flying qualities, to explore the more noteworthy highlights of the past as a back drop to explore where flying qualities research might be going in the future.

Before embarking on our historical journey, I believe we should take a moment to consider this technical area and those characteristics which make it unique and fascinating area of research. It is one of the few disciplines which ties together highly complex dynamics analyses with the performance and likes and dislikes of the human operator. practitioners of the art must:

o Be thoroughly schooled in aircraft dynamics and aerodynamics

o Be familiar with the limitations of the human operator

0 Understand the attributes of mission effectiveness

In essence flying qualities research presents the classic interface problem among the disciplines of aerodynamics/dynamics, electronics/controls, and human factors. Increased integration of these disciplines and a blurring of the distinction between them characterizes recent trends. There is no reason to believe this will change in the foreseeable future.

Thus far I have referred to "Flying Qualities" as the topic of my presentation, and indeed that is the title of this symposium. However when examining the agenda for this meeting, I note the intermixing of the term "handling qualities" with that of "Flying Qualities". I would not have thought much of this difference in terminology, had not our session chairman, Mr. Key recently brought it to my attention that he believes there is a distinct difference in the application of these terms. It seems that "Handling Qualities" is concerned with flight control (note-not controls) and produces design criteria, assessment techniques, and design tools. Whereas, "Flying Qualities" is concerned with the interface with the pilot primarily to reduce workload and improve task performance. If one accepts those distinctions, I believe history shows us that early experimenters were concerned primarily with "Handling Qualities". However, modern practitioners of the art must concern themselves with both aspects of the discipline.

The importance of handling qualities was certainly recognized by the Wright Brothers, as seen in this excerpt from a speech Wilbur made in 1901:

> "The difficulties which obstruct the pathway to success in flying machine construction are of three general classes: 1) those which relate to the construction of the sustaining wings, 2) those which relate to the generation and application of power required to drive the machine through the air, 3) those relating to the balancing and steering of the machine after it is actually in flight. Of these difficulties the first two are already to a certain extent solved; but the third, the inability to balance and steer still confronts students of the flying problem. When this one feature has been worked out the age of flying machines will have arrived, for all other difficulties are of minor importance."

We all know that during the next two years, the Wright Brothers succeeded in solving that difficulty, at least to the point that their flying machine had sufficient control to successfully accomplish relatively safe flight. However, we would all question the adequacy of the flying qualities.

Some eight years after the first flight of the Wright Brothers, the next major component of today's flying qualities was developed by contemporary mathematicians. It was the British applied mathematician, G. H. Bryan who placed the hole problem of airplane dynamics on a more rigorous mathematical basis in 1911. Bryan introduced the concept of "stability derivates," broke the total airplane motion down into symmetrical and rotative components and uncovered the nature of the airplanes natural frequencies. Thus the foundations of aircraft dynamic analysis were established.

It was not until almost forty years after the Wright Brothers first flight that the final element of modern flying qualities was established. The work at the national advisory committee for aeronautics or NACA at Langley Field in the U.S. which began in the 1930's, was coming together in 1940, just prior to World War II, in a group of sophisticated programs to correlate airplane stability and control characteristics with pilot's opinions on the airplane's "flying qualities." This work focused on those factors that could be measured in flight and could be used to define quantitatively the flying qualities of airplanes. The fundamental nature of that body of technology that we call flying qualities has not changed in almost fifty years.

Almost simultaneous with the establishment of flying qualities as a technical discipline, was the development of the first flying qualities specification. This work was performed by E. Warner of NACA the when he was asked to prepare the specifications for the Douglas DC-4 airplane. Warner discussed the problem with airline pilots, industrial development engineers, and the NACA staff, and his specification was the result of these studies. While Warner is credited with the development of the first flying qualities specification, the U.S. Army Signal Corps had some thoughts on this matter in 1907 when they drafted the specification for the first military airplane to be procured from the Wright Brothers. Flying qualities were not even given a stand alone paragraph in that one page specification, but merely a sentence which is as follows:

> "During this trial flight of one hour, it [the airplane] must be steered in all directions without difficulty and at all times under perfect control and equilibrium."

Those requirements are almost elegant in their simplicity.

We like to believe that we have come a long way from those simplistic times the flight regime of early manned aircraft was so rudimentary that control power was the key to success, and the aerodynamic instabilities could be stabilized by the pilot after a little practice. As airplanes flew faster and became more heavily loaded, the divergences of the unstable airplane became too rapid for the human pilot to cope with and design for inherent aerodynamic stability became mandatory. As the airplane moved into more and more complex flight regimes, the capability of designing for complete aerodynamic stability throughout the flight envelope became nearly impossible. Due to the great nonlinearities arising from power effects, compressibility and aeroelasticity we have had to introduce stabilization elements through the control system to aid the pilot in controlling the airplane to perform the mission. Indeed, modern control technology has progressed to the point where we now intentionally design airplanes to be unstable over a large portion of their flight envelope to enhance performance and mission effectiveness.

These advances while contributing to overall mission effectiveness have caused their share of problems for the flying qualities engineer. As designers rely more and more on the flight control system to compensate for poor dynamics and aerodynamics of the airplane, the flight control engineers have been forced to ever higher levels of complexity, resulting in total system dynamic characteristics several orders greater that those first identified by Bryan. The result is that we are no longer able to identify the characteristic modes, not only mathematically, but also in the airplane response characteristics.

As this situation evolved over the past fifty years, we have attempted to define and redefine criteria to insure "good flying qualities" with the result that today's specification is an extremely complex document which has to be supported by a back up manual over eight hundred pages in length. Still we have problems defining flying qualities criteria. This has resulted in efforts such as the notion of equivalent systems. In this concept an equivalent set of second order response characteristics are derived from the modal characteristics of the complete higher order airframe. The criteria for acceptability are now defined in terms of these equivalent characteristics. In my view we are attempting to "force" airplanes which exhibit new and unique dynamics into our traditional mode of thinking about flying qualities criteria. To what extremes can we continue this practice?

Coupled with the increase in control complexity is the increasing trend toward cockpit automation. We have such programs as integrated fire and flight control, wherein we demonstrated the ability to shoot down a drone under flight conditions where such a feat was considered impossible in the past. The pilot remarked, "I don't know if he [the automatic system] got it or I got it, but we got it!" Considering that much of our flying qualities research for military aircraft is devoted to defining the characteristics of a stable and precisely controllable weapon delivery platform, what then should be the nature of this research in the above example?

"The drone demonstration" was accomplished almost ten years ago, and we have been pursuing technology development for automated ground attack, automated target hand off, automatic terrain following/terrain avoidance, automated systems monitoring and flight management and much more. This leads me to the question, "What is the role of man in cockpit?" The traditional response has been that he will become a manager of the battle. I have not discussed this with very many pilots, but I do know from long experience that pilots like to fly airplanes, and all of our governments spend a considerable sum to insure that they can do it very well. Management training is significantly less expensive.

A survey conduct by the Naval Weapons Center in China Lake, California a few years back of Marine attack pilots indicted that while control of the aircraft trajectory was considered critical to the success of the mission, it was not consider a difficult or high workload task. What was surprising was that they rated weapon interface critical to the mission as well, but as a high workload task. The weapons were iron bombs! These admittedly sparse data raise the following question in my mind:

> "In our eagerness to automate cockpit functions, have we overlooked the preferences of the pilots and automated those things which we think will be beneficial?"

Advances in technology are also bringing in new modes of control. The U.S. Air Force is currently flying a modified F-15 airplane which integrates the flight control with the propulsion system which includes in flight thrust vectoring and reversing. A paper describing this program will be presented later in the symposium. The point I would make here is that this integrated control concept may result in response characteristics which are not generally recognized in the classical flying qualities sense, but which may be very acceptable to the pilot.

Closely related to the above is the topic of high angle of attack post stall flight, which is the subject of the final session of this symposium. Airplane response characteristics in this regime are far from classical. Accordingly, we have taken to developing a new term -- aircraft agility. There is currently great debate in the united states on identification of these agility parameters and the range of acceptable values. There is a tendency to agree these by consensus which I find very disturbing. I find this very disturbing because it ignores the foundations of Bryan, Gilruth and others who pioneered the concept of relating the flight mechanics parameters to pilot preference and performance.

Finally, we all recognize the importance of simulation, both ground based and in flight, to flying qualities research. However, from my observations of some recent simulations, we are no longer building systematic data bases upon which to base new or revised criteria. Rather, the researcher has a concept for a new mode of integration, writes an algorithm to implement it, and conducts simulation to prove it works. The result is that if the airplane designer does not wish to utilize that particular mechanization, he has no data base upon which to base alternate designs. It appears that the days of systematically varying parameters and studying their overall effect on pilot opinion and performance is becoming a lost art. I can only reflect that future designs will suffer.

By now you have heard some of my concerns and recognize why at the start of this talk, I suggested that it might be controversial. I would like to conclude by posing some challenges to the community here today. I ask you to consider the following questions as you go through your program this week, as well as in your future work.

- Why should we continue to try to define the characteristics of advanced airplane configurations which no longer exhibit classical response characteristics, such as the short period or dutch roll modes, in those terms?

- What is the role of man in the cockpit, and how should this role be addressed in future flying qualities research?

- Are flying qualities criteria and the resulting specifications becoming too complex? If so, what are some alternate approaches?

In closing, I ask you to again consider my original thesis, that flying qualities research is assuming new directions. Further, those here today must determine what those directions are and develop the research agenda to pursue them.

Thank you for your kind attention and I wish you success in your symposium.

L'ADAPTATION DES QUALITES DE VOL AU PILOTE ET A LA MISSION

par J. Coureau, Directeur de la Sécurité des Vols
DASSAULT AVIATION - 13800 - Istres - France

L'aviation a débuté comme un grand jeu sportif auquel se sont livrés des fanatiques assez fortunés pour fabriquer la machine, assez ingénieux pour qu'elle puisse voler, et assez téméraires pour la faire voler.

Ceux qui ont réussi avaient par dessus tout du génie !

Ces constructeurs-pilotes ont dû obtenir assez rapidement une stabilité suffisante pour permettre des vols de plusieurs minutes sans fatigue excessive, une bonne précision de pilotage pour réaliser de façon répétitive et sans trop de casse les manoeuvres de décollage et d'atterrissage, puis la tenue de palier, puis le virage. Les premières courses "au pylône" imposaient déjà une certaine maîtrise dans ces domaines.

Les règles en usage pour dimensionner les gouvernes, caler les surfaces portantes, tenir compte des couples d'hélice, etc... devaient se communiquer oralement. Bien peu d'écrits de l'époque font état de ces problèmes. Ils furent pourtant résolus et la première guerre mondiale voit relativement tôt apparaître une aviation militaire utilisable.

Les aviateurs sont d'abord chargés de régler les tirs d'artillerie. Mais pour déloger ceux d'en face, on crée l'aviation de chasse vers 1916. La qualité de la plate-forme devait être excellente car le pilote devait manoeuvrer pour se placer à "portée de fusil" et lâcher les commandes pour tirer sur un adversaire qui manoeuvrait pour éviter. Très vite l'armement de bord a évolué vers des mitrailleuses puis des canons fixes dans l'axe avion, preuve évidente que la maniabilité des avions en permettait un usage performant. L'abandon de cet armement est souvent évoqué, mais son usage est encore répandu de nos jours.

En France, un organisme d'Etat, créé en 1920, débute l'étude réellement scientifique des qualités de vol des avions; ses pilotes et ingénieurs trouvent les bonnes règles, les communiquent aux constructeurs puis tentent de les faire respecter.

Ils découvrent la stabilité longitudinale statique, la stabilité de route, le mouvement spiral...et aussi le flottement.

Un des constructeurs porte un grand intérêt aux qualités de vol de ses avions, il incite ses ingénieurs à participer aux vols de développement et à tenir compte de l'avis du pilote ; il conserve ces principes tout au long de sa brillante carrière. J'ai eu le plaisir de travailler sous ses ordres pendant 30 ans : vous avez reconnu Monsieur Marcel DASSAULT.

LES AVIONS CLASSIQUES :

La deuxième guerre mondiale réalise une grande confrontation des matériels.

De belles machines deviennent célèbres pour leurs qualités de vol. On demande aux bombardiers une bonne stabilité longitudinale et transversale car les moyens de visée et les calculateurs associés réclament pendant le "bomb run" des corrections précises et bien amorties.

Les chasseurs par contre ont besoin de manoeuvrabilité et de maniabilité pour conduire les combats basés sur l'utilisation des armes dans l'axe de l'avion. C'est la course à la puissance pour les moteurs; les gouvernes doivent équilibrer des couples d'hélice très importants.

A l'apparition du réacteur, l'accroissement de vitesse qu'il permet révèle un nouvel accident de comportement : la compressibilité. Les phénomènes les plus courants sont des vibrations (buffet), des oscillations peu contrôlables (purpoising), de l'auto-cabrage (pitch-up), des défauts de compensation des gouvernes. Cela créé de nouveaux soucis pour le pilote de combat.

L'emploi des profils minces guérit ces inconvénients mais en amène d'autres. L'aérodistorsion revient en force pour réduire les vitesses de roulis. On rencontre des décrochages dynamiques plus violents. Rendues plus plates par l'inertie accrue du fuselage, les vrilles sont réputées difficiles à contrôler.

Les qualités de vol requises :

1 - La sécurité d'emploi : l'avion doit pouvoir être employé sans danger jusqu'aux limites de son domaine de vol, tant à grande vitesse ou grand Mach qu'à basse vitesse, à haute altitude comme près du sol. Ceci suppose que soient maîtrisés les comportements en subsonique, transsonique et éventuellement supersonique aussi bien qu'au cours de décrochages statiques et dynamiques.

Dans différents pays, des normes strictes sont établies qui ont pour but d'éviter le renouvellement de défauts ayant amené des accidents. S'appuyant sur des cas particuliers, elles manquent parfois d'universalité. Le fait de les respecter ne garantit pas vraiment la qualité de l'avion.

Le transsonique provoque sur les gouvernes et les timoneries associées des moments à la dynamique et aux variations si rapides qu'il n'est pas pensable de les compenser efficacement. On a recours aux servo-commandes irréversibles généralement hydrauliques : la sécurité repose sur deux circuits indépendants alimentant des servo-commandes double-corps. Le pilote ne ressent plus qu'un effort proportionnel aux déplacements de la commande (boîte à ressorts) et aucune réaction de gouverne ne "remonte au manche".

L'efficacité des gouvernes varie tellement, dans un domaine de vol très élargi, qu'il faut adapter la commande aux conditions de vol. Ceci est surtout nécessaire en profondeur : un delta peut présenter une efficacité de profondeur de 0,3 degré par "g" à basse altitude et grande vitesse, lorsqu'il est centré arrière ! Le contrôle en devient alors très délicat : il y a risque de "pompage piloté". On recherche donc des centrages très avant, avion lisse, pour accepter les pertes de stabilité dues aux emports sous voilures.

Des amortisseurs à commande électronique sont installés, des lois de déplacement non linéaires, des lois d'effort à plusieurs pentes, variables en fonction de la vitesse et même du facteur de charge, des amortissements visqueux (dash pot)...

Bientôt une chaîne électronique fournit la commande normale, la commande mécanique servant de secours. On est cependant en simple chaîne, le "secours" doit être d'excellente qualité et conserve donc tous les dispositifs ci-dessus.

On réalise ainsi des avions polyvalents sur lesquels le pilote peut passer des faibles vitesses aux grands mach, de l'avion lisse à "l'étagère à bombes" sans devoir porter au pilotage une attention trop soutenue : la perte d'aides au pilotage n'est pas trop gênante et permet dans la majorité des cas la poursuite de la mission.

Sur les avions munis de radar, il est préférable de disposer d'une stabilisation lors de l'observation attentive de l'écran puis de retrouver, si possible, l'entière maniabilité du chasseur ; divers dispositifs ont tente de répondre à ces besoins contradictoires : auto-commande et stabilisateurs de roulis, pilote automatique embrayable et débrayable instantanément, pilote automatique transparent. Accessoirement ils réalisent tous un auto-trim bien agréable.

2 - <u>L'Efficacité</u>

Le combat classique impose toujours une maniabilité excellente mais dont la recherche va à l'encontre des qualités de vol classiques comme la stabilité. Sur les avions conçus pour le temps de paix on opère donc un compromis qui est plutôt du côté de la sécurité : efforts par g élevés, stabilité statique forte en volant centré avant.

<u>LES COMMANDES DE VOL ELECTRIQUES</u> :

L'idéal, connu depuis longtemps, est de faire piloter l'avion à travers un calculateur électronique qui assure les amortissements et les protections, et qui permet, grâce à ses réactions très rapides, d'accepter les accidents aérodynamiques, de reculer le centrage jusqu'au vol en instabilité statique, procurant ainsi un gain important en performance. Ceci a réhabilité la formule Delta.

La commande mécanique traditionnelle n'est plus utilisable dans ces conditions de vol car les écarts augmentent beaucoup trop vite pour être correctement contrôlés par un pilote. Il faut donc un autre calculateur en secours, puis comparer les deux. En cas de panne, lequel a tort ? On voit bien que c'est l'escalade vers trois et même quatre chaînes.

C'est l'arrivée sur le marché de calculateurs électroniques compacts qui a rendu possible la réalisation de ces commandes de vol électriques sans secours mécanique.

La suppression de la transmission mécanique a posé quelques problèmes psychologiques, bien vite oubliés devant la somme des avantages obtenus.

Dans l'aviation civile, ces problèmes ne sont pas encore tous oubliés !

Que permettent les commandes de vol électriques ?

- une très bonne stabilisation de plate-forme en conditions de vol instable, donc la possibilité d'une très grande maniabilité. Ceci résoud le problème des chasseurs précédents : un avion stable comme le roc et très ardent ! Quel avantage en combat !

- la manoeuvrabilité maximale grâce aux limitations automatiques en facteur de charge et en incidence, dans toutes les configurations. C'est l'avion que l'on manoeuvre très vite, à fond, sans souci de la perte de contrôle ou de la casse. Bien sûr les couplages roulis-lacet rendent le dosage du pied inutile. Autres énormes avantages en combat !

- la facilité de pilotage : l'avion à mettre entre toutes les mains, qui pousse vers le haut les performances individuelles, efface les différences de pilotage, accroissant ainsi l'efficacité d'une force aérienne avec le minimum de temps d'entrainement.

- les nombreuses fonctions nouvelles :

  le pilote automatique de base peut être inclus dans les calculateurs, donc multi-chaines, ce qui accroît la sécurité d'emploi à très basse altitude et la disponibilité,

. l'évitement de sol automatique devient possible, même en manoeuvre, grace au relief numérisé en mémoire,

. la récupération de l'avion, en cas de perte de conscience du pilote, est réalisable, par analyse de ses réactions au moyen de détecteurs peu contraignant : mouvements de la tête, efforts aux commandes,

. le contrôle aisé de l'énergie en combat, sans consulter d'instrument, par limitation présélectée des évolutions ou par signaux tactiles au pilote,

. la gestion optimale de la trainée ou de la portance par utilisation de toutes les surfaces mobiles,

. l'utilisation possible des six degrés de liberté pour découpler les translations des rotations autour du centre de gravité, bien que la complication des commandes et des stratégies de pilotage n'ait pas encore permis d'application opérationnelle en combat,

. l'allègement de certains efforts structuraux,

. l'anti-turbulence,

. l'approche dynamique ou même le maintien des grandes incidences (post décrochage ou post-stall), etc...encore que l'intérêt opérationnel en soit largement discuté. Ceci peut changer très vite en fonction de l'évolution des moyens de désignation de cible et des domaines de tir des missiles.

Les limites de cette liste sont celles de l'imagination !

<u>N'y a-t-il pas de revers à ces médailles</u> ?

Si, bien sûr.

- l'instabilité, point trop n'en faut : elle réclame de grands braquages de gouvernes ou des surfaces excessives ou même une poussée orientable pour le contrôle dynamique près des limites ; la performance s'en ressent, le poids et le prix grimpent.

- les manoeuvres faciles à facteur de charge élevé (9 g+) deviennent physiquement éprouvantes, accroissent fortement les risques de perte de conscience ("g" LOC) si l'on ne prend pas de précautions pour limiter les cadences et anticiper les prises de "g". Les avions "agressifs" le sont aussi pour leur propres pilotes !
Les risques de perte d'orientation spatiale sont accrus.

- la limite à la charge sûre est trop basse en cas de ressource tardive ou de manoeuvre d'évitement. La butée surpassable du MIRAGE 2000 résoud partiellement cet inconvénient.

- les innovations coûtent cher ! développer un logiciel et le qualifier exige des années de travail de nombreux spécialistes et l'emploi intensif de moyens de simulation.

- la complication des modes rend incertain le diagnostic du pilote en cas de panne, d'anomalie ou simplement de non prise en compte.

- les modes de pilotage découplés ne sont pas faciles d'emploi ou demandent un entraînement spécifique entretenu, donc cher.

Enfin, on peut se poser des questions sur l'intérêt de l'adaptation au combat tournoyant à l'heure où l'armement principal est constitué d'engins à grande portée ou de missiles de combat rapproché, beaucoup plus manoeuvrants que ne sera jamais l'avion piloté !

Pour la courte portée, la désignation par viseur de casque et le tir d'un missile agile non accroché peuvent remplacer des manoeuvres éprouvantes. Si la faisabilité de ces tirs est évidente, leur mise au point n'est pas terminée... On est par contre beaucoup moins optimiste au sujet du comportement d'un missile tiré pendant une manoeuvre style "cobra" ?

Il reste cependant que l'avion tireur doit bien se placer, très tôt, et donc manoeuvrer vite.

<u>Quelques questions se posent</u> :

- Jusqu'où doit-on pousser la sophistication et son prix ?

- Avec quelle précision doit-on approcher les limites de la structure ou celles du contrôle de l'avion ? Faut-il couvrir toutes les manoeuvres imaginables pour éviter quelques consignes restrictives ?

- Est-on parfaitement sûr de la protection contre la perte de contrôle ? Faut-il concevoir une détection de vrille et un mode de récupération, automatique ou non, qui traite le problème sans risque de l'aggraver ?

- Les pilotes accepteront-ils les modes de pilotage où un dispositif aura autorité sur leurs propres ordres (évitement de sol par exemple) ? Les cas de déclenchement intempestifs sont-ils exclus, vue la faible précision des capteurs ?

Au milieu de cet imbroglio, l'établissement des normes et de règles d'application posera problème ! La mise au point puis la qualification des logiciels est un travail ardu, difficile à contrôler par un service officiel. Quels principes devront être érigés en règlement ? Une bonne partie de ces techniques n'ont pas encore subi l'épreuve du feu.

LES MOYENS D'Y PARVENIR :

On constate que, comme par le passé, c'est l'expérience et la qualité des personnes impliquées, leur esprit d'équipe et la collaboration étroite des diverses parties prenantes, ingénieurs de conception, ingénieurs d'essais spécialisés, pilotes d'essais, et surtout utilisateurs, pour d'abord définir le besoin puis corriger les erreurs précédentes et assurer la qualité du produit : avant hier on maniait des dispositifs mécaniques, hier de l'hydraulique et de l'électronique, aujourd'hui surtout du logiciel.

COMMENT EVOLUE LE PILOTE ?

Les prédictions les plus pessimistes portaient sur sa résistance physique aux forts facteurs de charge. Musculation et entrainement intensif étaient fortement conseillés. Il se trouve que la musculation ne porte pas les fruits escomptés et que le problème principal n'est peut-être pas là !

La cause principale d'accidents sur les machines modernes est paradoxalement la perte de vigilance, le manque d'attention. La facilité de pilotage présente un revers inattendu ou du moins dépassant largement les prévisions.

La cause qui vient ensuite est l'excès de confiance, amplifiée par la grande facilité d'exécution des manoeuvres. Les militaires ne sont pas les seuls touchés par ce mal ! Les commandes de vol ne peuvent rien contre le danger des basses vitesses à trop basse altitude, ni contre le risque d'un retournement engagé trop bas.

Il paraît important de revoir les moyens et les programmes d'entrainement des pilotes d'avions à commandes de vol électroniques afin qu'ils s'adaptent mieux à ces nouvelles machines très sophistiquées, en attendant que nous sachions améliorer ces machines pour qu'elles tiennent compte des faiblesses humaines.

THE ART OF FLYING QUALITIES TESTING

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Basicly, there are two ways of testing the qualities of a piloted aeroplane: Either you do it "by the book" or you do it by experience and intuition.
When I decided to be a test pilot - 25 years ago - I started to learn all the necessary details by means of the available books. Thereby I got a thorough knowledge of most of the related paragraphs. Then I went to the test pilot school and learned to evaluate the flying machines and the practical methods to demonstrate compliance with all the related requirements. The first shock I got was during the final flight examination by Monsieur Plessier, the famous Le Bourget "Jupiter" when I showed him in the air and explained during the debriefing how I confirm lateral stability:
As teached, I lowered one of the wings while maintaining a constant heading with rudder and then giving free the ailerons for self-recovering. He asked me:"Do you think this is really a good method to simulate the recovery after an atmospheric gust?"
"But Sir, this is the way I was taught to do it at your test pilot school."
"Do you believe everything a teacher tells you?"
"But Sir, what shall I believe in?"
"If you want to be accepted as a test pilot and if you want to survive during your career, you must learn to question everything you are told. From now on, your main job will be to discuss and even fight for optimum solutions, this for the sake of flying safety and in the interest of your company to produce good aircraft."
And he was right. - The fights for the best solutions became daily routine, and by the way: steady sideslip is really not a good initial condition to prove lateral stability, it is only an established and accepted method to show compliance with a FAR paragraph.
If you really want to evaluate the roll stability, you need to fly the airplane normally in turbulence or in bad weather and then you know whether it is stable or not and this without questioning maneuvrability. You must be ready to evaluate between these contradictionary qualities of stability and maneuvrability keeping always in mind, the main task the airplane was designed for.

After the test pilot school, I was lucky enough to work for seven years in the Franco-German Alpha-Jet programme at the Dassault flight test center in Istres. My French partner test pilots did not carry the MIL Spec. unter their arms all day long, they preferred to create flying qualities by experience. They had enough experience by flying a new prototype practically every year - from the Alpha-Jet or the Etandard subsonic fighter bombers and the Mercure transport plane, to the different supersonic Mirages fighters and to the various Falcon bussiness jet family members.
There, I learned and after return to Germany I practised this at the Dornier company - not to test aircraft against requirements only, but to:

- fly
- evaluate
- judge
- discuss
- change and
- fly again

in order to create an aircraft which:

- fulfils the operational requirements it was intended for
- has qualities that please the customers and their pilots
- assures a maximum of safety when operated in the normal envelope
- shows ample and clear warnings, should unusual situations appear and
- remains flyable even in emergency situations (e.g. heavy icing conditions, system failures, etc.).

Fortunately I was able to reduce active development testing when the electronic industry started to take control of cockpit design and flight control systems. When they convinced the design engineers that the human pilot is

overstressed and needs the help of computers in the linking between cockpit handles and the flight controls surfaces. Links which were more and more of the electric wire or the fibre optic type only. They also spread the news that sophisticated software only could solve the future airtasks as the limited human intelligence infringes the possibilities to further optimise flight operations to increase profit or deterrence.
Initially, I had problems to understand my hesitation to follow this develoment trend. Today I know that this personal feeling comes from the fact that - at least for civil air transport systems - the industry is taking too fast an unnecessary risk in changing from quasi solid molecular flight control and display systems to soft and fragile electromagnetic or electrooptical variants.
Saying molecular, I mean systems where the interaction between crew and aircraft is either:

- solid (metal or plastic)
- liquid (hydraulic) or
- gaseous (pneumatic)

It is clear to everybody that these molecular systems can fail, as well as electromagnetic systems, but their failures and the subsequent system reaction is predictable. In the case of electromagnetic systems, their types of failures and system reaction are very hard to predict. In addition, they can easily be disturbed in our world of increasing electromagnetic pollution. This pollution and the disturbances are hard to test in every aspect before a new machine is released for public air transport. This should be of great concern to all of us.
Another point of concern is that today many aircraft companies unfortunately have completely separate military and civil aircraft divisions. Due to this fact, the civil products do not benefit enough from the normally more advanced military systems as it was the case in former times. The military experience - including the knowledge of this AGARD group - is not penetrating any more deep enough into civil design works, and here I see an unnecessary flying safety risk, simply due to a lack of communication

Coming back to my reluctance against soft display and flight controls.
When we started the initial definition on the Do 328 cockpit, I was the test pilot in charge. I was very embarrassed how reluctant the civil electronic industry behaved when I addressed vital flying safety considerations concerning the CRT-display presentation, the switchology and the failure modes. - An area of responsibility I really had, when working on military aircraft in close cooperation with the services. Most arguments were countered by the remark "we have discussed all this with x- and y-company, you take this system as it is or you leave it". I found myself in a situation where I had to fear loosing control over the design of my cockpit and I had to give away vital areas of classic test pilot responsibilities.
Parallel to this situation I learned about problems encountered with civil fly-by-wire aircraft by own experience and by aviation magazines. Here some examples:

I flew a big transport aircraft where:

- a pilot can give inputs to the control stick and the computer refuses to follow because a limit is reached - this without advising the pilot clearly of the software decision
  or
- the throttles do not move in the cockpit when engine power is changed by the computer and
- time constants, necessary for turbulent weather operations for autopilot operation, are not removed when flying manually. This results during turbulent weather approaches often in situations where the fast pilot's reaction is blocked by a heavily damped computer action: leading to a severe loss of pilot confidence in the aircraft.

Recently, I read that on another big transport plane, power of all engines was reduced on several occasions to idle after takeoff or in climb due to a false gear down signal.
This list could be continued with at least 10 examples more.
I see here a dramatic gap in aircraft definition which can only be solved by sticking to reliable systems as long as possible and leaving the molecular way only where necessary for safety and for a real reduction in pilot's workload. Electronic flight control and display system as they are designed today do not always fulfil this requirement. To prove this, I quote the German Airline Pilots Organisation "Vereinigung Cockpit" from their September meeting: "The Airbus A 320 is not yet fully developed. The computer systems are helping the pilot only if they are fully operational. In case of failures the crew workload will go well above the one in conventional aircraft."
In my opinion time has come to regain old fashioned team work between test pilots, aircraft design und test engineers - not to forget aircraft mechanics - and the legislative, placing the electronic industry back to its position of real subcontractor. Every system should be designed only according to aircraft type and flying safety orientated specifications written by the aircraft designer. If electronic systems are unavoidable, it would be of great help to the civil side, if the military experience would enter rapidly the brains of civil aircraft designers.

MIL 8785 Revision D is here an excellent example for the future. It should be taken as a basis also for future revisions of civil aircraft certification regulations.

Let me return to conventional flight testing as more than 95 % of all new aircraft types are still treated this way. When I get a new aircraft in my hands - prototype or already certified - I first fly it from takeoff to landing through its normal flight envelope (fast, slow, high and low, straight and crazy); ground handling is included in such a flight. The first flight is the most important one for the overall test pilots evaluation and to determine the area to concentrate on during the following mission. For sales flights, by the way, there is normally only one such flight possible - if too many excuses are needed during this flight the plane will not find its customer.
Already during the second flight the pilot starts to adapt even to the worst qualities because he has the natural tendency to master also the worst machine. This natural tendency to perform must be supressed for a while by a good test pilot keeping in mind the "normal pilot skill".
A normal pilot, by the way, is normally not familiar with the details of the topic "flying qualities". For him, a good aircraft is one which flies like the one he has flown for the last ten years. If he can fly it without thinking too much, then it is a good one - nothing else counts. It remains with the test pilot to discover and describe all the problem areas during initial evaluation. The detail work of "fine tuning" the behaviour of a machine is not very popular within the companies as the results are not clearly visible in dollars and cents because they touch the border of personal feelings and even philosophy. If the certification competition continues as it is practised today, that means every company outdoing all others in shortening more and more the flight test phases, flying qualities optimisations will be very soon a banned topic in aircraft development.
The flight test period is also the time when the conflict with existing airworthiness requirements takes place. Here some examples:

Example no. 1
Civil airworthiness regulations still require the old fashioned speed stability (figure 1), that means that during an increase or decrease of the speed outside the 10 % friction band a positive push or pull action is required by the pilot (figure 2). But looking at supersonic aircraft, this required steady positive stick force is reversed above Mach = 1 but the aircraft still remains perfectly flyable even with an apparent instability according to the books (figure 3).
At M > 1 the actual stability has even increased due to the backtravel of the center of pressure on the $C_l\alpha$-curve as can be seen on figure 4.

The Alpha-Jet fighter/trainer is accepted as one of the nicest flying machines that exist. One of the most praised qualities is the autotrim function of the flight control system when flying between 150 and 450 KIAS. The pilot does not need to touch the trim button because the control surface moves all by itself giving perfect stability but not by the civil books (figures 5 and 6). MIL Spec. is here more advanced than FAR's. Apparently there was a constant dialog between pilots and well experienced scientists who understood the problem and translated it to useful criteria, not calling them "certification requirements" but "decision guidelines" (figure 7 and figure 8).
The term "neutral apparent speed stability" which is used recently could solve a lot of certification problems if adopted to civil regulations and would even help to make better aircraft to fly by the pilots.

Example no. 2
Quite often I have seen aircraft which were perfectly stable according to FAR when stabilized long enough at certain speed intervals, but when accelerating towards higher speeds they were unstable with a definite pull force required to stop them even at V_D. This is often due to free masses within the control linkage forcing the stick forward on acceleration or in a descent and backwards when decelerating or when climbing. I have seen aircraft which were driven progressively into a stall without any selfrecovery tendency. Nobody can stop today a manufacturer delivering such critical airplane to new customers.
A rather funny experience was the reverse free mass action on the Do-24 ATT amphibian which I tested in 1983. The Do-24 flying boat was heavily stable when tested at well stabilized speed intervals (figure 9). But it was famous of being relatively free of the dangerous porpoise tendencies when aerodynamics fight hydrodynamics during speed excursions on the water (figure 10).
Porpoise is normally triggered by out-of-phase pilot inputs. Not so on the Do-24. As you touched the water, the control column was driven forward by the sudden decelleration when water drag started. When waves caught and left the hull changing periodically the water drag, the column was also pumping in the proper manner calming down not only the aircraft but also the skipper. We never found out if this funny detail was one of the secrets of Professor Dornier or whether it was just an inherent quality installed by chance.

Example no. 3
As already mentioned, the Alpha-Jet is one of the finest aircraft I have ever flown, also due to the nearly perfect harmony between aileron and elevator stick forces. These forces are considered light but well adapted, not only for the trainer but also for the low altitude attack mission. It was therefore clear that Dornier when starting the development of the Argentine IA 63 trainer several years ago, saw no need of changing these characteristics. We even bought the hardware, the ARTHUR variable stick force bellcrank, which was explained on figure 5, from Dassault. But already in the development phase a discussion started with Argentine specialists about MIL Spec. level 1 performance. They insisted that the proposed IA 63 stick forces per g did not meet the level 1 as contracted. They had plotted the required stick force gradients as seen on figure 11 creating a form of g-envelope. It was only at that time that we looked at the Alpha-Jet in respect to the MIL Spec levels and we found out that also this pilot-loved machine did not meet this presentation. The discussions were endless and are still going on. We optimized in the meanwhile the IA 63 giving it control forces that suit the envelope, by that giving up the elegance of operations.
I am sure that the founder of this MIL section did not want to create an envelope. But I must say, with all my experience it took me half a day to understand table V (figure 12) together with MIL spec figure 16 (figure 13). Only after plotting stick forces per g over airspeed I had something also normal trained test pilots could understand (figure 14).
Therefore, I urge this group to look for a simpler way of treating this subject in MIL spec and to leave the area below 2 g free for definition by aircraft designers and test personnel.

Example no. 4
Small aeroplanes are certified according to FAR 23 and airworthiness requirement related to aircraft with a takeoff mass equal or below 5700 kg. The Dornier 228 ist one of the airplanes in this category. The initial certification was at the specification limit that means 5700 kg. All documents including the POH were printed and delivered with this limit, until we found a customer in Japan. At that time, a young engineer of the Japanese certification agency refused to certify our aircraft because it did not meet the personal licencing requirments as this operational regulation applies only to aircraft below 5700 kg. We were forced to change all documentation by replacing the number 5700 kg by 5699 kg (figure 15).

From the last two examples we can learn a lot. If a group of specialists gives leading numbers for future regulations it has a big responsibility because once a number is published it will be used often by persons who do not see the difference between an apple and a pear. It may take years and years to change it if experience and progress finds it inadequate. This opinion was confirmed to me at the last SETP meeting in Arles when I had the honour to talk to Mr. Georges Cooper who invented together with Mr. Harper the famous flying quality scale. He explained to us that he never intended, when he first presented his paper to this AGARD panel, to fix his numbers for ever. He only intended to present a basis of discussion. But everybody grabbed the numbers and called them practically "the law for evaluation". And when it was found that in parts the explanations were inadequat nobody was able to change them in the whole western world. So Mr. Cooper himself was addressed again several years later and was asked to revise it.

CONCLUSION

To conclude this paper, I urge you, the scientists, for the sake of progress in aviation, not to give too many numbers and not to lay down regulation if they are not absolutely necessary for the safety of flying. Proposals, recommendations, however, and printed discussions are always welcome and highly desirable. In doing so, you give us, the test pilots and flight test engineers, the possibility to create good flying machines for the specified tasks, flying machines created by experience, thorough knowledge and intuition.
For a test pilot the ultimate goal must always be to "promote air safety by expressing pilot's opinion" - the leading objective of our society (SETP).
In the cockpit we do not need doctors in science, analysing prototypes in flight - what we need are technically orientated pilots creating good airplanes as part of an integrated team of aircraft designers, aircraft mechanics, development and test engineers. The team must be a mixture of elder, experienced, hesitating people and young, progressive and aggressive persons. This mixture gives maximum assurance that the resulting flying machines will be classified as:
"Safe, economic, mission orientated and beautiful to fly".
"Beautiful to fly" is the pilots' way of saying: good flying qualities.

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FIGURES

LONGITUDINAL STABILITY - THE "CIVIL" APPROACH

(a) GENERAL.
A pull must be required to obtain and maintain speeds below the specified trim speed and a push required to obtain and maintain speeds above the specified trim speed. This must be shown at any speed that can be obtained ... except that the control force necessary to maintain a speed differing by less than 10% from the trimmed speed may be supplied by control system friction.

FAR 23 : The stick force must vary with speed so that any substantial speed change results in a stick force clearly perceptible to the pilot.

FAR 25 : The average gradient of the stable slope of the stick force versus speed curve may not be less than 1 pound for each 6 knots

BCAR "K" : The stick force/speed curve shall be positive at all speeds within the required speed ranges except that the control force necessary to maintain a speed differing by less than 10% from the trimmed speed may be supplied by control system friction.

FIGURE 1

LONGITUDINAL STABILITY, THE VIEW OF FAR

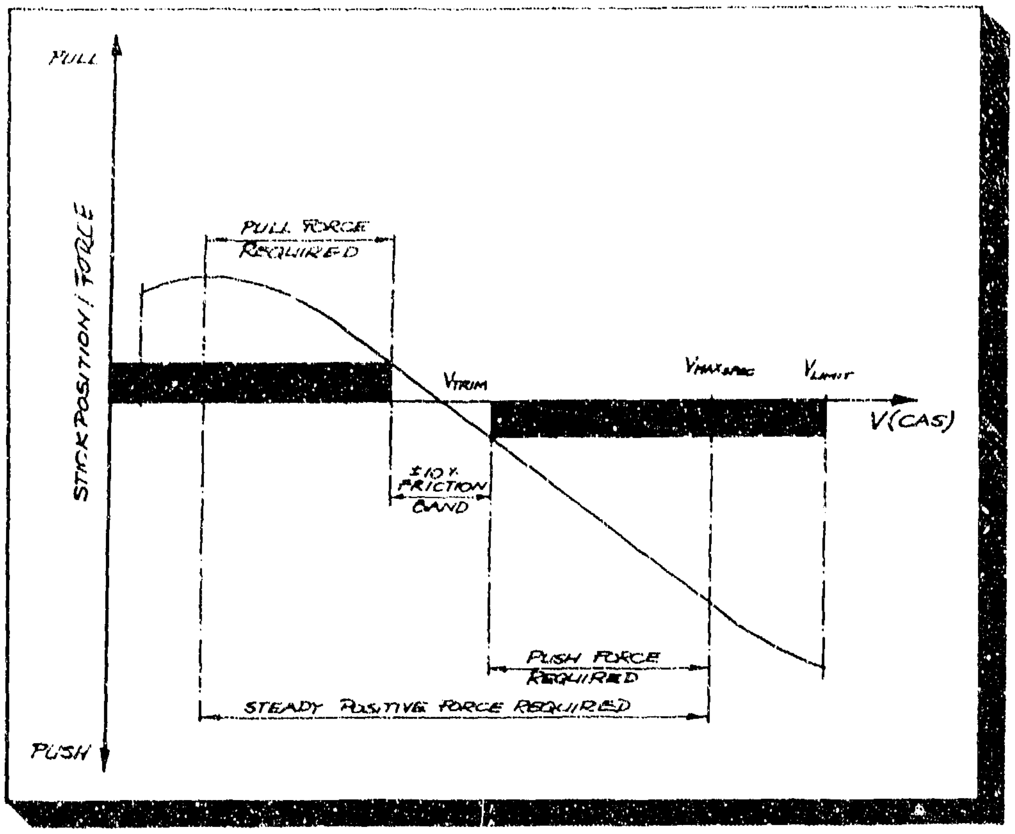

FIGURE 2

LONGITUDINAL STABILITY, A TYPICAL SUPERSONIC FIGHTER

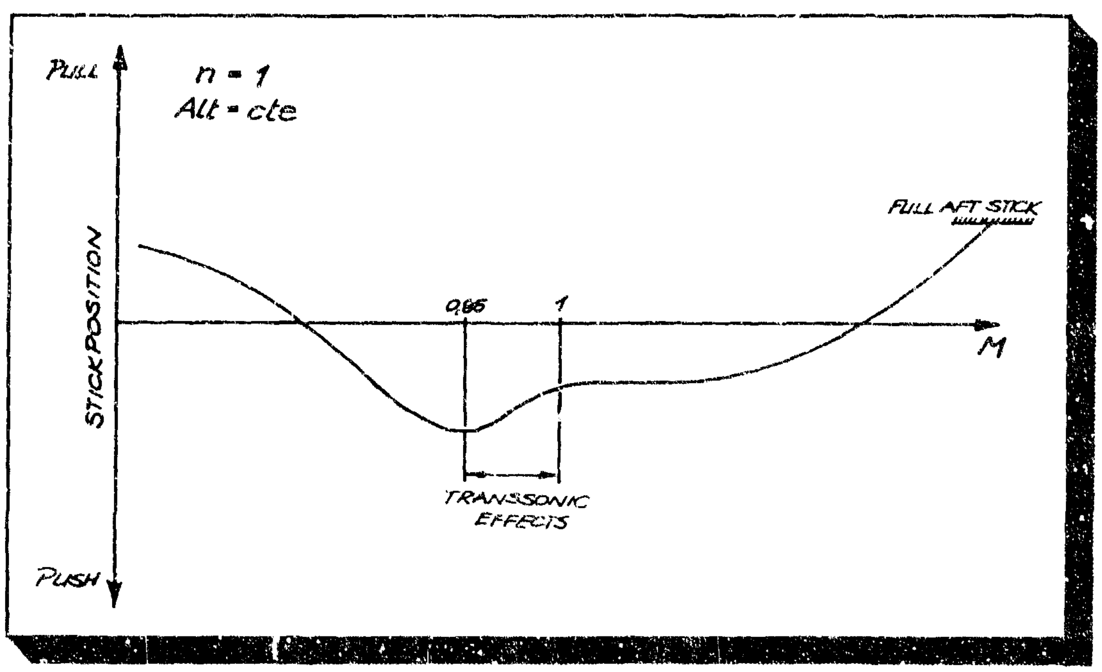

FIGURE 3

LOSS OF ELEVATOR EFFECTIVENESS
AND CENTER OF PRESSURE CHANGE
DURING MACH EXCURSION

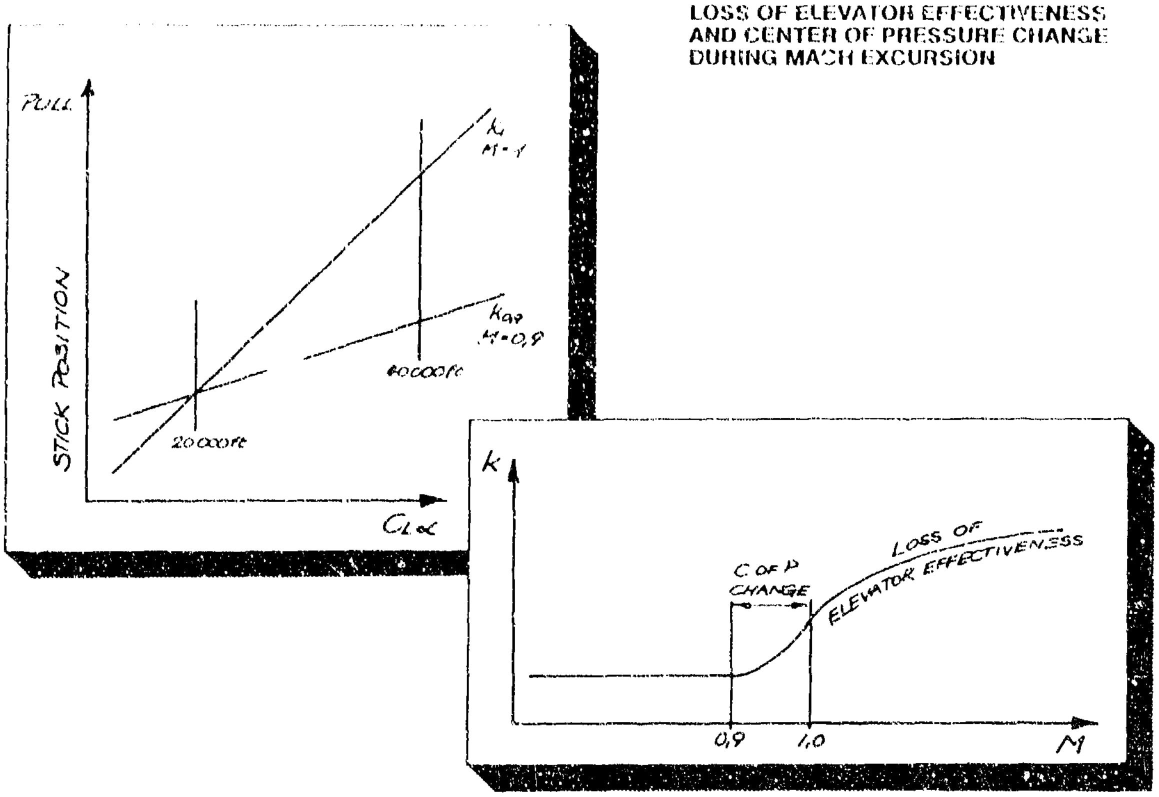

FIGURE 4

ALPHA-JET ELEVATOR CONTROL CINEMATIC

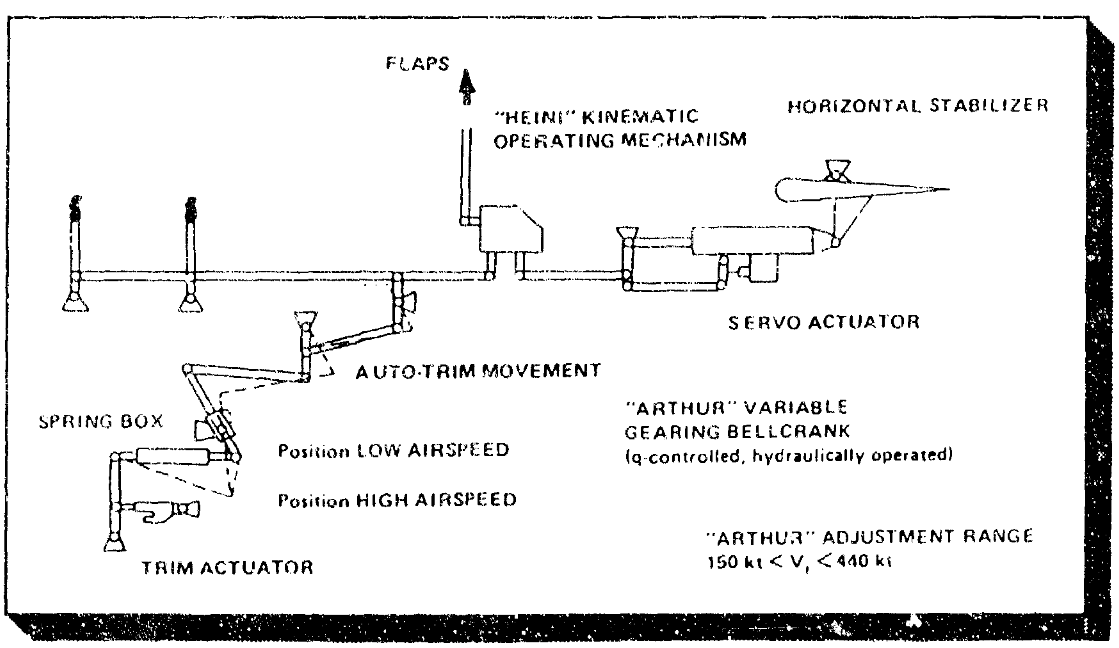

FIGURE 5

ALPHA-JET ELEVATOR CONTROL CHARACTERISTICS INCLUDING AUTOTRIM FUNCTION

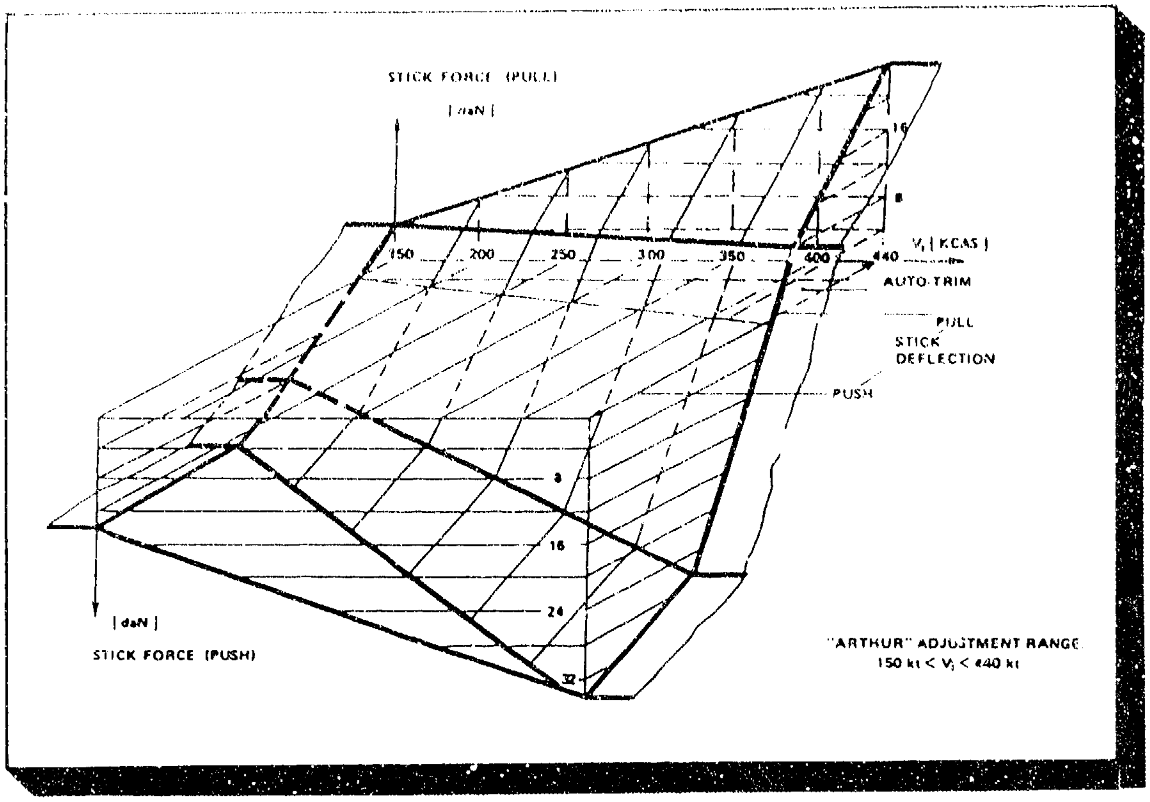

FIGURE 6

LONGITUDINAL STABILITY - THE MIL APPROACH

3.2.1.1 Longitudinal static stability
For levels 1 and 2 there shall be no tendency for airspeed to diverge aperiodically when the airplane ist disturbed from trim with the cockpit controls fixed and with them free. This requirement will be considered satisfied if the variations of pitch control force and pitch control position with airspeed are smooth and local gradients stable, with :

a) Trimmer and throttle controls not moved from the trim settings by the crew and
b) 1g acceleration normal to the flight path, and
c) constant altitude

over a range about the trim speed of ± 15% or ± 50 knots equivalent airspeed, whichever is less ...

3.2.1.1.1 Relaxation in transsonic flight
a) Levels 1 and 2 - For center stick controllers, no local force gradient shall be more unstable than 3 pounds per 0.01 M nor shall the force change exceed 10 pounds in the unstable direction. The corresponding limits for wheel controllers are 5 pounds per 0.01 M and 15 pounds, respectively.
b) Level 3 - For center-stick controllers, no local force gradient shall be more unstable than 6 pounds per 0.01 M nor shall the force ever exceed 20 pounds in the unstable direction. The corresponding limits for wheel controllers are 10 pounds per 0.01 M and 30 pounds respectively.

3.2.1.1.2 Pitch control force variations during rapid speed changes.

3.2.1.3 Flight path stability

FIGURE 7

FLIGHT PATH STABILITY ACCORDING TO MIL SPEC

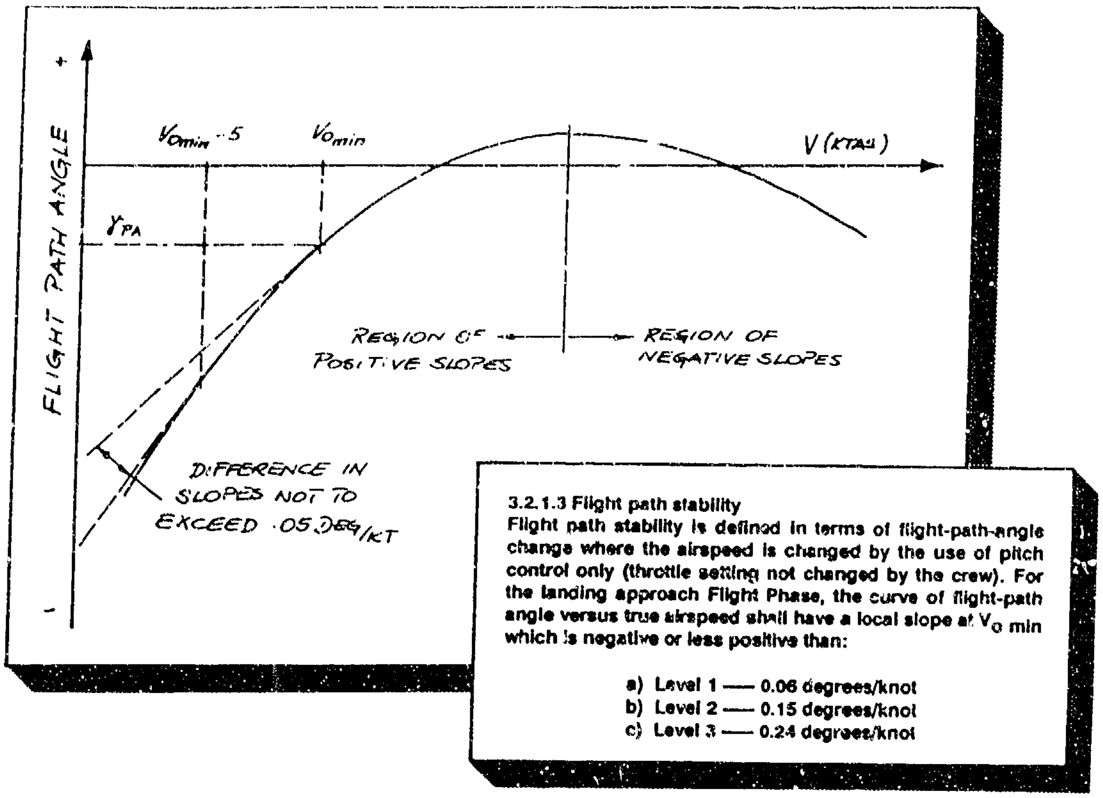

FIGURE 8

LONGITUDINAL STABILITY OF DO 24 ATT AMPHIBIAN

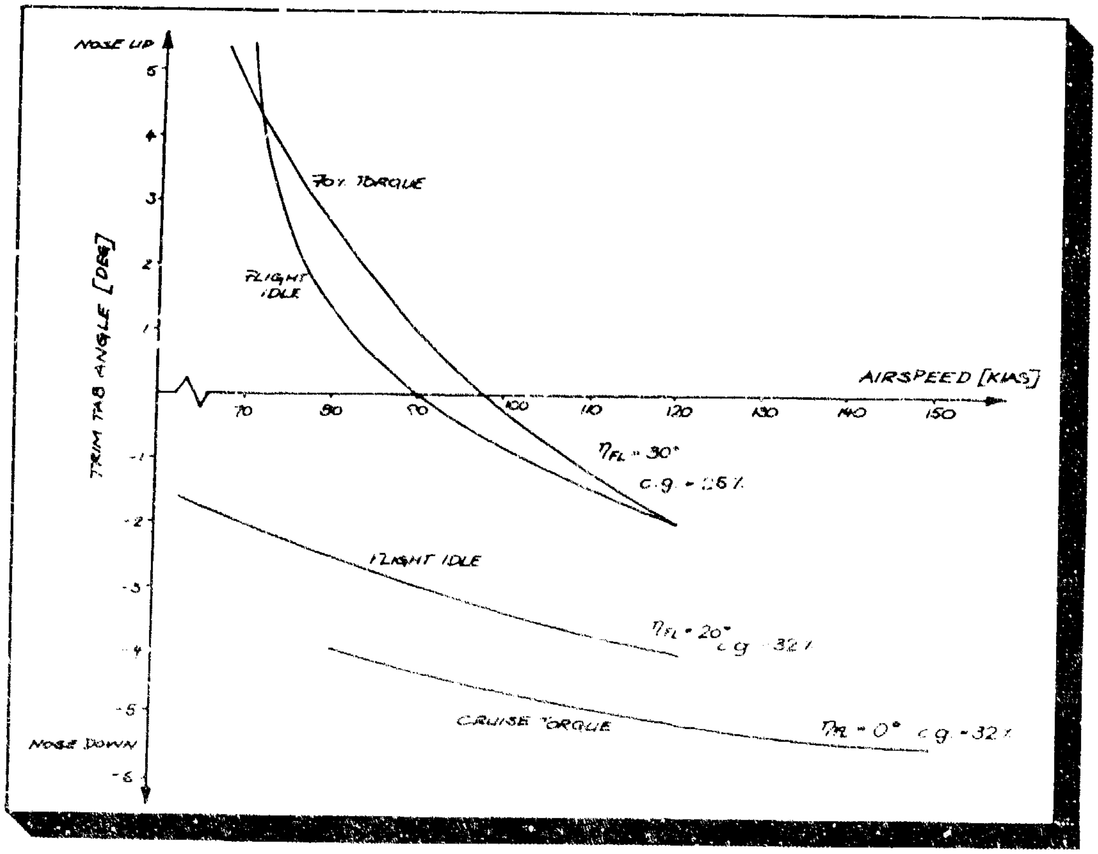

FIGURE 9

DO 24 ATT TRIM ANGLE AND PORPOISE LIMITS

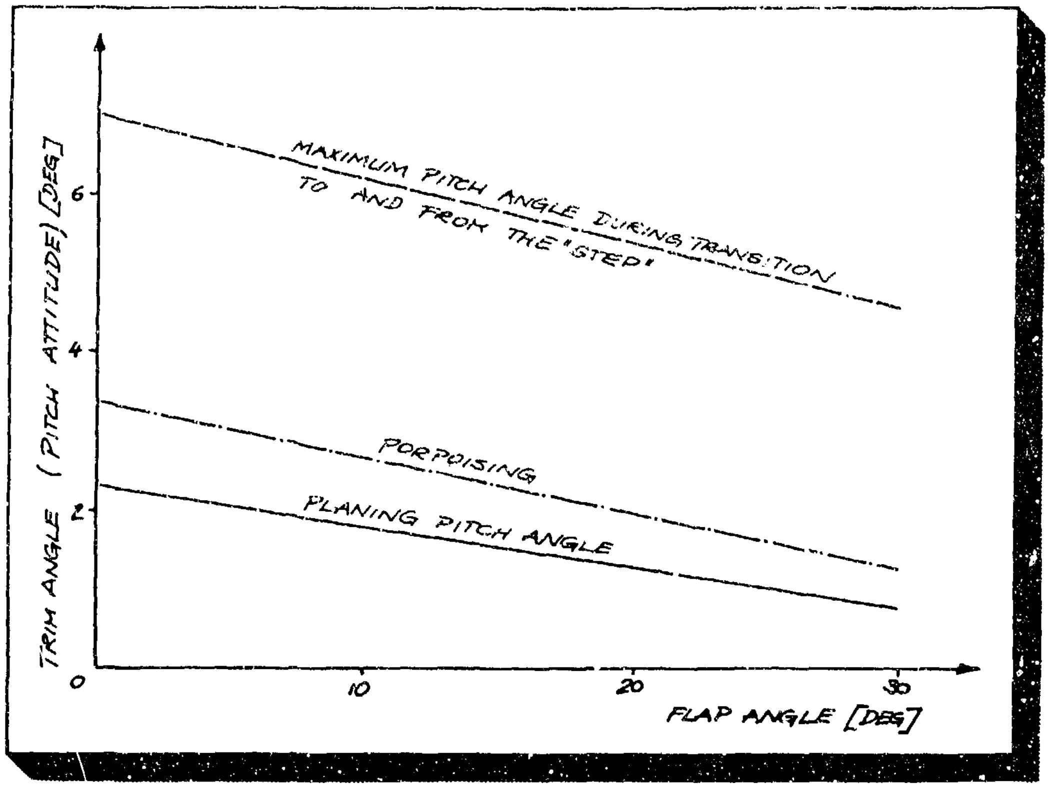

FIGURE 10

STICK FORCE PER g ACCORDING TO MIL SPEC

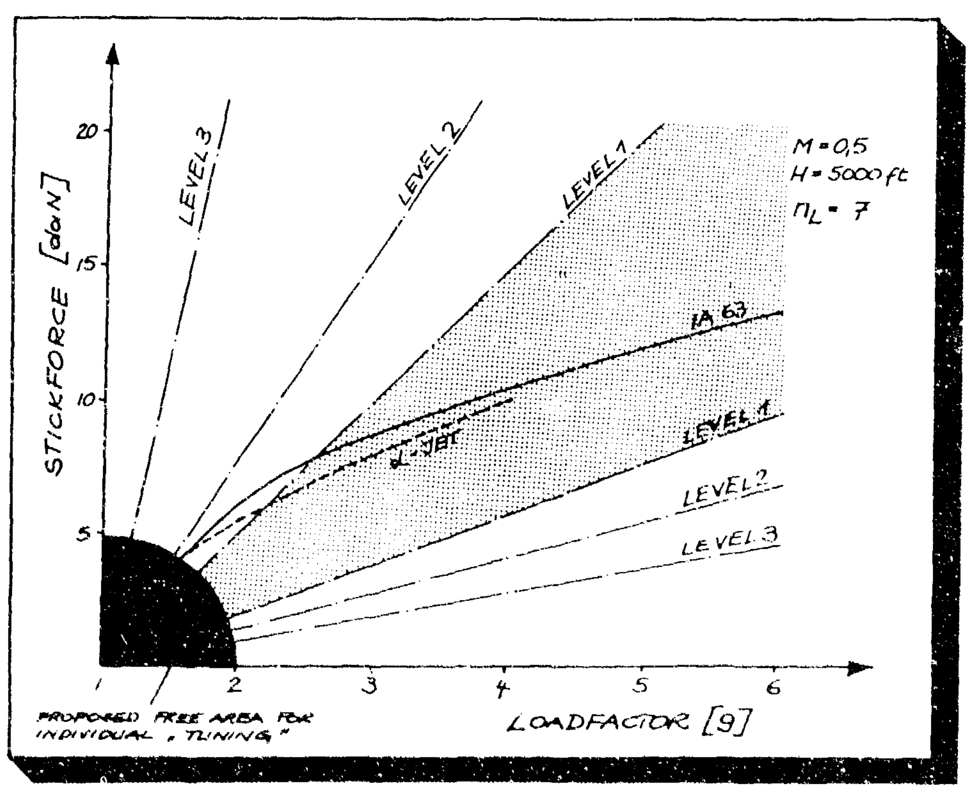

FIGURE 11

STICKFORCE PER g FROM MIL SPEC 8785 C

TABLE V. Pitch manuevering force gradient limits.

Center Stick Controllers

Level	Maximum gradient $(F_s/n)_{max}$, pounds per g	Minimum gradient $(F_s/n)_{min}$, pounds per g
1	240/(n/) but not more than 28.0 nor less than $56/n_L-1$ *	the higher of $21/n_L-1$ and 3.0
2	360/(n/) but not more than 42.5 nor less than $85/n_L-1$	the higher of $18/N_L-1$ and 3.0
3	56.0	the higher of $12/n_L-1$ and 2.0

* For nL <3, (Fs/n)max is 28.0 for Level 1, 42.5 for Level 2.

FIGURE 12

EXAMPLE OF PITCH MANEUVERING FORCE GRADIENTS LIMITS : CENTER-STICK CONTROLLERS, n_L = 7.0

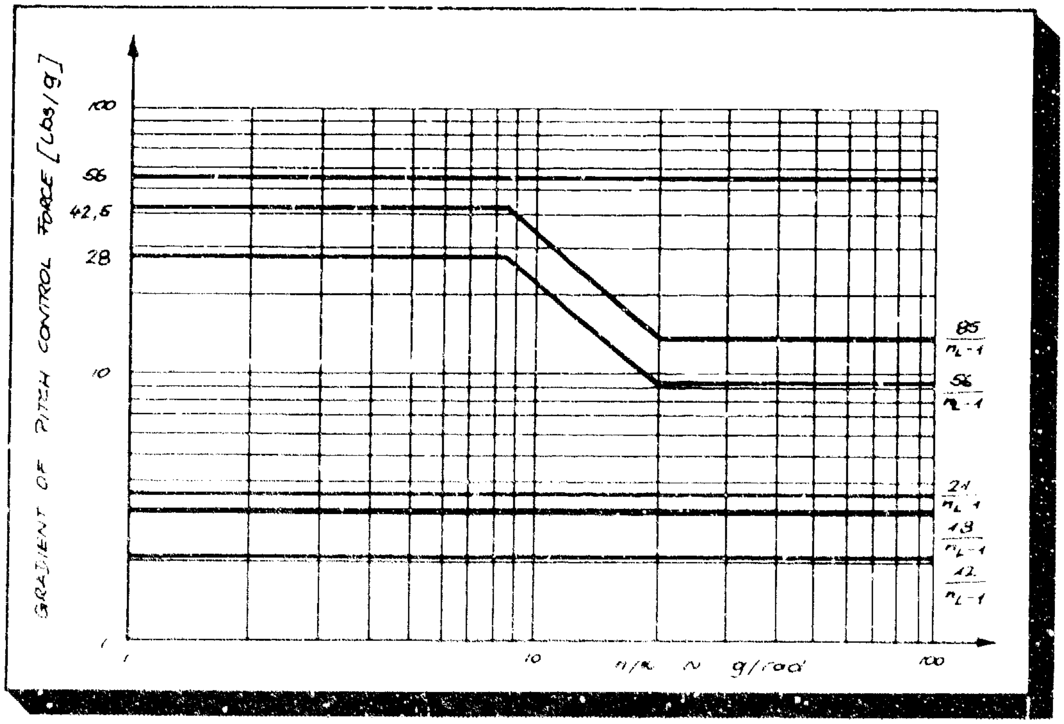

FIGURE 13

EXAMPLE OF STICKFORCE PER g OVER AIRSPEED : CENTER-STICK CONTROLLER, $n_L = 6.0$

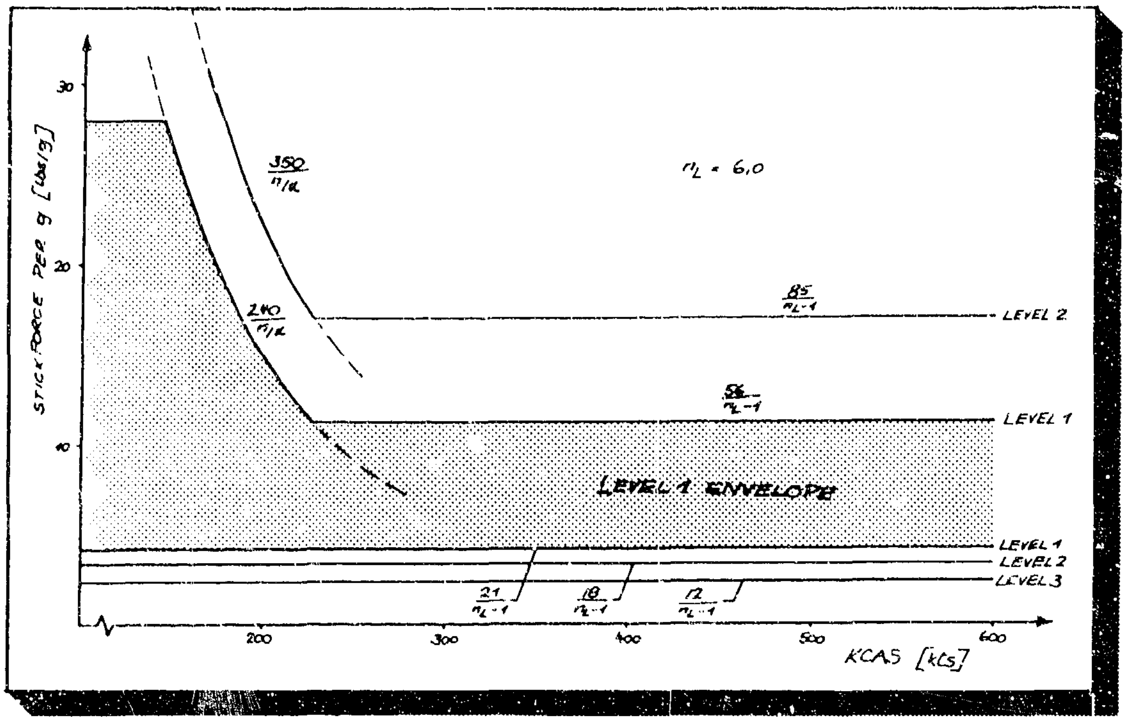

FIGURE 14

DORNIER DO 228 - 200
SECTION 2 - LIMITATIONS
OPERATING LIMITATIONS:
MAXIMUM CERTIFICATED WEIGHTS

Whenever the Pilots Operating Handbook refers to a maximum takeoff and landing weight of 5700 kg (12,566 lbs), it should be read as 5699 kg (12,564 lbs).

FIGURE 15

ADFCS AND NOTAR™: TWO WAYS TO FIX FLYING QUALITIES

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INTRODUCTION

Since the first helicopter flight, "Desirable helicopter flying qualities" has appeared under Webster's definition of the term oxymoron. That is THE paradoxical phrase. Early helicopter designers were limited in their capability to affect flying qualities. There seemed to be little the designer could do to improve the pilot's task until the development of flight control augmentation devices.

These augmentation devices took two forms: Mechanical and Electronic. Of late, electronic has been the preferred method of tailoring helicopter flying qualities due to its inherent flexibility and to the rapid advances in computational power. These advances have recently taken the form of artificial flying qualities generated entirely by digital computers. While this tends to work very well in theory, the basic helicopter has some nonlinear aerodynamic characteristics which even the digital computers of today have trouble dealing with in real time.

As the Project Test Pilot for the McDonnell Douglas Helicopter Company (MDHC) Advanced Digital Flight Control System (ADFCS) and NOTAR™, I have had the pleasure of participating in the design and flight test of two systems which significantly reduce the workload of the helicopter pilot by improving the helicopter's flying qualities. This paper reviews the development, flight test and flying qualities improvements of these two systems. Emphasis is placed on some of the directional control problems faced on the ADFCS program in left sideward flight and the potential for the NOTAR™ system to improve the flying qualities of an advanced, highly augmented rotorcraft.

ADVANCED DIGITAL FLIGHT CONTROL SYSTEM

In 1983, a general question was presented to the then Hughes Helicopters Experimental Test Pilots: "What is required to make a mission effective, single seat scout/attack helicopter." The assumption was made that the primary pilot task was to respond to the volatile battlefield situation and to make complex logical/tactical decisions. On this assumption, any task which diverts the pilot's attention from the primary task decreases his capability to survive. Extending this logic to the flight control system, aircraft control had to become a secondary task, including night and poor usable visual cue conditions. Another assumption was that the level of sensor data available would preclude a sensor coupled flight control system, analogous to an automatic terrain following radar system, for helicopter combat operations. This meant the pilot would still be responsible for the direct control of the aircraft flight path and provide the input to the flight control system. Expanding on these two assumptions it became apparent that the flight control design should minimize the pilot cognitive and physical interaction required to manage control of the aircraft flight path.

McDonnell Douglas Helicopter Company undertook a flight control development program to evolve a system which would make flight control a secondary task for the combat helicopter pilot. The ADFCS control logic would be developed through an iterative process of concept, simulation, and flight test to evolve toward the goal of making flight control a secondary task. MDHC configured a prototype Apache, YAH-64 AV-05 (77-23258), with an experimental flight control system for development. Since that beginning in 1983, 6 years of design and development, 1800 hours of real time piloted simulation, and 270 hours of flight test have gone into the development of the ADFCS. Reference 1 contains a description of the program, the aircraft and the testing. The flight test aircraft is shown in figure 1 and a schematic of the ADFCS installation are shown in figures 2, 3 and 4.

The cockpit controls include the capability to fly the ADFCS with 4+0 sidestick controllers left or right, 3+1 collective, 3+1 pedals or 2+1+1. Controller configuration is a subject unto itself and will not be reviewed as part of this paper. Advanced flight controls are often confused with the flight controller configuration and the two are interrelated but not dependent on one another. The evolution of the flight control logic is described in reference 2. The flight control system logic and flight testing as evolved up to October 1989 will be reviewed here.

FLIGHT CONTROL LOGIC

Distillation of the flight experiences of the experimental test pilot staff resulted in some basic rules for development of the flight control system:

> Flight control workload should not be traded for button workload. Automatic moding would be required because there is no workload advantage if the pilot has to manually

select the flight control mode required. This automatic moding includes flight control logic, gain shaping, and control stick characteristic shaping.

Envelope limiting was required to free the pilot from monitoring aircraft maneuver margins and enhance his ability to use the entire flight envelope.

Some basic automated emergency condition handling was required if the flight control task was to become secondary.

A hierarchy of performance trade-offs was required to allow for an orderly prioritization of aircraft capability and pilot commanded performance.

Starting with a clean sheet of paper, the flight task of the helicopter pilot was defined as flight path management and two environments were seen as critical: An inertially referenced flight environment where the aircraft flight path is relative to the earth and an airframe referenced environment where the aircraft flight path and attitude control is independent of the earth. The inertially referenced environment includes takeoff/landing, hover/low speed, NOE/terrain flight, enroute/navigation (both instrument and visual conditions), and air to ground weapons delivery. The airframe or air referenced environment includes aircraft to aircraft maneuvering (formation or Air Combat Maneuvering), aerobatics, and some airspeed dependent performance conditions. Switching between systems was originally intended to be automatic. During simulation and flight test, this became a difficult proposition and a cyclic switch was added with the switching left exclusively to the pilot. The desire for automatic switching remains but even very complex logic schemes of combined rate/attitude/command have proven inadequate. The inertially referenced flight control system came to be known as Flight Path Vector (FPV) control and the air referenced system Aerobatic control.

FLIGHT PATH VECTOR SYSTEM

The Flight Path Vector control system evolved into a collection of modes which handle different parts of the inertially referenced flight environment. The modes currently include GROUND, HOVER, LOW SPEED, and CRUISE. Switching between these modes is automatic and transparent to the pilot. Some symbology cueing on the Helmet Mounted Display (HMD) shows the active mode and pilot transition between the different mode logic is very natural. A short review of the FPV modes should help clarify how the initial assumptions and requirements were evolved into a full envelope flight control system.

GROUND MODE - The ground mode is used for aircraft start/shutdown, taxi, takeoff and landings. Switching in and out of ground mode is commanded by the combined state of the weight on wheels switches. The development of the ground control mode was one of the more difficult developmental tasks. A helicopter has its full control power available as soon as the rotor is accelerated to operating speed. This necessitates great care in how control inputs are sent to the control surfaces and the sensor suite on the development aircraft presented some unique development problems. During ground operations, the cyclic control of the rotor is used as basically a tip path plane trimming system. No tip path plane position feedback was available on AV-05 so the pilot was responsible for manually positioning the main rotor. Vertical control was used for main rotor collective pitch (power). Directional control was used for heading control. Limited body rate feedbacks were allowed in pitch and roll with rate and heading information used in yaw. Prior to the first engagement of the ground control system on the ground, airborne engagements and a flight envelope was established using the ground control system. This enabled initial checkout and gain settings without the risk of ground contact, flight control induced ground resonance, or system hardover. The system was surprisingly easy to fly and the envelope was expanded from hover to approximately 150 KCAS. Takeoffs and landings were made from different slope conditions and at speeds up to 50 knots. After a development period, the ground to air and air to ground transitions were easily accomplished and no control transients were detectable.

Figure 5 illustrates the control logic for stick command and hold functions of the HOVER, LOW SPEED, and CRUISE modes.

HOVER MODE - Through experimentation a 5 knot hover capture region was established. If the aircraft ground velocity is less than 5 knots and there is no commanded pilot input, the system establishes a hover and holds hover position. This hover capture region makes hover very easy to establish. The pilot has only to drive the commanded FPV to less than 5 knots in his HMD and release the controls. Longitudinal stick commands longitudinal acceleration. Lateral stick commands lateral acceleration. Directional control inputs command yaw rate with heading hold. Vertical control inputs command vertical velocity with radar altitude hold.

LOW SPEED MODE - The low speed region is from 5 knots to 20 knots with the inertial flight path vector aligned laterally within 15 degrees of the aircraft nose; 5-45 knots in the rest of the circle. In this region, longitudinal stick commands longitudinal acceleration and the longitudinal inertial velocity is held in the absence of a command. Lateral stick commands lateral acceleration with inertial velocity hold. Above 20 knots with the nose aligned with the flight path vector and above 45 knots, lateral stick commands roll rate with turn rate hold. Early work was done with a more conventional bank angle hold. This was refined to the current turn rate hold to make the commanded ground track in the turn independent of speed changes. With the turn rate hold, bank angle is varied with speed changes and the ground track is held relatively constant. This reduced the multiple control inputs required for a turning acceleration or deceleration close to ground obstacles. Directional control commands yaw rate with heading hold and the established flight path is held constant. Directional trimmed flight is defined as zero inertial beta up to 60 knots. If the directional command displaces the nose less than 15 degrees from the flight path vector the aircraft returns to trim when the control is released. If the directional command displaces the nose more than 15 degrees from the flight path vector, the new heading is held. The vertical command becomes vertical acceleration with gamma hold. Level flight is a subset with gamma equal to zero and a capture band around gamma = 0 degrees allows easy capture of level flight. This allows descending/decelerating approaches with only a deceleration command once the descent angle is set. In the terrain flight environment, the frequency of vertical commands were reduced because the aircraft can be easily trimmed to follow the general slope of the terrain by placing the flight path vector symbol in the display over the desired point on the terrain. This flight path vector is displayed as a virtual symbol in the helmet mounted display. A recent addition to the level flight hold mechanism has added a radar altitude reference with a relatively long time constant up to 45 knots, which was not intended as a terrain following mode, but does reduce the workload in the NOE environment.

CRUISE MODE - Cruise mode is defined as inertial speed above 20 knots if the nose is within 15 degrees of the flight path vector, 45 knots in any case. Longitudinal stick command remains longitudinal acceleration, inertial speed hold. Lateral stick command remains roll rate with turn rate hold. Directional commands heading to the sideslip limit of the airframe and returns to trim when the command is released. Lateral trim is defined as wings level, inertial ground track hold. Vertical command is vertical acceleration with gamma hold. The level flight subset has a small capture window near gamma = 0 with barometric altitude hold.

The Flight Path Vector system includes the full envelope of the Apache test vehicle. The flight test and demonstration of the system has shown a dramatic workload reduction in the inertially referenced environment. The best indication of the low workload during the demonstration program was the performance of people without any flying experience. With a preflight briefing on the system and some in cockpit coaching, non-pilots could command near envelope limit performance from the aircraft in the course of a one hour demonstration flight. Experienced pilots found that in the normally intense low altitude terrain flight environment relatively few flight control inputs were required and far more time was available for other tasks - the flight control task was secondary and the fewer flight control inputs the better the performance of the aircraft.

AEROBATIC SYSTEM

The Aerobatic system has a much lower level of flight control automation. It is essentially a rate command/attitude hold system. The envelope limiting features of FPV are retained. Classically good conventional flying qualities were built into this rate command/attitude hold system. Additional coordination includes the addition of pitch coordination with bank angle to maintain 1 g flight at all attitudes with the pilot able to modulate g level above or below that reference. A high level of decoupling was retained to allow single axis inputs with single axis response. Heave damping was added to the vertical axis for reduced workload in collective control. Future additions to the aerobatic mode could easily include automation as a function of weapon and sensor status to include features of integrated flight and fire control (IFFC). This system was also designed for full Apache envelope operation and switching between the two modes is smooth. No rate or attitude limits are imposed for switching and the pilots have adapted well to the inflight change in reference systems.

While the original intent of the concept was that these two systems would be completely mission dependent, it has turned out that they can be used in concert to achieve the characteristics desired by the pilot at the moment. That is, as experience with the total system increased, the pilot was more likely to switch between the systems dependent on workload and used the FPV system to unload the flight control task and apply himself to other tasks. It was particularly interesting during the development process that control difficulties in the aerobatic system were usually resolved by switching to the FPV system. For example, flight test of the aerobatic/rate command-attitude hold system sometimes revealed regions of pilot induced oscillation (PIO) or flight control system instabilities and the FPV system was a quick transition to controlled flight.

In my experience, the ADFCS is the most advanced and sophisticated augmentation system implemented on a helicopter. At the opposite extreme of flight control augmentation is the NOTAR™ system applied to an MD500 series helicopter - no electronics, no hydraulics and no mechanical stabilization - yet very desirable flying qualities.

NOTAR™ SYSTEM

NOTAR™ is the McDonnell Douglas Helicopter Company (MDHC) trademark for a system which replaces a conventional helicopter's tail rotor. "Conventional" meaning a single main rotor helicopter with a tail rotor or fenestron.

NOTAR™ DESCRIPTION

The NOTAR™ system design philosophy was to replace the tail rotor with something "as good" and eliminate the exposed auxiliary rotor for safety considerations. A conventional helicopter tail rotor or fenestron system provides directional control for the pilot or flight control system and antitorque to counter the fuselage moment generated by the main rotor torque. Two concepts were postulated as a means to this end:

1. That directional control could be satisfactorily accomplished by an air thruster directed by the pilot.

2. That a significant percentage of the required anti-torque in the low speed regime could be accomplished by producing lift on the tailboom. Lift produced using a low pressure circulation control tailboom as the wing and the main rotor downwash as the free stream.

The research program was structured to validate these concepts and then to integrate them onto a helicopter for evaluation. The flight test vehicle was originally an OH-6A and it has been used throughout the NOTAR™ development process. The system development process is outlined in table 1.

Some of the significant events in the NOTAR™ development program were: validation of the concept that circulation control lift could be produced under the rotor of a hovering helicopter using a low pressure slot system; addition of the second slot; demonstration of an efficient fan design; qualitative flight evaluation; empennage flight test and measurement of the flow field at the rear of the aircraft.

The 13 year research and development program resulted in the configuration shown in figure 6.

The NOTAR™ system has now been integrated into the MD500 series and the MD530N and MD520N are in FAA certification testing. Figure 7 shows the configuration of these aircraft.

NOTAR™ IN PRACTICE

A discussion of the flying qualities and operational impact of a NOTAR™ equipped helicopter must be preceded by a discussion of tail rotor characteristics, particularly thrust required and thrust produced.

The tail rotor operates in the environment at the rear of the helicopter. This exposes it to trees, bushes, fences, wires, people, rocks, ground, and the air. The exposure to the first seven usually result in some change in thrust output and structural damage. The exposure to air can be described as the aerodynamic environment and leads to the following generalized equations.

TAIL ROTOR THRUST REQUIRED = f(MAIN ROTOR TORQUE + DIRECTIONAL CONTROL INPUT)

TAIL ROTOR THRUST PRODUCED = f(DIRECTIONAL CONTROL INPUT + RELATIVE WIND AZIMUTH AND VELOCITY + YAW RATE + YAW ACCELERATION + AIR QUALITY + WIND AND MAIN ROTOR RELATIVE POSITION)

A comparison to the same equations for NOTAR™ provides some insight into the relative flying qualities.

NOTAR™ THRUST REQUIRED = f(MAIN ROTOR TORQUE + DIRECTIONAL CONTROL INPUT)

NOTAR™ THRUST PRODUCED = f(MAIN ROTOR TORQUE + DIRECTIONAL CONTROL INPUT)

Figure 8 shows some of the influences on tail rotor thrust.

The directional control task in trimmed flight is to make thrust produced equal thrust required. For NOTAR™ the thrust required and the thrust produced are functions of the same parameters, main rotor torque and directional control input. There are two equations in two variables and the solution for the pilot is very simple. The tail rotor problem is far more complex, two equations in eight variables, and it's surprising the pilot can solve it at all. The impact of these simple generalizations on pilot workload is significant. The developmental and certification flight testing of the MD500N series has demonstrated the following characteristics with a mechanical control system and no hydraulic boost or artificial stability augmentation.

Power changes in low speed flight do not require pedal inputs to compensate for increased or decreased main rotor torque as in a conventional helicopter. This reduces pilot workload and leads to a reduced total power required for maneuvering flight. This can be easily seen using a helicopter quick stop maneuver as an illustration. With a conventional anti-torque system, the large power increase required at the end of a quick stop must be accompanied by a large increase in tail rotor thrust which reduces the power available to the main rotor to control sink rate and establish a hover. Contrast this to the situation with NOTAR™. The large power increase to complete the quick stop does not require a corresponding directional control input and the horsepower required for anti-torque is the same as required for a stable hover. This provides an increase in power available to the main rotor to arrest a sink rate or to maneuver.

The pilot directional control workload to hover and control yaw rate is not impacted by relative wind velocity, direction or gust spread. The force from the tail rotor is a function of these external factors while the force from the NOTAR™ thruster is not affected by them. Any change in the thrust of the anti-torque system must be compensated for by the pilot to balance the moment equation. This is particularly significant when considering relative wind/motion which induces vortex ring state at the tail rotor or loss of tail rotor effectiveness/stall. We have been unable to induce any similar effect with the NOTAR™ system.

Yaw rate and acceleration do not impact the NOTAR™ thruster force. Conventional anti-torque systems become very difficult to control at yaw rates above 60 degrees per second and large control inputs at these rates can produce loss of directional control and large torque fluctuations. NOTAR™ has demonstrated useful yaw rates in excess of 120 degrees per second with full throw pedal reversals and very linear control response. This useful yaw rate was achieved with 100 degree per second per second yaw acceleration and 40 percent per second control inputs. This control capability has also been demonstrated in 30-40 knot winds.

Current generation military helicopters are specified to be capable of 45 knots crosswind and downwind hover. To meet this specification, the designer must include a large, powerful tail rotor. For clearance, the large tail rotor must be mounted in a high position above the tailboom on a pylon. This combination of powerful tail rotor and mounting above the centerline of the boom introduces structural problems which are common to most modern helicopters. The structural problems can be identified as tailboom torsion, tailboom bending, and tail rotor hub strain due to precession flapping. These problems are aggravated with high roll rates or large, rapid pedal inputs. Typically, the pilot must be prevented from making large, rapid pedal inputs by flight manual restriction or tail rotor control rate damping. The incidence of structural damage is increased during aerobatic maneuvering. During structural testing with NOTAR™, we have not encountered any structural limitations associated with the size or rate of pedal input. This includes full pedal reversals in less than .3 seconds at 120 degrees per second yaw rate. The implication for the helicopter pilot is significant when combined with the linear control response in these conditions. The NOTAR™ pilot can make large, high frequency directional control inputs and maintain precise aircraft control enabling him to safely perform maneuvers well outside the safe envelope for a tail rotor. The operational impact is significant. The structural flight test for the MD-500N included full aerobatic maneuvers to qualify the aircraft for aerobatic flight. This testing was very successful and the third production prototype demonstrated its aerobatic prowess at the Farnborough Air Show.

Downwind hover in conventional helicopters has been accompanied by increases in torque, vibration level and workload. This has forced the helicopter pilot to be very aware of the wind direction in all his low altitude/low speed tasks. In most situations, the penalty for inattention was degraded flight control at best, loss of aircraft control or inadequate power available at worst. Good pilot technique to overcome these problems has been to find targets and landing zones into the relative wind - not always practical or possible. NOTAR™ has freed the helicopter pilot from this slavery to the wind direction in the same manner that tandem, coaxial, and syncopters have in the past. The improvement over those configurations is the retention of excellent yaw maneuverability/agility.

The helicopter is aerodynamically and dynamically coupled in all axes. This has the effect of requiring secondary control inputs in three axes for every primary control input. All the experienced helicopter pilots who have flown NOTAR™ have commented that the workload reduction extends to all control axes. This synergism of reduced workload is a result of reducing the pedal workload and thereby eliminating the corresponding inputs in pitch, roll, and yaw, in addition to reducing the required pedal input to compensate for inputs in the other axes.

Implied in the above workload discussion is the elimination of interference effects between main and tail rotors. The downwind hover effect of tail rotor vortices impinging on the main rotor is one area of impact. Main rotor vortices effect on tail rotor performance must also be considered.

All these workload effects are significant to the operation of the helicopter. Digital flight control could enhance the NOTAR™ system by optimizing the configuration for the flight regime. The current system configuration is fixed throughout the flight envelope but only has a significant impact on the flying qualities of the helicopter in the low speed environment. A digital control system and flexible configuration could easily optimize the system configuration for the each part of the flight envelope. Conversely, the NOTAR™ system characteristics in the low speed environment can greatly ease the job of the helicopter digital flight control designer.

Conventional helicopters have a very nonlinear response region in crosswind conditions; sideward flight to the left for counter clockwise main rotor rotation (U.S., British, German, Italian) and to the right for clockwise main rotor rotation (French, Russian). Tail rotor thrust can change dramatically and be very nonlinear in these wind conditions. The vortices from the tail rotor can also aggravate main rotor response by interference effects in these same conditions. This condition is usually self correcting in that; if nothing is done, the helicopter will yaw to a new relative wind azimuth and take the tail rotor out of this difficulty. When the pilot, ground obstacles, or a flight control system interfere with this natural response, the problem is magnified and sometimes catastrophic. In the case of a digital flight control system, any flight regime in which the aircraft response becomes nonlinear or the sign of the response changes for a given control input special design considerations are required. The development of the ADFCS on AV-05 exposed some of these problems. In left sideward flight at approximately 40 knots, the AH-64 enters a region of nonlinear control response marked by a tail rotor control sense reversal. Pilots can easily adapt to this effect and the aircraft is easily controlled by the pilot in 40 knot crosswinds. The digital flight control system had more difficulty. Even after gain and system tuning a pitch, roll, and yaw oscillation remains divergent in this flight region. Figure 9 shows this aircraft response in the form of attitude-time history traces.

Figure 10 is a comparable time history of the basic aircraft with pilot control.

The NOTAR™ system has the potential to eliminate this problem for the flight control designer. Development of the flight control system in the right sideward flight regime remains incomplete. With design and flight test effort, the digital flight control system has the potential to eliminate the instability shown here; however, a software fix for this anomaly in the helicopter's flight envelope may not be as good a solution as an aerodynamic/configuration change. This is intended as one illustration of a flight condition which does not respond easily to digital flight control augmentation. There are many others, in both helicopter and fixed wing aircraft, which involve regions of the flight envelope that cannot be modeled accurately with a second order differential equation. In spite of the power of the digital computer, the project/program manager must continue to view the development of new aircraft as a big picture and pursue the entire range of design solutions for the best combination to fulfill the aircraft's mission objectives.

CONCLUSION

By combining the capabilities of all the design tools, aircraft potential is enhanced. An excellent example in the fixed wing community is the trend to reducing the static stability of modern aircraft to enhance maneuver capability and/or performance. It has been used as a design technique in the fighter mission to increase agility and in the transport mission to increase cruise performance. These designs depend on electronic augmentation for acceptable handling qualities. Structural design of large wings has been modified to reduce weight by using flight control to reduce g and gust loading on the outboard wing sections.

This synergism of design is only possible through open minded pursuit of good total design solutions. The helicopter community can certainly benefit from this approach. Advancement of aircraft design can be easily stagnated by imposing requirements/restrictions on the designers through over-specification. Specifications tend to lag design innovation because they must be based on existing knowledge and technology. The specifications should ease the way of design innovation and not restrict it. Design innovation can be restricted directly by over-specification and indirectly by program management perception of penalties for not meeting the letter of a specification. This is particularly true in competitively bid programs.

In these instances, the entire aircraft design team must be open to solutions outside their areas of expertise. Good solutions, like medical cures, must treat the cause of the symptoms not just the symptoms.

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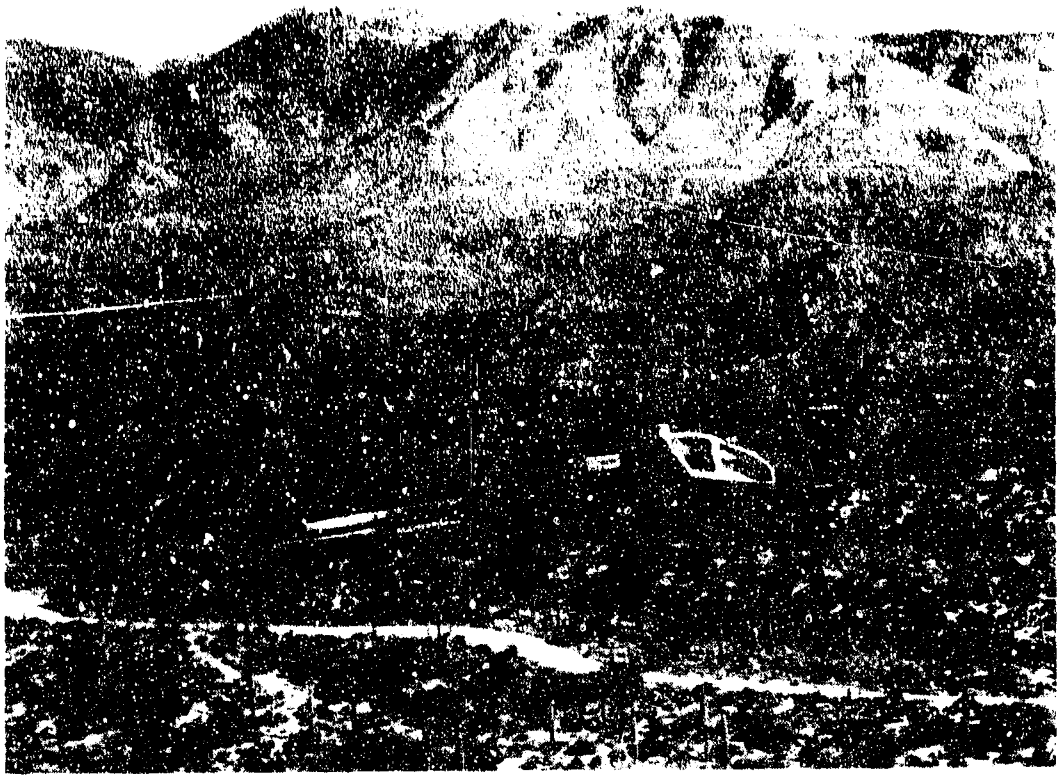
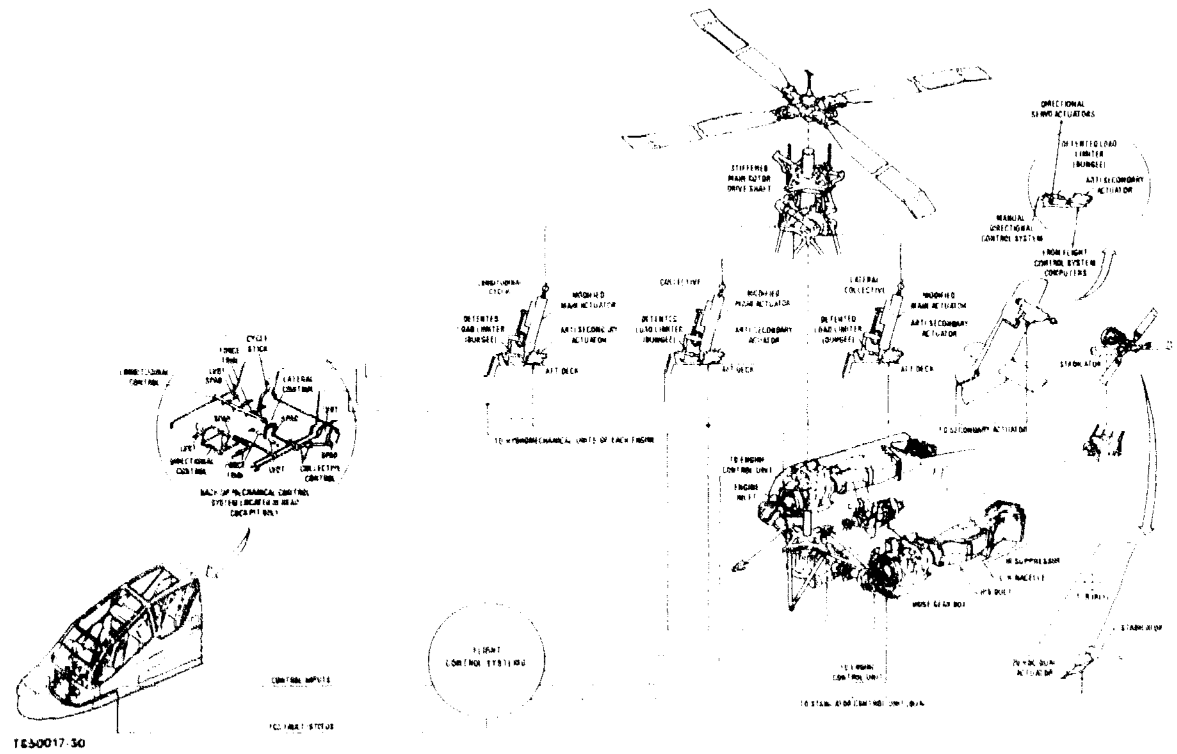
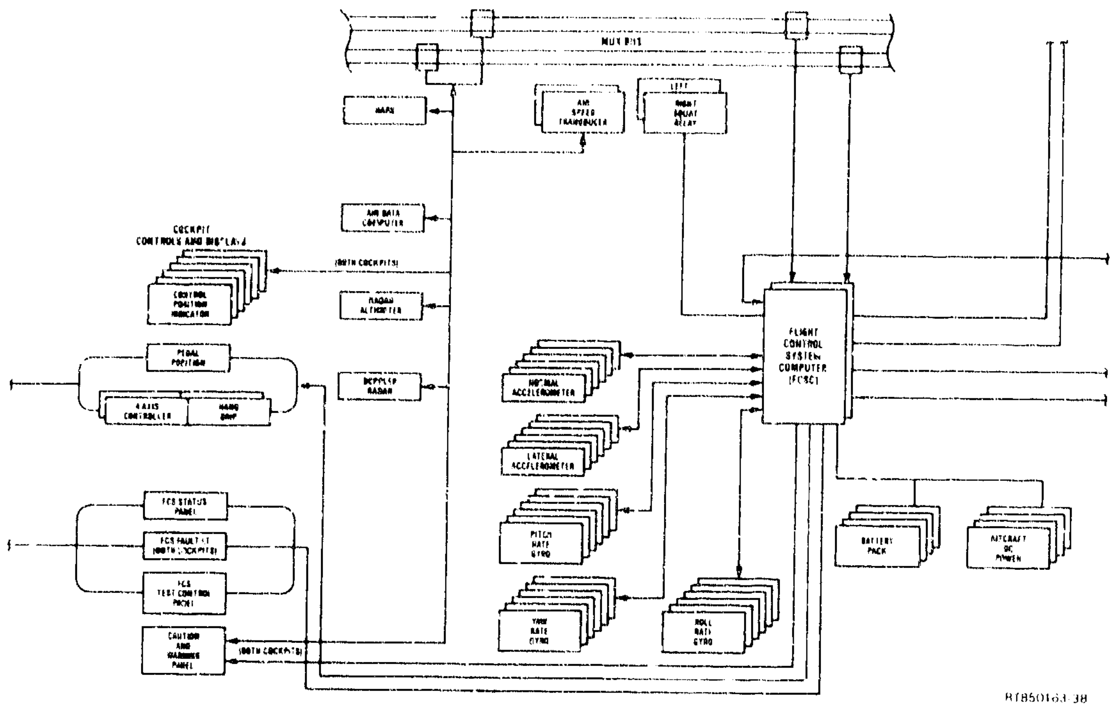


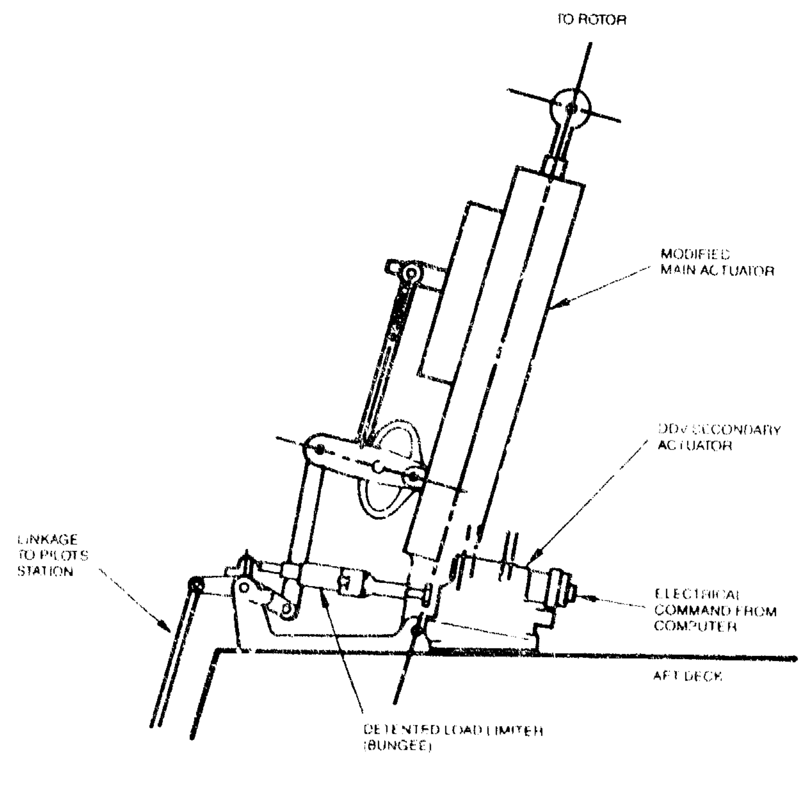
AV-05 FLIGHT TEST AIRCRAFT
FIGURE 1



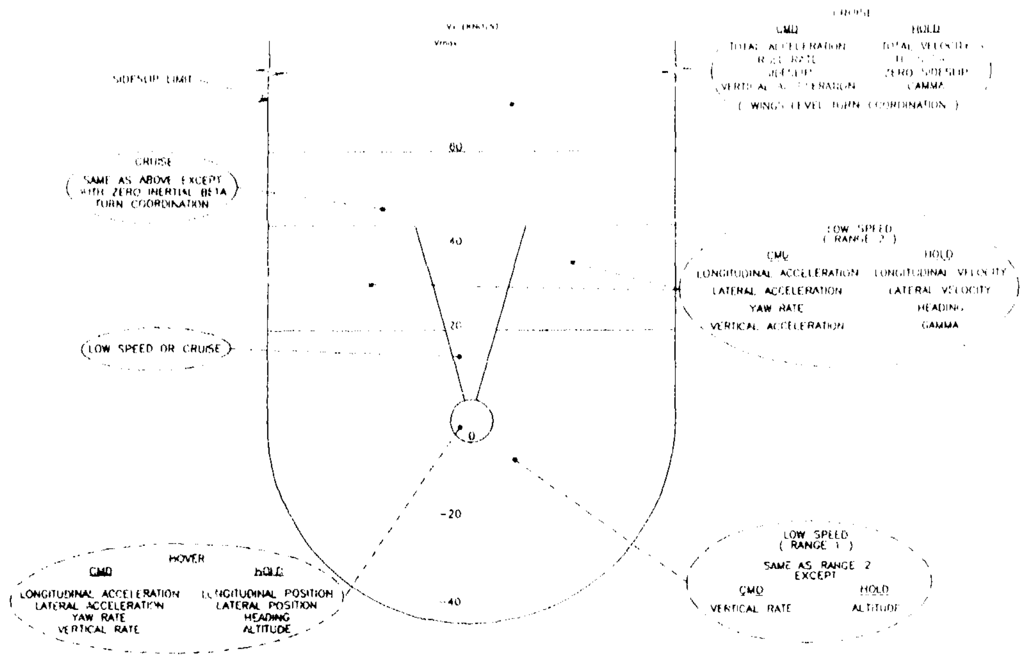
SCHEMATIC OF ADFCS INSTALLATION
FIGURE 2



FLIGHT CONTROL SYSTEM
FIGURE 3



ACTUATION SCHEMATIC
FIGURE 4



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POLAR PLOT OF SPEED/AZIMUTH CONTROL LOGIC
FIGURE 5

TABLE 1
NOTAR™ TEST/ANALYSES HISTORY

AUG 1976 — BASIC DATA GENERATION — MDHC WHIRL TOWER
- DEC 1977 — FIRST CONCEPTUAL FLIGHT
- APR 1980 — NOTAR PATENT ISSUED
- SEP 1981 — GROUND TESTS — FAN/THRUSTER/THRUSTER TRANSIENT RESPONSE

DEC 1981 — FIRST TOTAL SYSTEM FLIGHT, OH-6A
- SEP 1982 — SIMULATION — FLIGHT SIMULATION ON FLIGHT SIMULATOR FOR ADVANCED AIRCRAFT AT AMES

MAY 1983 — GOVERNMENT PILOT EVALUATION
- MAY 1983 — USAAVRADCOM TECHNICAL REPORT
- DEC 1983 — AERODYNAMIC PANEL MODELS
- FEB 1984 — TIEDOWN TESTS — AIRCRAFT ON TOWER TO SIMULATE OUT OF GROUND EFFECT HOVER
- AUG 1984 — NEW FAN DESIGNED, INLET MODS
- SEP 1985 — FAN/STATOR GROUND TESTS

OCT 1985 — WATER TANK TESTING
- OCT 1985 — WIND TUNNEL TESTS
- DEC 1985 — GROUND TEST WITH NEW FAN/INLET AND MORE POWERFUL ENGINE

MAR 1986 — SECOND SLOT FLIGHT TEST

MAR 1987 — GOVERNMENT PILOT EVALUATION

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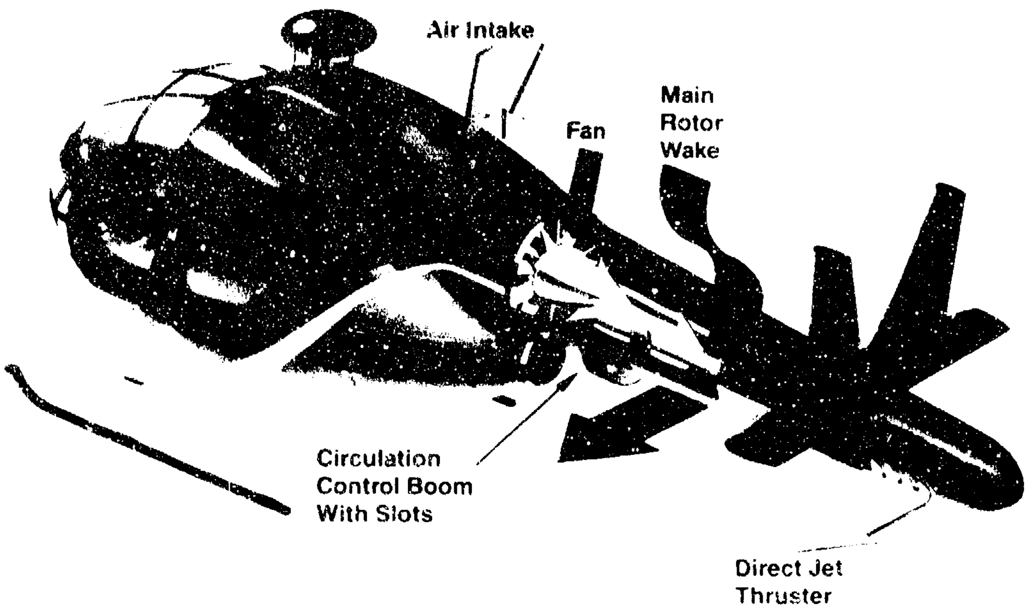

CUTAWAY VIEW OF NOTAR™ DEMONSTRATOR
FIGURE 6

MD500N
FIGURE 7

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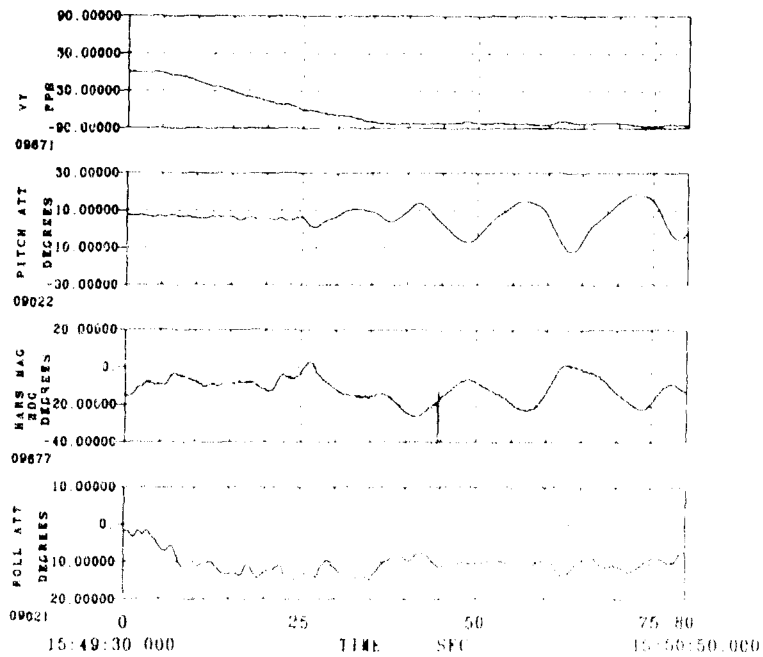
INFLUENCES ON TAIL ROTOR THRUST
FIGURE 8



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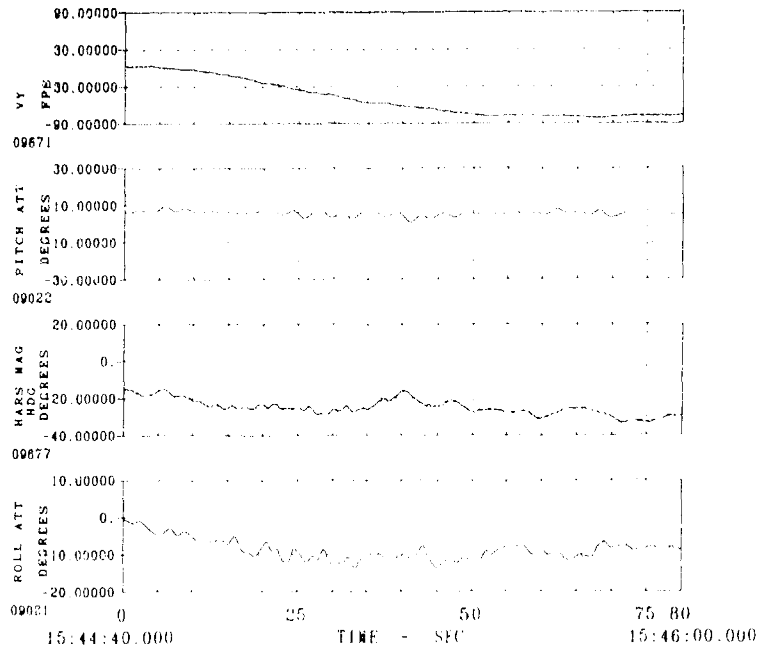
TIME HISTORY OF AV-05 IN LEFT SIDEWARD FLIGHT
FIGURE 9



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TIME HISTORY OF AH-64 WITH PILOT CONTROL
FIGURE 10

QUALITES DE VOL LATERAL D'UN AVION DE TRANSPORT CIVIL EQUIPE DE COMMANDES DE VOL ELECTRIQUES. EXPERIENCE DE L'AIRBUS A320.

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Résumé.

Les commandes de vol électriques, dans un environnement de calculateurs numériques, permettent d'élaborer des lois de pilotage manuel sophistiquées dont le but principal est d'améliorer les qualités de vol d'un avion, en particulier au niveau stabilité, pilotage, et protections du domaine de vol. Cependant, les degrés de liberté effectifs dans la conception de ces lois sont limités par des contraintes physiques aussi bien que humaines, puisqu'il s'agit de pilotage manuel.

Sur l'Airbus A320, les calculateurs de commandes de vol ont accès aux informations anémométriques et inertielles ainsi qu'à d'autres données caractéristiques de l'état de l'avion, et peuvent asservir toutes les gouvernes. Ceci a permis de réaliser des lois de pilotage répondant à des objectifs de pilotage simples et adaptés aux contraintes rencontrées. Des résultats d'essais en vol seront présentés pour mettre en évidence les points caractéristiques des qualités de vol latéral ainsi obtenues.

Nous évoquerons pour conclure les possibilités d'amélioration de la disponibilité de ces lois évoluées par une meilleure tolérance aux pannes des systèmes embarqués, et en conséquence le rôle potentiel qu'elles peuvent jouer dans l'optimisation du dimensionnement des avions futurs

0. Notations

β, angle de dérapage, (°);
ny, facteur de charge latéral, (g);
p_1, vitesse de roulis, (°/s);
r_1, vitesse de lacet, (°/s).
ϕ_1, angle d'assiette latérale, (°);
δp, commande des gouvernes alaires, ailerons et spoilers, (°);
δr, commande de gouverne de direction, (°);
(dx/dt), dérivée de x par rapport au temps;
ξ, facteur d'amortissement d'un mode;
ω, pulsation d'un mode, en rd/s,
ϵ, symbole mathématique d'appartenance;
CDVE, Commandes De Vol Electriques;
Vc, Vitesse air conventionnelle;
conf, configuration des becs/volets;
PIO, pompage piloté (Pilot Induced Oscillations);
ADC, centrale anémométrique (Air Data Computer);
IRS, centrale inertielle (Inertial Reference System);

1. Généralités sur la conception de lois de pilotage.

a) Contexte.

Le contexte nécessaire à des lois de pilotage évoluées, et donc à des qualités de vol améliorées (par rapport à l'avion "naturel"), est schématisé fig. 0: des capteurs électriques sur les organes de pilotage (manche et palonnier pour le pilotage latéral) permettent de transformer ces braquages en objectifs de pilotage; ceux-ci sont ensuite comparés à l'état réel de l'avion, mesuré par des capteurs anémométriques, inertiels, etc... De cette comparaison, on déduit l'ordre à envoyer aux asservissements des gouvernes de roulis (ailerons & spoilers) et de lacet (direction).

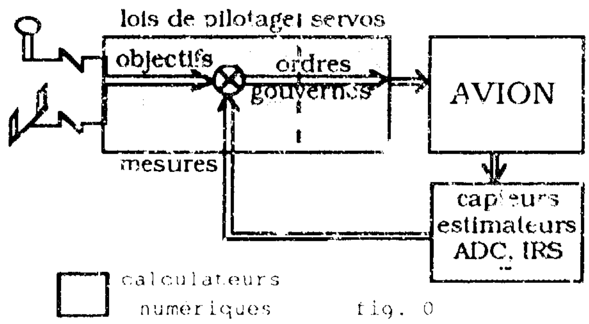


fig. 0

b) possibilités.

Cette structure permet d'utiliser les principes de contrôle actif généralisé (CAG) autrement dit les techniques d'automatique, dont la caractéristique de commander des braquages de gouvernes fonctions de l'ordre pilote, bien sur, mais aussi de l'état de l'avion (ce point est important, car il peut être nécessaire de fournir au pilote des informations sur la position des gouvernes).

Il existe, sur la plupart des avions actuels, des dispositifs permettant d'en augmenter la stabilité en pilotage manuel, en particulier des stabilisateurs de lacet pour le mode de roulis hollandais. On peut aller plus loin: sur un avion de transport civil, on dispose maintenant d'informations complètes sur les 4 variables représentatives de l'état de l'avion (le dérapage, β, ou le facteur de charge latéral, ny, la vitesse de roulis, p_1, la vitesse de lacet, r_1, et l'assiette latérale, ϕ_1), et de plus les modèles fournis soit par les calculs ou essais en soufflerie, soit par l'identification, sont suffisamment représentatifs du mouvement de l'avion. Comme l'on dispose de deux commandes indépendantes, en roulis, δp, et en lacet, δr, l'application des principes de l'automatique nous assure que l'on peut contrôler les quatre modes latéraux de l'avion (le roulis hollandais, composé de deux modes complexes conjugués, en général relativement mal amorti, le mode de roulis pur, mode réel assez rapide, et le mode spiral, mode réel proche de 0, et donc très lentement convergent ou divergent); de plus il reste certains degrés de liberté pour gérer d'autres objectifs concernant le comportement de l'avion ou l'activité des gouvernes: en effet, une gouverne est en théorie suffisante pour contrôler autant de modes que l'on a de mesures indépendantes. Différentes méthodes de l'automatique linéaire permettent de gérer ces degrés de liberté suivant les objectifs choisis:

- contrôle optimal, si l'on veut minimiser un critère, généralement représentatif d'un compromis performance/coût,
- placement de structure propre (modes et vecteurs propres) pour moduler l'impact de certains modes sur certaines variables d'état ou sur les gouvernes, etc...

L'augmentation de stabilité ainsi obtenue améliore donc les qualités de vol, et contribue aussi à la sécurité de l'avion: sur une perturbation telle que rafale ou panne moteur, une stabilité spirale importante permettra d'assurer (dans les limites des gouvernes disponibles évidemment) que l'avion restera stable, même sans réaction du pilote, ce qui n'est pas le cas sur les avions conventionnels.

Bien sur, le pilote reste rarement inactif en commandes de vol manuelles, et le pilotage constitue un autre aspect des qualités de vol, relié toutefois au précédent par le fait qu'un avion stable est plus facile à piloter. Les commandes de vol électriques dans leur contexte de calculateurs (appelées CDVE en abrégé) permettent de réduire notablement la charge de travail de pilotage, en adoptant un pilotage par objectifs, rejoignant les concepts de pilotage automatique où le pilote représenterait la "grande boucle", qui gère les objectifs, et les CDVE la "petite boucle" qui réalise ces objectifs, avec une réponse adaptée à la grande boucle.

Pour augmenter la sécurité et le confort de pilotage, il est intéressant d'introduire des protections contre les sorties potentiellement dangereuses du domaine de vol normal. Le principal avantage de tels dispositifs est de permettre au pilote d'agir très rapidement et franchement, puisqu'il sait que son action ne risque pas de mettre son avion dans une situation critique.

S'il est possible de modifier ainsi de façon importante les qualités de vol d'un avion, on se heurte vite à différent types de contraintes limitant les degrés de liberté effectifs dans la conception des lois de pilotage.

La première contrainte concerne les entrées des lois de pilotage, c'est-à-dire les capteurs. En effet, supposons le mouvement de l'avion bien identifié, et modélisé par exemple par une équation différentielle du type:

$$(dx/dt) = A(x) + B(u)$$

où $x=[\beta, p_1, r_1, \phi_1]^t$ est le vecteur d'état

et $u=[\delta p, \delta r]^t$. le vecteur des commandes.

Dans ce modèle, la matrice A représente la dynamique de l'avion, et la matrice B la contribution des gouvernes. Elles sont toutes deux fonctions de l'état complet de l'avion (vitesse, configuration des becs et volets, altitude, masse, inerties, souplesse, etc...). Si l'on en connait (par des capteurs, calculateurs, ...) les paramètres principaux, on peut réaliser une loi évoluée $u=f(x^*, A^*, B^*)$, où x^* représente les mesures ou estimations de l'état x, A^* et B^* les estimations de A et B en fonction des paramètres disponibles et de la puissance de calcul utilisable dans les CDVE.

Mais il faut toujours considérer les pannes de capteurs, et assurer la sécurité et la pilotabilité de l'avion dans ces cas, en fonction de la probabilité d'occurrence, toute panne non extrêmement improbable ne devant pas avoir de conséquence catastrophique. Cela limite donc la différence entre l'avion muni de CDVE en fonctionnement normal, et l'avion dans sa configuration la plus dégradée, plus proche de l'avion "naturel": il faut en particulier que le pilote puisse s'adapter, c'est-à-dire que les qualités de vol ne soient pas trop mauvaises. C'est dire qu'il y a une corrélation importante ent e l'architecture des systèmes (redondances, fiabilité,...) et la possibilité de réaliser des lois évoluées.

Le même type de contrainte tient aux possibilités physiques des gouvernes. Il est en effet intuitif que plus on s'éloigne des caractéristiques de l'avion naturel, en particulier dans le sens d'améliorer les amortissements et temps de réponse, plus les gouvernes sont sollicitées. Par exemple , si l'on veut accélérer le mode de roulis pur, il faudra un débattement et une vitesse de débattement plus importants, et l'on amplifiera les hautes fréquences. Fig 1 et 2 montrent l'effet d'un mode de roulis pur accéléré de 50% (les autres modes restant inchangé) sur un avion de type A320: l'entrée est un créneau de manche de 5°/s de vitesse de roulis commandée. On voit sur la figure 1 un dépassement d'environ 30% de la commande en roulis "accélérée" par rapport à la "nominale", tandis que la figure 2 montre l'effet sur la vitesse de roulis (p), le dérapage restant nul dans les deux cas.

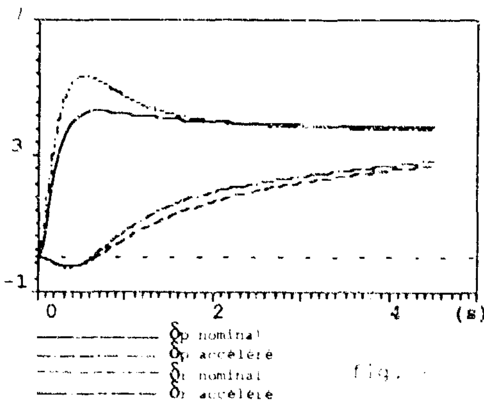


fig.

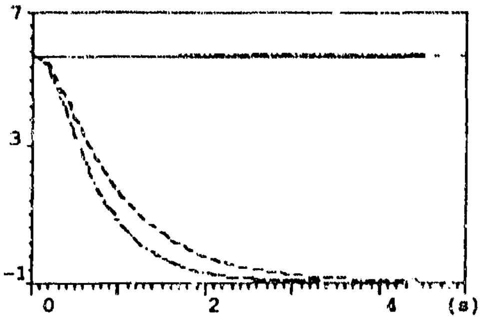


fig. 2

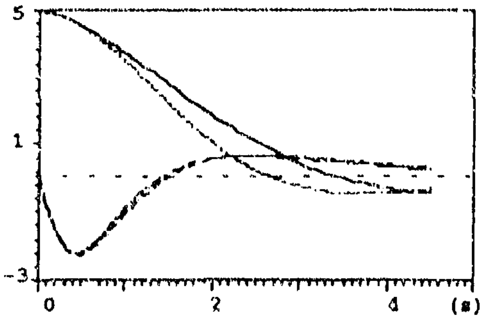


fig. 4

On est donc limité dans les modifications des qualités de vol par rapport à l'avion naturel par les caractéristiques des gouvernes et servocommandes (braquages maximaux, vitesses maximales de braquages, fatigue, etc...).

En plus des limites dûes aux gouvernes proprement dites, il existe des contraintes résultant de leur interaction avec la cellule de l'avion, c'est-à-dire le confort des passagers, le flottement, ou les charges: contrairement à un avion conventionnel, les lois de pilotage d'un avion muni de CDVE peuvent modifier ces caracteristiques.

Par exemple, une pulsation du roulis hollandais trop augmentée par rapport à l'avion naturel peut avoir tendance à augmenter l'effet en lacet d'une rafale de vent latéral, donc nuire au confort (ou l'améliorer), en particulier au niveau des passagers situés à l'arrière de l'appareil. Cette modification risque aussi de modifier les calculs de charge sur la dérive. Les figures 3 et 4 montrent l'effet d'une telle modification du roulis hollandais sur la réponse à une condition initiale en dérapage (5°), représentant un créneau de vent latéral. On voit sur la figure 3 que la commande "accélérée" de la gouverne de direction part dans le sens opposé à la commande nominale, c'est-à-dire qu'elle a tendance à amplifier le mouvement de lacet en mettant de la gouverne vers la droite pour un dérapage qui vient de la droite; par contre le dérapage (figure 4) revient alors plus vite à 0.

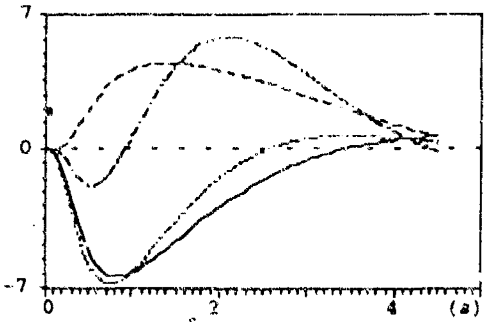


fig. 3

Nous avons aussi mentionné des contraintes humaines dans la conception des lois de pilotage: il s'agit essentiellement des besoins des utilisateurs, c'est-à-dire des pilotes, qui ont certaines habitudes de pilotages bien ancrées, et adaptées au pilotage de la plupart des avions, car fondées sur des sensations ou des visualisations. Il faut d'autre part que les qualités de vol (temps de réponse,...) soient compatibles avec le comportement qu'attend le pilote, afin d'éviter des phénomènes de pompage piloté (PIO) qui peuvent être dangereux à l'approche du sol; enfin, il faut donner au pilote la possibilité de maîtriser les pannes de systèmes amenant une dégradation des qualités de vol.

Si de nombreuses études théoriques ont été menées sur les critères de qualités de vol, et sur la sensibilité d'un avion au PIO, il convient de mentionner ici le rôle essentiel joué par les essais au simulateur, puis en vol, et la nécessité de disposer de moyens pour adapter les lois de pilotage en fonction des appréciations des pilotes.

Enfin, les autorités de certification imposent certaines contraintes: leur exigences vont en général dans le sens de la sécurité et d'une bonne manœuvrabilité; ainsi en latéral, il faut essentiellement démontrer des taux de roulis minimaux, et une bonne résistance aux pannes moteur(s).

IV. Expérience de l'A320.

a) Architecture

L'architecture des commandes de vol électriques de l'A320, schématisée figure 5, est la suivante: sur l'axe de roulis, les ELAC (Elevators and Ailerons Computer) reçoivent les informations anémométriques, inertielles, etc, nécessaires à l'élaboration des lois de pilotage, qui sont donc calculées dans les ELAC (en fonctionnement normal). Ceux-ci élaborent ainsi les ordres pour toutes les gouvernes, et asservissent les gouvernes d'ailerons. Les SEC (Spoilers and Elevators Computer) réalisent l'asservissement des spoilers aux ordres issus des ELAC.

Sur l'axe de lacet, les FAC (Flight Augmentation Computer), reçoivent aussi l'ordre des ELAC pour commander la servocommande de "Yaw Damper Actuator", qui vient s'ajouter au braquage commandé mécaniquement par le palonnier, et électriquement par le trim. (L'ordre total est ensuite limité par une butée mécanique fonction de la vitesse de l'avion, afin de limiter les charges sur la dérive). Ainsi toutes les gouvernes sont activables électriquement par des calculateurs numériques (dont le rôle n'est pas limité aux fonctions présentées ici).

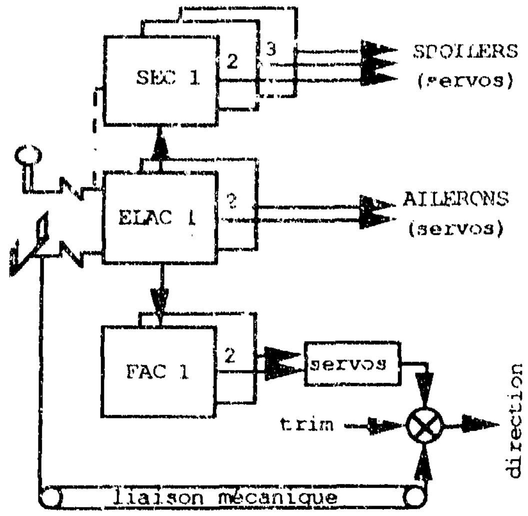


fig. 5: schéma simplifié des CDVE latérales de l'A320.

Les organes de pilotage sont donc le mini-manche et le palonnier, munis de capteurs électriques de position. Le mini-manche n'a aucune liaison mécanique avec les gouvernes, contrairement à la gouverne de direction qui reste disponible en dernier secours (perte totale de la génération électrique).

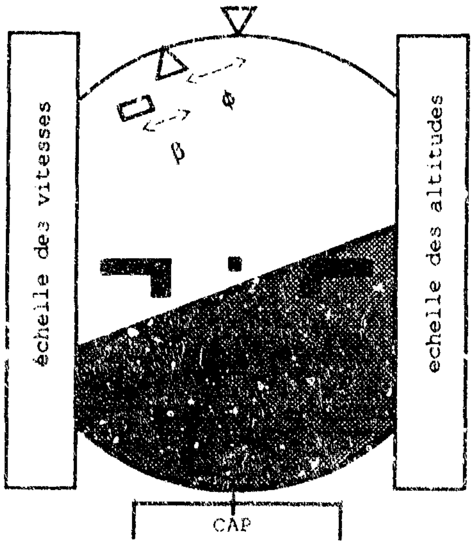


fig. 6: schéma simplifié du PFD de l'A320.

La visualisation associée au pilotage est essentiellement constituée du PFD (Primary Flight Display), qui rassemble les paramètres primaires du vol, dont ceux qui nous intéressent pour le pilotage latéral de l'A320: l'assiette latérale, le dérapage, et aussi le cap: fig. 6.

L'architecture des CDVE de l'A320 est donc prévue pour réaliser ces lois de pilotage latéral (et longitudinal) "évoluées", dont nous allons détailler les objectifs, et la manière dont ont été traitées certaines contraintes.

b) Objectifs et contraintes.

Au niveau stabilité, il s'agissait d'amortir le roulis hollandais (avec un facteur d'amortissement $\xi>0.6$ alors que celui de l'avion naturel est en général $\xi<0.2$) en conservant sensiblement la même pulsation, de garder le mode de roulis pur proche de celui de l'avion naturel, et d'augmenter nettement la stabilité spirale. On reste ainsi relativement proche des modes de l'avion naturel lorsqu'ils sont compatibles avec nos objectifs, afin d'éviter une activité des gouvernes trop importante, ou un changement de pilotage trop radical en cas de panne. Ce placement des modes de l'avion avec CDVE en assure la stabilité sur panne moteur ou autre dissymétrie raisonnable.

Les degrés de liberté restant ont été utilisés pour découpler l'assiette latérale ϕ et le dérapage β c'est-à-dire faire en sorte en particulier qu'un dérapage soudain, dû à une rafale par exemple, crée peu de roulis induit (pour améliorer le confort), et que le roulis hollandais ne soit pas perceptible sur l'assiette latérale (pour éviter dépassement et flottement dans la prise d'assiette).

Une fois la stabilité spécifiée, il reste à fixer les objectifs de pilotage: sur un avion conventionnel (à "câbles"), le manche commande une mise en virage, dont la coordination est assurée par le pied. Pour que l'A320 reste cohérent avec ce comportement, tout en réduisant la charge de travail du pilote, le braquage du manche est transformé en une commande de vitesse de roulis, mais à dérapage nul, c'est-à-dire une mise en virage coordonné. Quantitativement, la réponse (en p_1) de l'avion doit être semblable à un 1er ordre, de constante de temps correspondant au mode de roulis pur, pour un pilotage précis et disposant d'une bonne marge vis-à-vis du PIO. Ainsi on obtient en fait une stabilité spirale importante sur perturbation, et parfaitement nulle au regard du pilotage (assiette constante manche au neutre).

Quant au palonnier, il commande du dérapage, plus une légère assiette de roulis induit, qui a été demandée par les pilotes pour retrouver un comportement plus conventionnel qu'un dérapage obtenu les ailes parfaitement à plat. La principale contrainte dans la commande en lacet était de permettre le contre de la panne moteur, en particulier à basse vitesse, et le décrabe par fort vent de travers.

Ainsi en fonctionnement normal, le pilotage est effectué au manche seul, et par impulsions puisque l'avion reste equilibre manche au neutre.

Les protections du domaine de vol latéral sur l'A320 se limitent en fait à une protection en assiette latérale, qui est introduite de la manière suivante: pour $|\phi_1| \in [33°, 66°]$, une stabilité spirale est rétablie de manière à ce que l'avion revienne à 33° d'assiette manche au neutre, et que plein manche il atteigne la limite fixée à 66°; ceci correspond, en virage stabilisé, à un facteur de charge de 2.5 g cohérent avec les protections du domaine de vol longitudinal en facteur de charge, et permet de protéger l'avion contre la vrille.

Cependant, du fait que la position des gouvernes de roulis n'est pas proportionnelle à la position du manche, il a été nécessaire d'aider le pilote en cas de panne moteur au décollage, où l'objectif de performance se traduit par des gouvernes alaires non braquées (surtout les spoilers). Un dispositif automatique de remise à zéro de ces surfaces par le trim de direction à cap constant a été testé, mais peu apprécié des pilotes, car il interférait trop avec leur propre réaction. Un objectif de dérapage a donc été visualisé sur le PFD, le "β target", qui permet au pilote de rejoindre ces performances optimales, par une action au pied instinctive (le pied chasse la bille) annulant le derapage visualisé sur le PFD (dérapage décalé du β target).

c) Réalisation.

La realisation de ces objectifs , en utilisant les techniques maintenant classiques de l'automatique multivariable (techniques de placement des modes pour le contrôle de la stabilité, et des vecteurs propres pour le découplage ϕ/β), nous donne une loi avec un interaction complète entre les deux axes de roulis et de lacet (fig. 7).

Les objectifs de stabilité sont réalisés par les gains de retour Kret en β^*, p, r, et ϕ, où β^* est une estimation du dérapage fondée sur la mesure de l'accélération latérale ny, et la connaissance a priori de l'équation des forces latérales. La matrice Kret est de dimension 2x4 et se déduit entièrement et de manière unique des objectifs de placement des pôles et de découplage ϕ/β, ce qui a facilité les ajuste-ments rendus nécessaires après essais pilotés au simulateur, ou identification après essais en vol. Cette matrice dépend du cas de vol, estimé par la vitesse conventionnelle (Vc) et la configuration des becs et volets (conf).

Quant aux objectifs de pilotage, la vitesse de roulis commandée par le manche est transformée en consigne d'assiette après intégration ($\int$) et un gain direct d'avance de phase (K), dont le rôle est d'accélérer la réponse de l'avion sur un ordre pilote, en particulier en compensant le mode spiral. Le palonnier commande du dérapage et une assiette latérale (ϕ_c-β_c). La matrice de précommande Kp permet de relier ces consignes à l'avion stabilisé par les gains de retour: elle est calculée pour obtenir, en régime stabilisé et hors perturbations, β=β_c et ϕ=ϕ_c.

Hors protection, il n'y a pas d'intégrateur de précision, puisque les variables β_c et ϕ_c ne sont pas accessible au pilote. Par contre, en mode de protection, c'est-à-dire pour $|\phi_1| \in [33°, 66°]$, l'intégrateur se transforme en intégrateur de précision pour assurer ϕ_1=33° manche neutre et ϕ_1=66° plein manche, le terme de protection étant fonction du ϕ_1 mesuré.

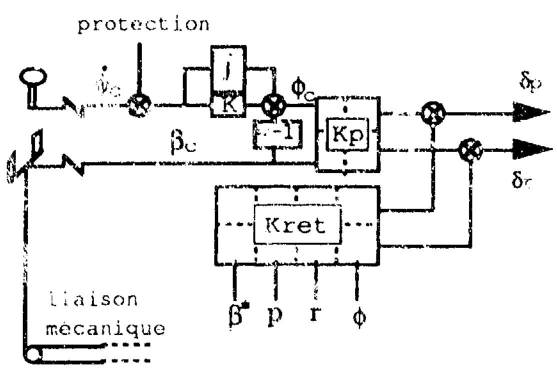


fig. 7: structure de la loi latérale

Cette structure a permis de réaliser les objectifs fixés, comme le montrent les résultats d'essais en vol retranscris sur les planches 1 à 4.

La première planche montre la réponse de l'avion à des créneaux de manche latéral. Le δp représente l'ordre global de roulis (avant distribution aux gouvernes alaires), qui suit à peu près l'ordre manche sans trop de dépassement. La direction assure automatiquement la coordination de virage et l'amortissement du roulis hollandais, alors que le palonnier n'est pas sollicité. La vitesse de roulis réagit bien comme un 1er ordre, ce qui se traduit par une assiette rejoignant la consigne sans dépassement (on remarque l'avance de phase présente sur la consigne ϕ_c, assurant un bon temps de réponse en roulis), tandis que le dérapage reste très faible.

La deuxième planche montre la réponse de l'avion à une sollicitation au palonnier: la gouverne de direction suit la commande avec toutefois un ordre de stabilisation superpose, tandis que la commande en roulis agit immédiatement pour contrer la majeure partie du roulis induit par la direction; l'avion se stabilise alors à un dérapage d'environ 6° avec une assiette latérale d'environ -3°. Sur un avion conventionnel, cette manœuvre amènerait vite l'assiette à diverger.

La troisième planche montre la limitation en roulis, active à partir de 33°: le manche est continûment braqué à droite (d'environ 12°) et l'assiette se stabilise à 50°, à dérapage toujours nul.

La dernière planche montre l'effet d'une panne moteur: les vitesses de lacet et de roulis commencent à dériver mais sont contrées par les braquages automatiques de la direction et des surfaces de roulis, qui aboutirait à un état stable si le pilote ne réagissait pas au manche pour replacer l'avion dans une trajectoire à cap constant, toujours sans toucher au palonnier. Le dispositif automatique (abandonné par la suite), ramène ensuite l'ordre de roulis à zéro en braquant progressivement la direction par le trim, tout en restant à cap constant.

Finalement, les objectifs fixés et leur réalisation ont été validés après de nombreux essais au simulateur et en vol, et seront reconduits sur les prochains Airbus (A330 et A340). En effet, la stabilité ainsi obtenue est bonne, le pilotage simple (pilotage principalement au manche par impulsions), et la protection en assiette latérale efficace.

III. Perspectives.

Sur l'A320, les lois de pilotage implantées dans les calculateurs de CDVE ont pleine autorité sur les gouvernes, et peuvent donc contrer, par leurs protections en particulier, un ordre du pilote qui amènerait l'avion dans une situation potentiellement dangereuse. Ceci implique donc, en particulier, d'élaborer ces lois à partir d'informations sures, c'est-à-dire provenant d'au moins deux sources concordantes. Avec une redondance suffisante des calculateurs de CDVE, la disponibilité des lois évoluées repose donc surtout sur celle des capteurs: les A320 disposent de trois centrales ADC/IRS, mais si une ADC tombe en panne, ou était en panne en cas de départ sous MMEL (Master Minimum Equipment List), ces lois sont à la merci d'un désaccord entre les deux ADC restantes, auquel cas les lois dites "normales" sont reconfigurées en lois simplifiées, c'est-à-dire loi directe en roulis (braquages des gouvernes alaires proportionnels au manche) et commande mécanique de la direction, avec toutefois un stabilisateur de lacet à autorité limitée si une IRS est disponible.

Une meilleure disponibilité des lois évoluées, nécessaire si l'on veut élargir leur possibilités d'action, suppose donc une meilleure disponibilité des systèmes embarqués et surtout sur une redondance élevée sur les mesures, redondance soit matérielle (et onéreuses, surtout proportionnellement sur les petits avions), soit analytique. Par redondance analytique, on entend en général reconstitution d'une variable par un estimateur, fondé sur des capteurs ne mesurant pas directement la variable, et sur la prédiction du comportement de l'avion. Ainsi un même capteur peut être artificiellement démultiplié pour fournir plusieurs informations, et, à nombre de capteurs équivalent, améliorer la disponibilité de lois évoluées.

Ceci est déjà utilisé sur l'A320 sur l'axe longitudinal: lorsqu'il ne reste plus qu'une IRS, la vitesse de tangage qui en est issue est utilisée pour estimer le facteur de charge normal à la trajectoire, estimation comparée aux mesures accélérométriques disponibles (capteurs spécifiques peu onéreux) afin de valider cette IRS.

De manière générale, on tend à rendre les lois de pilotage tolérantes aux pannes (voir les nombreux articles sur le sujet dans les publications), une methode étant de rendre les surveillances des capteurs plus intelligentes en tenant compte de la connaissance de la dynamique de l'avion, pour éliminer les sources incohérentes.

Ceci conduit donc à considérer les lois de pilotage des CDVE de plus en plus tôt dans le dimensionnement d'un avion. Il n'est qu'à prendre l'exemple des avions militaires qui sont naturellement instables (ce qui n'est pas le cas des avions de transport civil actuels!), et qui comptent sur les CDVE pour être pilotables. Plus raisonnablement, sur les avions de transport civil, il est possible de relâcher, par exemple, les exigences d'amortissement du roulis hollandais naturel, pour bénéficier de gains de masse sur la dérive au prix de quelque(s) gyromètre(s) de lacet supplémentaires pour assurer la fonction de stabilisation de ce mode.

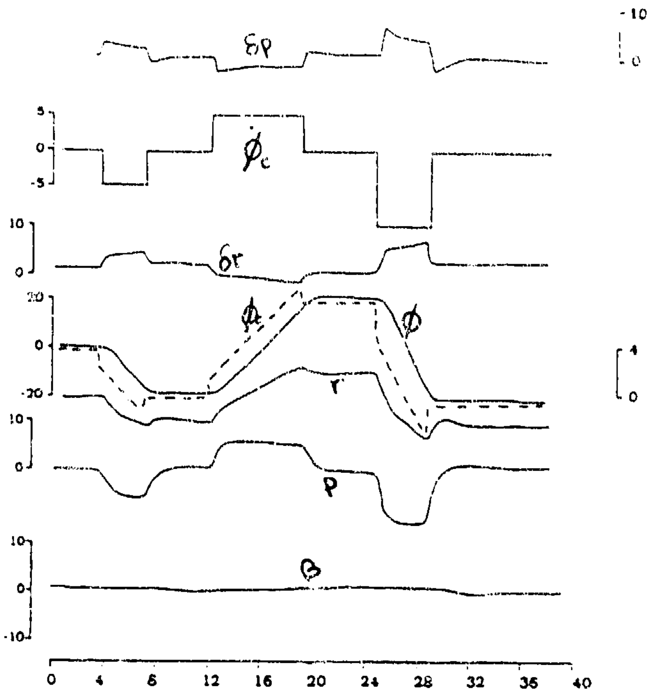
De même, les charges induites sur la structure par les gouvernes sont dépendantes des lois de pilotage, qui peuvent les augmenter, mais aussi les diminuer: il existe sur A320 une fonction LAF (Load Alleviation Function) pour réduire les charges sur la voilure en rafale verticale par braquage automatique des spoilers; sur A340, un dispositif MLA (Manœuvre Load Alleviation) permettra de réduire le moment de flexion à l'emplanture des ailes par braquage des ailerons et spoilers externes lorsque le facteur de charge mesuré dépasse 2 g.

Des études sont aussi en cours (sur les A340/A330) pour évaluer un amortissement de certains modes structuraux basse fréquence en utilisant la gouverne de direction par l'intermédiaire des CDVE, couplées à un capteur accélérométrique situé à l'arrière de l'appareil.

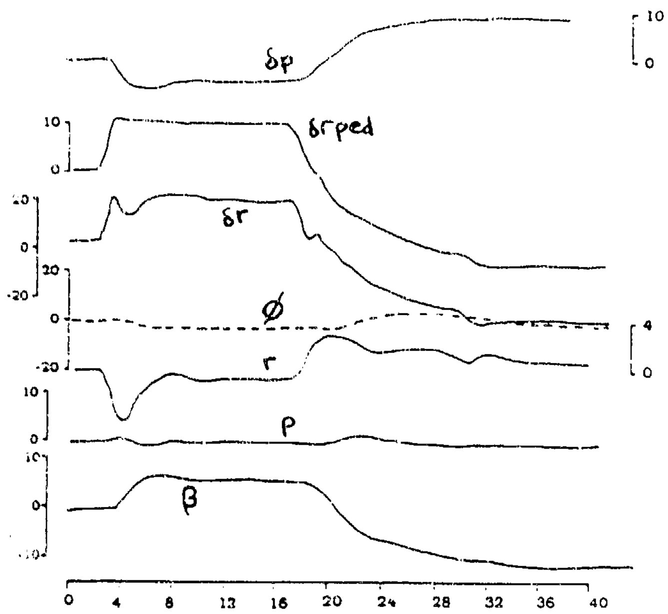
En outre, les protections du domaine de vol devraient permettre de justifier une réduction le domaine de dimensionnement de la structure, ce qui équivaut généralement à un gain de masse... Pour le mouvement latéral par exemple, on peut imaginer une protection en dérapage afin de réduire les charges maximales sur la dérive.

Inversement, si les objectifs de qualités de vol améliorées sont prioritaires, le dimensionnement de la cellule, des servocommandes, etc... devra tenir compte des lois de pilotage.

La difficulté essentielle devient donc la coordination, c'est-à-dire la détermination de priorités entre tous les objectifs et contraintes de domaines traditionnellement relativement indépendants, au moins au stade de la conception: les qualités de vol, les performances, les systèmes, et les structures (sans oublier les contraintes de coût et de maintenance).



REPONSE A DES CRENEAUX DE MANCHE



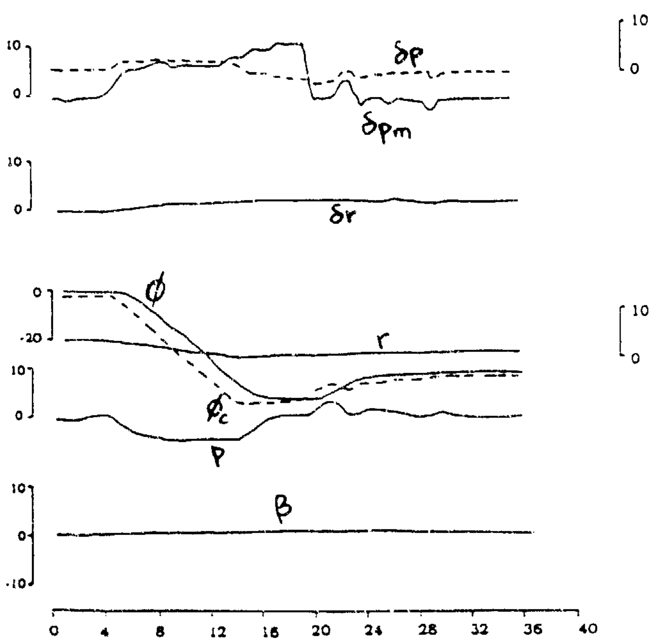
REPONSE A UN CRENEAU DE PALONNIER

PLANCHE 1

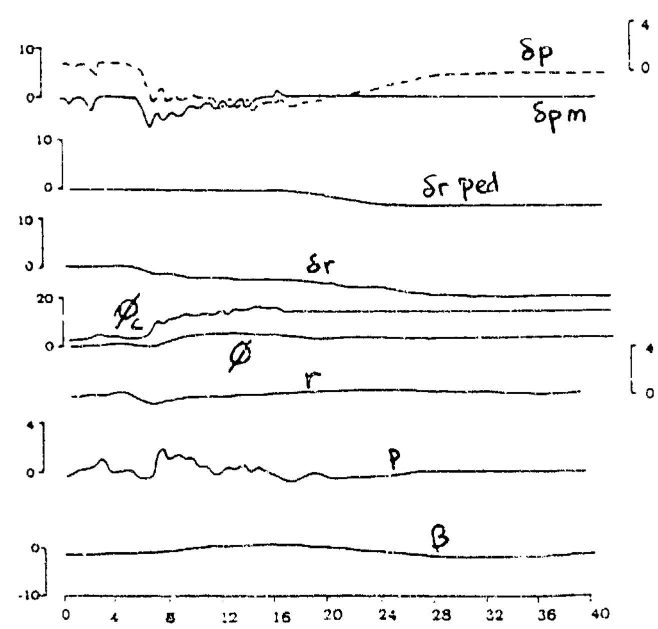
PLANCHE 2

PLANCHE 3

PLANCHE 4



LIMITATION D'ASSIETTE LATERALE



PANNE MOTEUR

MIL-STD-1797 IS NOT A COOKBOOK

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ABSTRACT

At the 1989 AIAA Atmospheric Flight Mechanics Conference, participants in the Flying Qualities Workshop engaged in a lively discussion regarding the content and application of the military flying qualities specification. As a result of this and other discussions it has become apparent that, despite many years of experience, some confusion still exists concerning the nature, purpose, and application of the flying qualities specification.

Much of this confusion stems from the form of the requirements themselves. A question frequently raised is whether flying qualities are pilot-oriented properties or whether they are the parameters defined in the requirements of the flying qualities specification. This question arises from the fact that most of the objective criteria in the specification are not closed-loop (pilot-in-the-loop) performance criteria or pilot acceptance criteria, but rather are criteria on open-loop (pilot-out-of-the-loop) characteristics of the augmented aircraft. Another source of confusion concerns the role of the specification itself: is it only a contractual document, or is it also a design guide? If the latter, is it equally effective in both roles? Consideration of the above questions leads to yet another. If the specification is intended as a design guide and the criteria are open-loop properties instead of closed-loop properties, which is more important: pilot satisfaction with closed-loop performance or compliance with the open-loop requirements?

In this paper the authors will address these questions by reviewing the background of the United States military flying qualities specifications. They will discuss the advantages and disadvantages of different types of requirements. Finally they will describe the way the specification is used by the USAF Aeronautical Systems Division program offices, for whom, among others, the flying qualities specification is intended.

BACKGROUND

The current version of the flying qualities specification is MIL-STD-1797A, "Flying Qualities of Piloted Aircraft", published in January 1990 (Reference 1). This is the tri-service version of MIL-STD-1797, which was first published in March 1987 as an Air Force specification (Reference 2). The MIL-STD-1797 series is the successor to the MIL-F-8785 series, the last revision of which was MIL-F-8785C (Reference 3). Though there are a few new requirements and some modifications to old ones, MIL-STD-1797A is primarily a remodelling of MIL-F-8785C into a Mil-Prime standard and handbook format. The standard is meant to be a framework for a specification that a procuring agency can tailor to each individual procurement. The quantitative and qualitative values of most of the requirements contained in the standard have been left blank. These blanks are to be filled in by the procuring agency when writing a specification for a particular program. The handbook is Appendix A of MIL-STD-1797 and actually comprises the greater part of the document. The handbook provides the procuring agency with guidance to fill in the blanks in the standard with appropriate criteria, and lessons learned from previous experience.

TYPES OF REQUIREMENTS

A survey of the system requirements of MIL-STD-1797 suggests that they may be divided into four different types. The first type is a Descriptive requirement. This type does not place any requirements on the aircraft. It requires the contractor (or sometimes the procurement agency) to define or describe certain aspects of the aircraft. Examples of this type of requirement are 4.1.1 Loadings, which requires the contractor to define the c.g. envelopes and corresponding weights, and 4.1.4.2 Service Flight Envelopes, which requires the contractor to define the Service Flight Envelopes for each Aircraft Normal State. Table I shows those paragraphs of MIL-STD-1797 which consist predominately of Descriptive requirements.

The second type of requirement are upper-level requirements. We will call these Primary requirements. This type places requirements on the aircraft but only through lower-level requirements. For example, 4.1.6.1 Allowable Levels for Aircraft Normal States requires that flying qualities for Aircraft Normal States within the Operational Flight Envelope be Level 1. Obviously this is a requirement on the aircraft, but it does not define the constraints of Level 1. That is left to lower-level requirements. Table II lists those paragraphs of MIL-STD-1797 which consist predominately of Primary requirements.

The lower-level requirements can be divided into two more types. The first type is a Subjective requirement. This type of requirement is qualitative in nature and thus open to different semantic interpretations. A good example of this is 4.1.11.2 Release of stores which requires that the "intentional release or ejection of any stores shall not result in objectionable flight characteristics or impair tactical effectiveness of Levels 1 and 2". Obviously the question of whether a particular characteristic is "objectionable" or not would be open to interpretation.

The other type of lower-level requirement is an Objective requirement. This is a quantitative requirement and thus less subject to different interpretations. One example of this is the first part of 4.2.1.1 Long-term pitch response which requires that any oscillation (in the pitch response to a step input) with a period of 15 seconds or longer shall have an equivalent damping ratio greater than 0.04 for Level 1, a damping ratio greater than 0.0 for Level 2, and a time to double amplitude greater than or equal to 55 seconds for Level 3.

Figure 1 shows the proportion of each of these types of requirements in MIL-STD-1797. The Objective requirements constitute about 85% of the requirements in the standard. The Subjective requirements constitute less than 10% of the total number of requirements. The Descriptive and Primary requirements constitute somewhere between 3% and 4% each.

Examination of the lower level requirements reveals that they may be divided in another way. This second approach depends on whether the requirement applies with the pilot in or out of the control loop. The first type places requirements on the characteristics of the aircraft without the pilot. We will call these Open-Loop requirements because the outer control loop, the one with the pilot acting to control the aircraft, is open. An example of this is 4.2.1.1 cited above.

Another type of requirement under this approach is one that applies to the behavior of the pilot-aircraft combination. We will call this a Closed-Loop requirement. This type requires the pilot to perform some task or maneuver in order to determine compliance. However, for some requirements, 4.1.8 Dangerous flight conditions, for example, the pilot does not necessarily have to evaluate the actual aircraft. Sometimes a piloted evaluation of an accurate simulation is sufficient to show compliance.

A third type of requirement under this system of classification is one that applies to the closed-loop response with a pilot model closing the loop. This type places requirements on either the performance of the closed-loop system with a given pilot model, or on the characteristics of the pilot model in order for the closed-loop system to achieve a given level of performance. Only one paragraph in MIL-STD-1797 contains requirements of this type: Alternative E. of 4.2.1.2 Short-term pitch response. This is a modified Neal-Smith criteria which places requirements on the pitch tracking performance of the closed-loop system with a given form of pilot model.

Figure 2 shows the proportion of lower level requirements that fall into each category of requirement type. Almost 79% of the lower-level requirements are of the Objective, Open-Loop type. About 12% are the Objective, Closed-Loop type. The Subjective, Closed-Loop and Subjective, Open-Loop types constitute about 8% and 1%, respectively, of the lower-level requirements. Less than half a percent are of the Objective, Pilot Model type. Since a Primary requirement will apply to several lower-level requirements, a given Primary requirement may simultaneously be Open-Loop, Closed-Loop, and a Pilot Model type. The Descriptive requirements are not requirements on the aircraft, so the control loop approach to grouping these requirements does not apply.

SUBJECTIVE REQUIREMENTS

The Subjective Closed-Loop requirements are the oldest form of flying qualities requirements. The very first flying qualities specification in the US was of this form. The specification for the US Army's first heavier-than-air aircraft called for it to "be steered in all directions without difficulty and at all times under perfect control and equilibrium" during the course of a one hour trial flight (Reference 4). This requirement serves as an excellent example of the general advantages and disadvantages of this form of requirement. Table III lists those paragraphs of MIL-STD-1797 which are predominately Subjective, Closed-Loop requirements. There are no paragraphs in which Subjective, Open-Loop requirements predominate.

The most significant advantage of subjective requirements is that they tend to describe the behavior we want (or do not want) from the aircraft in the terms the pilots would describe it. These requirements tend to be very pilot-oriented. Usually the purpose and value of these requirements are self-evident by reading them. They do not require a complex analytical derivation to understand.

The Subjective requirements also tend to be very general. Unlike Objective criteria, they do not need to be conditioned by Aircraft Class, or speed, or other flight parameters. As an example consider 4.1.12.1 Control centering and breakout forces, part of which requires that "the combined effects of centering, breakout force, stability and force gradient" of cockpit controls "shall not produce objectionable flight characteristics". This one statement suffices for all classes of aircraft, all Aircraft States, and all Flight Envelopes. The reason this type of requirement is so general is that the meaning of the qualitative terms is interpreted by the pilot in light of these other factors. What a pilot would call "objectionable" under one set of circumstances would change when given another set. But the qualitative term "objectionable" is a valid descriptor in both sets of circumstances.

The real attraction of this type of requirement for the people who write the flying qualities standard is that it can be used in situations where we do not know how to quantify what we want or perhaps even what we want to quantify. But we can usually describe what we want in qualitative terms. The best example of this is that first US Signal Corps specification cited above. If there was ever a time when we did not know how to quantify what we wanted in aircraft characteristics that was it. But it was possible to describe what was desired qualitatively. It is still a valid qualitative description of what we want in an aircraft today.

Despite the advantages we have listed for this type of requirement, the authors of this paper subscribe to the guidance given in MIL-STD-1797A under 4.1.9 Interpretation of subjective requirements. In general, "the focus in the flying qualities specifications has been, and will continue to be, on quantifying all requirements for which sufficient data exists." The desire for Objective specifications stems from the inherent disadvantages of Subjective requirements.

From a legal or contractual standpoint, the biggest problem with Subjective specifications is disparate interpretation of the qualitative terms. Obviously the test of compliance with this type of specification is a piloted evaluation of the aircraft or of a simulation of the aircraft. Different pilots will have different interpretations of the meaning of the qualitative terms, and if the differences between two pilots is big enough, the two pilots will come to different conclusions about compliance with a given requirement. In that case, whose judgement do you use? The contractor and the procuring agency will probably disagree. The first recourse is to get some more pilots. Hopefully enough of them will agree that a consensus can be reached one way or the other - but what if opinion remains evenly divided? This is not an ideal way to run a specification.

There are a couple of other problems from the design engineer's standpoint. First of all, Subjective requirements give the designer absolutely no guidance on how to design an aircraft to comply. (This is called "design freedom"). The designer must rely on his experience, knowledge, and judgement to determine what design parameters to play with to achieve compliance. Even after he decides what parameters will affect the behavior of interest, such quantities as "not objectionable", "realistic", "normal", or "not excessive" are extremely nebulous objectives to try to achieve.

The second problem for the design engineer is that he does not really know if his design has complied with a Subjective requirement until the development has reached a stage where a pilot can fly a simulation of it. At this point in the development it may be too late to make changes to key aspects of the design which affect the behavior in question.

An analogy to Subjective requirements in a cookbook (an idea suggested by the title of this paper) would be an instruction in a soup recipe to "make the soup taste good". The requirement is obviously desireable and easily stated on paper. But there is no guidance on what to do or how much of what to add in order to comply. Compliance can only be tested by tasting and then is subject to the whims of personal preference. Such a statement in a recipe serves no purpose. However, an airplane is not a soup, and a flying qualities requirement is not a recipe. The purpose of Subjective requirements in the flying qualities standard is to force the contractor to at least satisfy the pilots for those aspects of aircraft behavior that the engineers do not know how to quantify.

OBJECTIVE OPEN-LOOP REQUIREMENTS

The Objective Open-Loop requirements are the ones we most often think of when we talk about flying qualities specifications. These are the requirements on the open-loop transfer function parameters or on the characteristics of the time response to a step input. The theory behind these requirements is that some of these open-loop parameters or characteristics correlate with pilot opinion of aircraft behavior during closed-loop tasks, a theory which has been confirmed by experience and research. These open-loop parameters or characteristics can then be used to quantify aircraft flying qualities. Table IV shows those paragraphs of MIL-STD-1797 in which Objective, Open-Loop requirements predominate.

The primary advantage of the Objective Open-Loop requirements is that determination of compliance is not subject to interpretation or pilot variability. Compliance is not subject to interpretation because the criteria have quantitative values: you either meet those values or you don't. Compliance is not subject to pilot variability because the criteria are open-loop characteristics: no pilot is required in order to evaluate them. The requirements apply solely to the aircraft. Engineers like these qualities in a specification, as do contracting and legal departments.

Another big advantage of this type of requirement is that these parameters or characteristics can usually be related to aircraft design parameters. Thus the design engineer gets some guidance on what needs to be done to achieve compliance. Furthermore, in theory, he does not have to wait until the design is developed enough for piloted evaluation to decide whether he is on the right track or not. The engineer can evaluate his design analytically.

The big disadvantage of this type of requirement lies in the inconstant success of open-loop characteristics as measures of flying qualities. Though this approach has worked fairly well, history is replete with examples of aircraft which did not meet particular open-loop requirements, but still had satisfactory flying qualities in the behavior those requirements were supposed to address. There are also cases where aircraft met existing open-loop requirements but still had handling problems that those requirements were supposed to preclude. References 1, 2, and 5 through 14 all recount examples of both cases.

The reasons that Objective Open-Loop requirements have been only partially successful are numerous. The heart of the problem lies in the way the open-loop requirements are derived. The typical approach is to first define a closed-loop task to investigate some aspect of flying qualities. Pilots fly the closed-loop task in in-flight or ground-based simulators in which the open-loop dynamics can be varied. The pilots evaluate the flying qualities of various combinations of open-loop dynamics in the performance of the closed-loop task. Typically, one or more parameters are varied over a range of values while other parameters are set at values known to correlate with good flying qualities. The pilots evaluate the open-loop combinations qualitatively by their comments and quantitatively through use of a pilot rating scale. The most common scale in use today is, of course, the Cooper-Harper scale, which uses task performance, pilot workload, and controllability as the bases for the ratings. The engineers analyze the pilots' evaluations and try to determine which open-loop parameters correlate with the pilots' opinions and over what range of values the pilots found these parameters satisfactory, acceptable, controllable, or uncontrollable. The results form the ranges or boundaries of the Objective Open-Loop requirements.

The first problem with this approach is that of pilot variability. As we mentioned in the section on Subjective requirements, the meaning of qualitative comments will vary from pilot to pilot or, for that matter, from pilot to engineer. For Objective Open-Loop requirements this variability affects, not the issue of compliance with the requirement, but, rather, the issue of the validity of the requirement in the first place. Pilot rating scales do not completely avoid this variability because the decision process used to arrive at the ratings depends on how the pilot interprets the qualitative terms in the "decision tree". To minimize this, at least one basis of the Cooper-Harper rating scale, the task performance, is usually discussed in advance, with the pilots and engineers agreeing to use specific values for desired and adequate performance. This still leaves the pilot workload and the degree of controllability as subjective evaluations by the pilots. The problems of piloted evaluation and Cooper-Harper rating variability have received considerable attention recently (References 15 through 18).

Explicitly defining the task performance actually aggravates another problem however. Flying qualities are known to be task dependent. Pilot ratings and comments for a particular set of open-loop dynamics could be changed simply by changing the definitions of desired and adequate performance for a given evaluation task. MIL-STD-1797 recognizes this task dependence by dividing flight tasks into Flight Phase Categories. Several Objective Open-Loop requirements in MIL-STD-1797 have different Level ranges for each Flight Phase Category. But the Flight Phase Categories in MIL-STD-1797 are very broad and nonspecific. Users of MIL-STD-1797 should bear in mind that the tasks or the level of task performance desired in a new aircraft may not correlate exactly with the tasks or level of performance used to derive the criteria for a given requirement.

Another problem with the usual approach to flying qualities research is the practice of varying one parameter and setting all of the others to values known to correspond to satisfactory flying qualities. The obvious weakness of this approach is that the resulting criteria will not account for the interaction of various parameters when they are less than optimum. This interaction can seriously degrade the overall flying qualities. Even combinations of parameters which individually would be considered borderline Level 1 can degrade overall flying qualities to Level 2. This interaction is demonstrated in Figure 3, taken from Reference 19. This shows the Cooper-Harper ratings for various configurations in a combined pitch and roll tracking task, plotted against ratings for the same configurations in separate pitch and roll tracking tasks. Note that, as a general rule, the ratings for the combined pitch and roll tracking tasks are always worse than the worst rating for either of the separate single-axis tasks. This factor is not accounted for in the requirements of MIL-STD-1797.

The bottom line of all of this is that the ranges and boundaries of the Objective Open-Loop requirements of MIL-STD-1797 should not be treated as precise boundaries, but as broad "gray areas"; regions of transition from one Level to the next. Figure 4, taken from Reference 1, is a typical illustration of this. Figure 4 shows the Level 1 and Level 2 boundaries for Category A Flight Phases on a plot of Cooper-Harper ratings from several flight investigations of short-period pitch response dynamics. Note that along the Level 1 boundary line there are several instances of Level 2 pilot ratings inside the Level 1 boundaries. There are also instances of Level 1 pilot ratings outside the Level 1 boundaries.

To again make a comparison with a cookbook: the Objective, Open-Loop requirements would, at a casual glance, seem to have the closest resemblance to cookbook instructions, and this is the way engineers would like to treat them. But cookbooks usually give their instructions in specific quantities: a specific temperature to cook at, a specific length of time to cook, a specific quantity of an ingredient to add, etc. In the flying qualities standard the Objective requirements usually specify ranges of values or limits. These ranges or limits often depend on the values of other parameters. Furthermore, these ranges or limits are not absolute.

OBJECTIVE CLOSED-LOOP REQUIREMENTS

The Objective, Closed-Loop requirements are the requirements on aircraft characteristics or performance of the pilot-vehicle system in some task. Since MIL-STD-1797 is meant to be a general standard the type of tasks associated with these requirements are those common to all manned aircraft: takeoffs, landings, crosswind takeoffs and landings, control of failure transients, etc. Frequently, specifications for individual programs have also added other Objective, Closed-Loop requirements associated with the specific tasks expected for their aircraft. For example, the STOL and Maneuver Technology Demonstrator was required to make a precision landing with very demanding performance criteria. Table V illustrates those requirements of MIL-STD-1797 which are predominately Objective, Closed-Loop.

There are several advantages of this type of requirement. First, if you know what kind of performance you want for the pilot-vehicle system, this is the direct way to require it. Because the requirements are closed-loop they are very pilot-oriented; like the Subjective requirements. Unlike the Subjective requirements, however, the quantitative nature of this type of requirement is not subject to questions of interpretation. Furthermore, since the quantitative criteria of this kind of requirement are task performance parameters, they are mutually understandable for both the pilots and engineers.

The big question in this type of requirement is what level of performance to require in a task. This is an extremely tricky problem because this type of requirement can easily overdrive a design in one direction. In order to achieve the required level of performance for this kind of requirement a designer may make compromises in other aspects of flying qualites covered by subjective or open-loop requirements. For example, the stick sensitivity needed to achieve an extremely tight tolerance in fine tracking performance might cause a designer to sacrifice some control authority for gross maneuvering. If the pilot never really needs this level of precision in operational use, the designer may have sacrificed some maneuverability that the pilot could have used. The specific criteria to be used in these kinds of requirements must be carefully tailored to the actual needs of the operational user and must be balanced with other requirements to insure that the design is not needlessly driven to do one task well at the expense of handling in other tasks.

Another problem with this type of requirement is pilot variability. Pilots differ in their levels of training, experience, and techniques. Since this type of requirement is on the pilot-vehicle combination, differences in pilots will result in differences in pilot-vehicle performance. If the pilot-vehicle combination fails to meet the performance requirement, is the failure the fault of the pilot or the aircraft? If no pilots can make the aircraft meet the requirement, then the aircraft is obviously at fault (provided the requirement is not outrageous). But what happens when some pilots can meet the requirement and some cannot. This is the same sort of situation we faced with interpretation of subjective requirements. In this case, because the criteria are quantitative, we can use statistical analyses of multiple evaluations to determine if the probability of achieving the desired level of performance is acceptable. Even so, determination of compliance is still not as straight forward as with the Objective, Open-Loop requirements.

Another problem that the Objective, Closed-Loop requirements share with the Subjective requirements is that they must be evaluated with the pilot in the loop to determine compliance. In performing a closed-loop task a pilot introduces additional dynamics to those of the vehicle. What the pilot introduces is not always well-described for the designer, particularly for complex tasks. Thus, this type of requirement frequently does not offer the designer much guidance on how to achieve compliance.

Because this type of requirement applies to the pilot-vehicle combination, not just the vehicle, and because the performance criteria tend to be so specific for each procurement, there have never been very many requirements of this type in the general flying qualities specifications. However, in the past, individual programs have added requirements of this type to their particular flying qualities specifications and this trend will undoubtedly continue in the future. The Army's new Aeronautical Design Standard for rotorcraft flying qualities, ADS-33C, has an entire section consisting of this type of requirement. This section, called Flight Test Maneuver explicitly describes several demonstration maneuvers that must be performed by the vehicle and defines the level of performance that must be achieved and still get Level 1 pilot ratings. Most, if not all, of the requirements of ADS-33C will probably be incorporated in the next revision of the military rotorcraft flying qualities specification, MIL-H-8501 (Reference 21). The Flight Dynamics Laboratory has begun to look at similar ideas for fixed-wing aircraft to be incorporated in MIL-STD-1797 sometime in the future.

USE OF MIL-STD-1797A

We will now consider how the flying qualities specification is used by the USAF's Aeronautical Systems Division (ASD). The System Program Offices (SPOs) of ASD are representative users of the document in the procurement of aircraft for the Air Force's using commands.

To a SPO, MIL-STD-1797A represents a contractual document. It is not, however, a stand-alone specification, but is rather part of a heirarchy of specifications which are levied against a weapons system being procured by the U. S. Air Force. This heirarchy, along with pertinent specifications is illustrated in part in Figure 5. Of note is that each level in the hierarchy takes precedence over lower levels.

The contract is the legal document which states exactly what the contractor is going to supply to the government, under what conditions, and for what price. The Statement of Work (SOW) indicates the tasks to be performed, while the Contract Data Requirements List (CDRL) defines what data will be supplied to the government to show compliance with the contract. Data Item Descriptions (DIDs) define exactly what form and format these data are to be in.

For an aircraft procurement, the System Specification (SS) defines exactly what the complete weapon system, including avionics, weapons, etc., must be able to do. Performance requirements are usually specified at this level, and are in the form of a tailored MIL-STD-1793 performance specification (Reference 22). Requirements unique to the air vehicle itself appear at the next level in the form of an Air Vehicle Specification (AVS); flight control system and flying qualities requirements are levied at this level.

When a contractor responds to a Request for Proposals (RFP) issued by the government, he will offer a proposed contract with supporting hierarchy. This will include a tailored flying qualities specification. This proposed tailored specification may be further negotiated during the source selection process. Upon the signing of the contract between the selected contractor and the government, the tailored flying qualities specification becomes a contractual document at the appropriate level in the hierarchy of specifications. (Of course, methods do exist to amend the specification later if necessary.)

The purpose of the above discussion is not to detail the legal processes or documents of any specific program, but rather to make the point that, as used in the procurement of weapons systems, MIL-STD-1797A is a contractual document and is part of a hierarchy of specifications. As such, it levies requirements on the design which are binding. In order to show compliance with these requirements, the design is evaluated via analyses, simulations (both piloted and non piloted), and, ultimately, flight test. Some of the flight tests may be oriented toward identification of the closed-loop characteristics of the vehicle or toward specific flying qualities requirements. However, in recent years, there has been a trend toward "operational" testing, in which flying qualities characteristics may be evaluated as part of a larger scenerio. In some cases, this operational testing is a carefully-controlled approximation to service use, such as Handling Qualities During Tracking (HQDT) testing (Reference 23). In other cases it may actually be a true operational task. Brandeau (Reference 24) provides an example of the latter in his discussion of the development of the directional Stability Augmentation System (SAS) for the A-10. With the advent of the Air Force Operational Test and Evaluation Center (AFOTEC) and the requirement for Operational Testing and Evaluation (OT&E) in the 1980s, it is expected that the evaluation of flying qualities via operational or operational-type testing will become even more the rule.

A significant portion of the flying qualities testing examines various failure cases or degraded modes. This is to insure that catastrophic degradations in flying qualities following component or subsystem failures or damage are minimized, allowing the pilot to discontinue combat operations and return to base when necessary. While objective requirements may be levied against degraded-mode flying qualities, often such requirements are subjective.

In evaluating these and other subjective requirements, as well as evaluating the suitability of the aircraft for the operational tasks, heavy dependence is made on pilot opinion. This opinion is in the form of pilot comments and, in some cases, pilot ratings (usually Cooper-Harper ratings, Reference 25). In determining compliance with the subjective requirements, several representative pilots should be used, and each should have adequate time to fully evaluate the characteristic(s) in question. Pilot ratings should be individually considered rather than averaged, as one pilot may through his technique find a questionable or objectionable characteristic not noted by the other evaluating pilots. (For a more detailed discussion of inter- and intra-pilot variability and its implications, the reader is referred to Riley, References 15 and 16).

This of course raises the issue of specification noncompliance. In the case of the performance specification, compliance or noncompliance is (usually) obvious, while with flying qualities the issue is not so clear cut. This is because a performance specification is an objective specification, while flying qualities are by nature subjective. Thus the flying qualities specification is an attempt to reach a subjective goal, i. e., pilot acceptance, by means of a specification mixing objective and subjective criteria. Even the specification itself recognizes this dichotomy, and allows for demonstration via compliance with subjective as well as objective requirements (Paragraph 4.1.9). Thus there are not two but four possible results when evaluating flying qualities, as illustrated by Figure 6.

We would argue that in the end the pilots must find the flying qualities of the aircraft acceptable. This raises the possibility of the "off-diagonal" cases illustrated in Figure 6. In the first case (shown in the upper right of the figure), the aircraft does not meet some of the objective requirements of the specification, yet the flying qualities are judged as acceptable by the evaluating pilots. At this point, we would ask the contractor to "prove it" to the satisfaction of the SPO. If a sufficient number of pilots of the same class of aircraft, who collectively encompass the range of experience and background of future operators of the aircraft rate the characteristic(s) in question as acceptable for operational use, we would advocate accepting the characteristic as is. If, on the other hand, the aircraft meets the objective requirements of the tailored specification yet is rated as unacceptable by evaluating pilots (the lower left case of Figure 6), the characteristic(s) in question must be corrected. These two cases together probably account for less than 10% of the results of flying qualities testing yet occupy over 90% of the time and effort of the flying qualities engineers involved in any program. These cases require the most judgement, experience, and commitment by all parties and organizations involved to find and fix the problem(s) and make the customer - the operational user - happy with the flying qualities of the final product. This requires a dedication to doing what is right, not what is expedient.

CONCLUSIONS

To conclude this paper we would like to revisit some of the questions we mentioned in the introduction. To begin with we contend that flying qualities are actually subjective and closed-loop in nature. Thus the Subjective, Closed-Loop requirements come closer to specifying flying qualities than do the Objective, Open-Loop requirements. However, as we have already discussed, the Subjective requirements are not very satisfactory as specification criteria because their qualitative nature leaves the question of compliance open to different interpretation of the qualitative terms. Also they do not provide any design guidance.

This brings us to the second question: Is MIL-STD-1797 intended solely as a specification, or is it also intended to provide design guidance? Though the first priority of MIL-STD-1797 is as a specification, it has always been the intent of the Flight Dynamics Laboratory and ASD that it also provide design guidance. This is the reason for the great predominance of Objective, Open-Loop requirements in MIL-STD-1797.

The answer to the last question is the most important point in this paper. Pilot satisfaction with the flying qualities of the actual aircraft is more important than compliance with the Objective, Open-Loop requirements. Users of MIL-STD-1797, both contractors and procurement agencies, should always bear in mind that compliance with the Objective, Open-Loop criteria (which constitute the majority of the requirements in MIL-STD-1797) does not necessarily guarantee good flying qualities. The ultimate objective is not to meet the Objective, Open-Loop requirements but to get flying qualities with which the pilot can satisfactorily accomplish the required missions. In other words, it is more important to meet the intent of the specification than to meet the specification criteria. However, that statement should not be interpreted as a recommendation to ignore the criteria in the specification. Most of the criteria in the specification are backed up by extensive flight testing and experience with past aircraft. To ignore this experience is a considerable risk. Our recommendation to the designer is to use the criteria, but bear in mind their limitations and do not try to use the specification as a cookbook.

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TABLE I

PARAGRAPHS WITH PREDOMINATELY DESCRIPTIVE REQUIREMENTS

4.1.1	Loadings
4.1.2	Moments and products of inertia
4.1.3	Internal and external stores
4.1.4.1	Operational Flight Envelopes
4.1.4.2	Service Flight Envelopes
4.1.4.3	Permissible Flight Envelopes
4.1.5	Configurations and States of the aircraft
4.1.6	Aircraft Normal States
4.1.7	Aircraft Failure States
4.1.7.1	Aircraft Special Failure States

TABLE II

PARAGRAPHS WITH PREDOMINATELY PRIMARY REQUIREMENTS

4.1.6.1	Allowable Levels for Aircraft Normal States
4.1.6.3	Ground operation
4.1.7.4	Generic failure analysis
4.1.7.5	When Levels are not specified
4.1.9	Interpretation of subjective requirements
4.8.4	Flight at high angle of attack
4.8.4.2	Stalls
4.8.4.3	Post-stall gyrations and spins

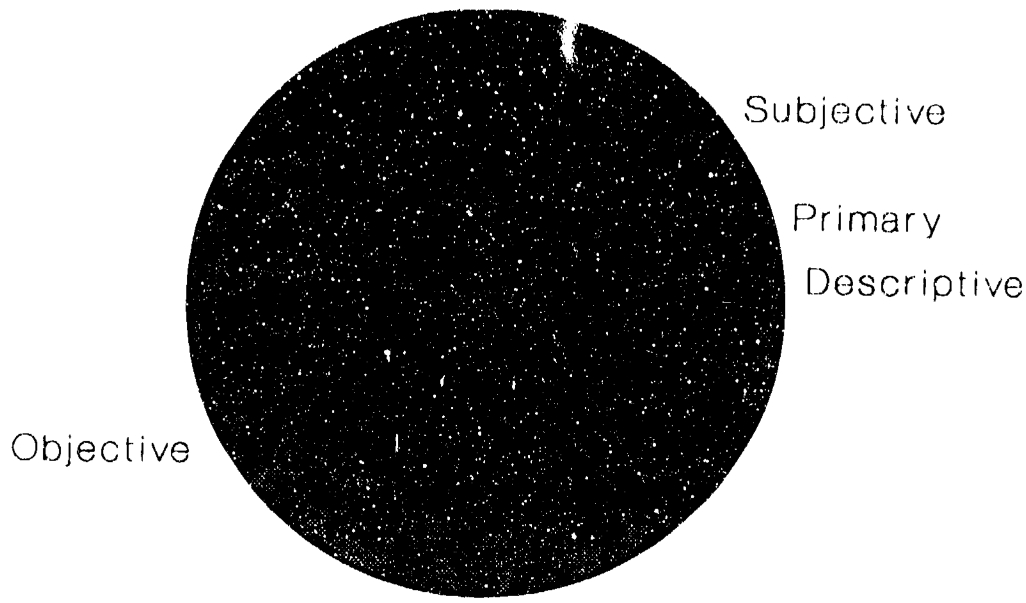

Figure 1. Proportion of each type of requirement in MIL-STD-1797

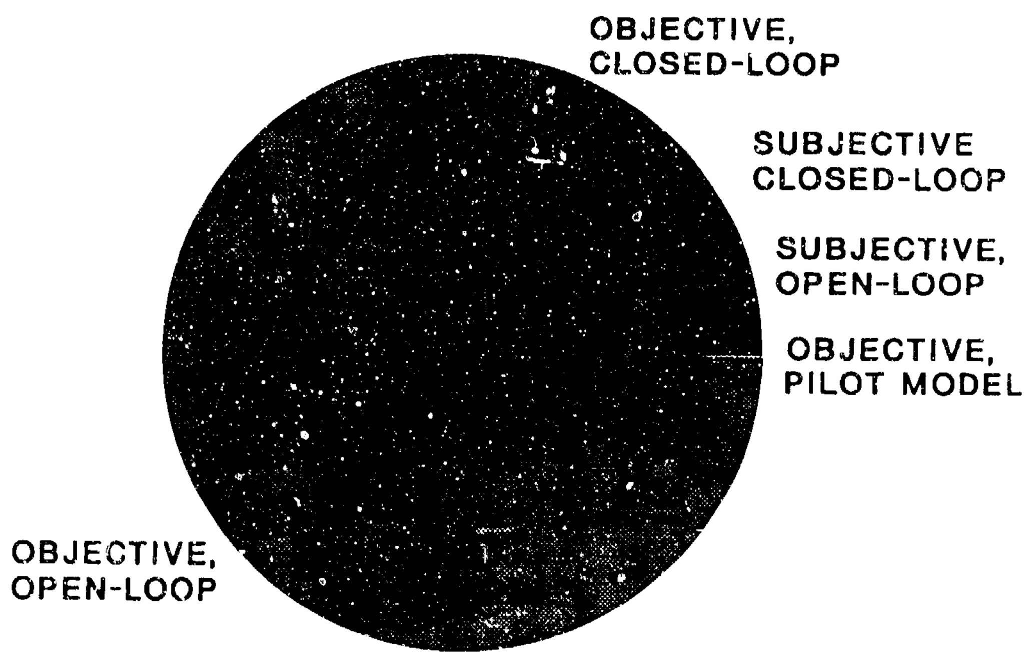

Figure 2. Proportion of each type of
lower-level requirement in MIL-STD-1797

TABLE III

PARAGRAPHS WITH PREDOMINATELY SUBJECTIVE, CLOSED-LOOP REQUIREMENTS

4.1.6.2	Flight outside the Service Flight Envelopes
4.1.8	Dangerous flight conditions
4.1.8.1	Warning and indication
4.1.11.1	Buffet
4.1.11.2	Release of stores
4.1.11.3	Effects of armament delivery and special equipment
4.1.11.4	Failures
4.1.11.5	Control margin
4.1.11.6	Pilot-induced oscillations (PIO)
4.1.11.7	Residual oscillations
4.1.11.8	Control cross-coupling
4.1.12	General flight control system characteristics
4.1.12.4	Rate of control displacement
4.1.12.6	Damping
4.1.12.7	Transfer to alternate control modes
4.1.12.8	Flight control system failures
4.1.12.9	Augmentation systems
4.1.13.2	Rate of trim operation
4.2.7.1	Pitch axis control power in unaccelerated flight
4.2.8.3	Pitch axis control forces - control force variations during rapid speed changes
4.5.2	Pilot-induced roll oscillations
4.5.3	Linearity of roll response to roll controller
4.5.6	Roll axis control for takeoff and landing in crosswinds
4.5.8.3	Roll axis control power in crosswinds
4.5.8.5	Roll axis control power in dives and pullouts
4.5.8.6	Roll axis control power for asymmetric loading
4.6.3	Pilot-induced oscillations
4.6.6	Yaw axis control power
4.6.6.1	Yaw axis control power for takeoff, landing and taxi
4.6.6.2	Yaw axis control power for asymmetric thrust
4.6.6.3	Yaw axis control power with asymmetric loading
4.6.7	Yaw axis control forces
4.8.2	Crosstalk between pitch and roll controllers
4.8.4.1	Warning cues
4.8.4.2.2	Stall characteristics
4.8.4.2.4	One-engine-out stalls
4.8.4.3.1	Departure from controlled flight

TABLE IV

PARAGRAPHS WITH PREDOMINATELY OBJECTIVE, OPEN-LOOP REQUIREMENTS

4.1.12.10	Auxiliary dive recovery devices
4.1.13.1	Trim system irreversibility
4.1.13.5	Trim for asymmetric thrust
4.1.13.6	Automatic trim system
4.2.1.1	Long-term pitch response
4.2.1.2	Short-term pitch response
	Alternative A - CAP or $\omega_{sp}^2/(n/\alpha)$, ζ_{sp}, τ_θ
	Alternative B - $\omega_{sp}T_{\theta_2}$, ζ_{sp}, τ_θ
	Alternative C - Transient peak ratio, rise time, effective time delay
	Alternative D - Bandwidth, time delay
	Alternative F - Time- and frequency-response criteria by Gibson
4.2.6.1	Pitch axis response to failures, controls-free
4.2.7.2	Pitch axis control power in maneuvering flight
4.2.8.1	Pitch axis control forces - steady-state control force per g
4.2.8.2	Pitch axis control forces - transient control force per g
4.2.8.4	Pitch axis control forces - control force vs. control deflection
4.2.8.5	Pitch axis control breakout forces
4.2.8.6.3	Pitch axis control force limits - dives
4.2.8.6.4	Pitch axis control force limits - sideslips
4.2.8.7	Pitch axis trim systems
4.3.1.2	Steady-state flight path response to attitude change
4.4.1	Speed response to attitude changes
4.4.1.1	Speed response to attitude changes - relaxation in transonic flight
4.5.1.1	Roll mode
4.5.1.2	Spiral stability
4.5.1.3	Coupled roll-spiral oscillation
4.5.1.4	Roll oscillations
4.5.1.5	Roll time delay
4.5.4	Lateral acceleration at pilot station
4.5.7.1	Roll axis response to augmentation failures
4.5.8.1	Roll axis response to roll control inputs
4.5.8.2	Roll axis control power in steady sideslips
4.5.9.1	Roll control displacements
4.5.9.2	Roll axis control forces to achieve required roll performance
4.5.9.3	Roll axis control sensitivity
4.5.9.4	Roll axis control centering and breakout forces
4.5.9.5.1	Roll axis control force limits in steady turns
4.5.9.5.4	Roll axis control force limits in steady sideslips
4.5.9.5.5	Roll axis control force limits for asymmetric thrust
4.6.1.1	Dynamic lateral-directional response
4.6.1.3	Wings-level turn
4.6.2	Yaw axis response to roll controller
4.6.4	Yaw axis control for takeoff and landing in crosswinds
4.6.5.2	Yaw axis response to failures
4.6.5.3	Yaw axis response to configuration or control mode change
4.6.7.2	Yaw axis control force limits in steady turns
4.6.7.3	Yaw axis control force limits during speed changes
4.6.7.5	Yaw axis control force limits with asymmetric loading
4.6.7.7	Yaw axis control force limits for waveoff (go-around)
4.6.7.11	Yaw axis breakout forces
4.7.1	Dynamic response for lateral translation

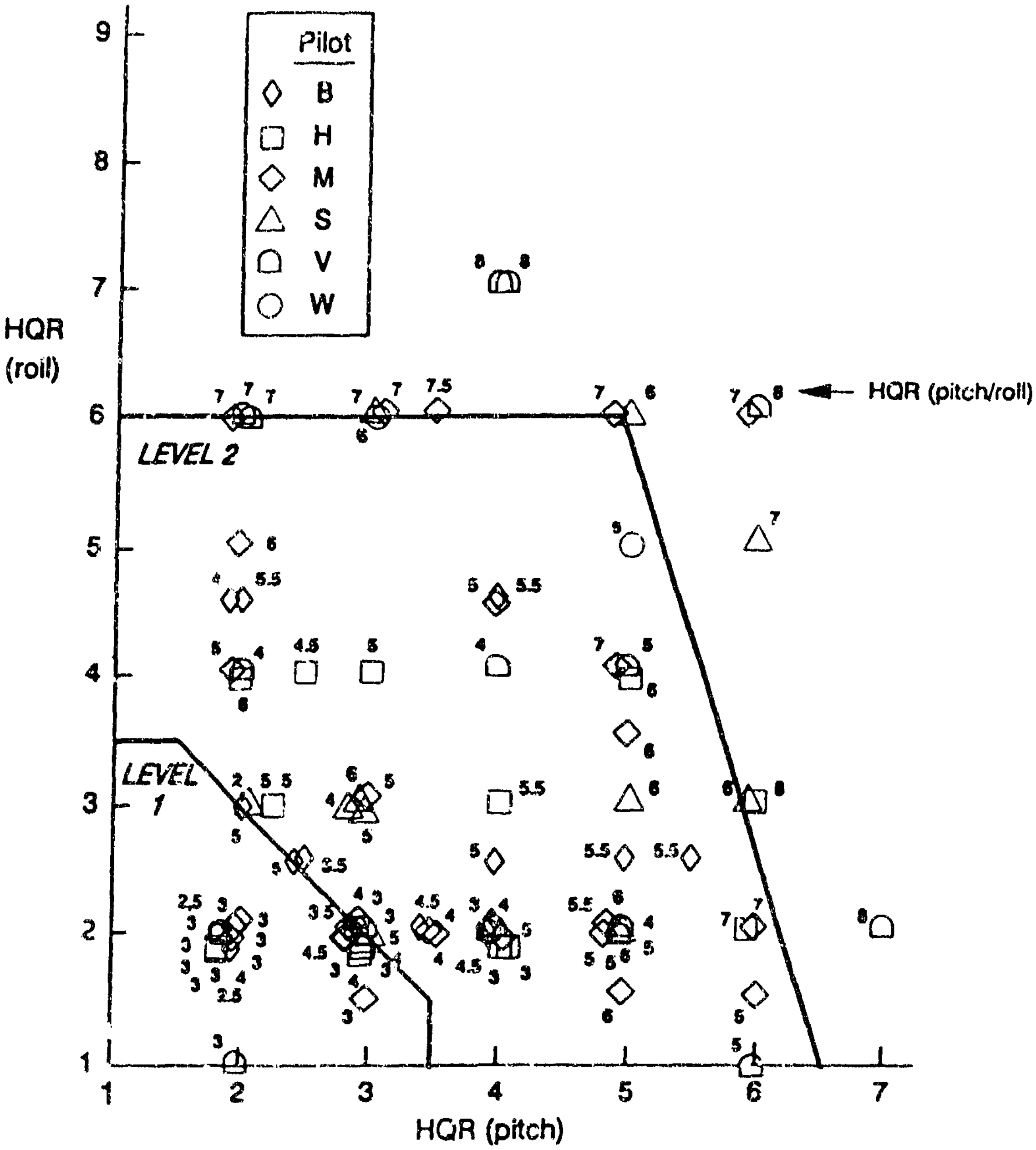

Figure 3. Multi-Axis Handling Qualities Ratings Plotted Against Single-Axis Ratings for a HUD Tracking Task (Taken from Ref. 19)

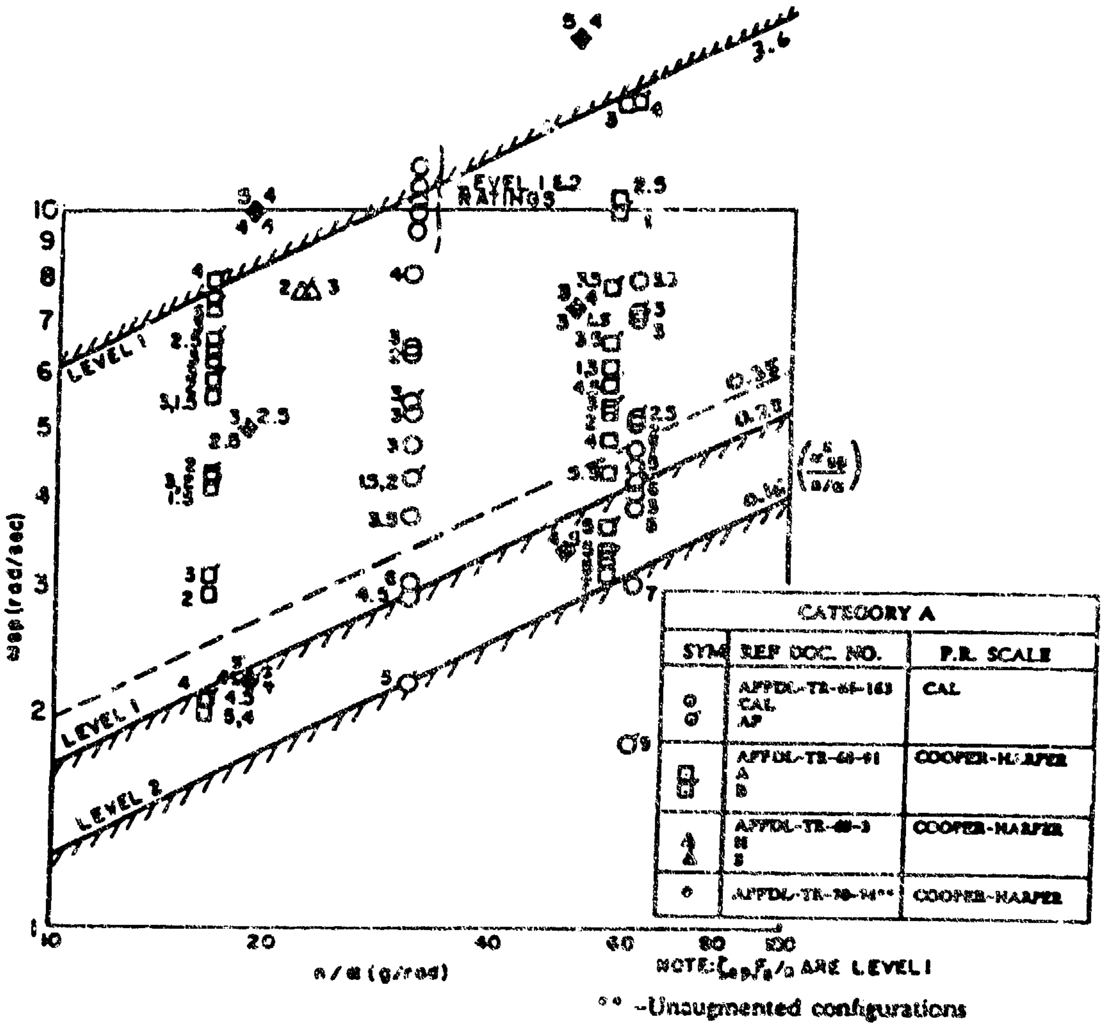

Figure 4. Comparison of pilot ratings with Category A short-period frequency requirements (Taken from Ref 1)

TABLE V

PARAGRAPHS WITH PREDOMINATELY OBJECTIVE, CLOSED-LOOP REQUIREMENTS

4.2.5	Pitch trim changes
4.2.8.6.1	Pitch axis control force limits - takeoff
4.2.8.6.2	Pitch axis control force limits - landing
4.2.8.6.5	Pitch axis control force limits - failures
4.2.8.6.6	Pitch axis control force limits - control mode change
4.5.8.4	Roll axis control power for asymmetric thrust
4.5.9.5.2	Roll axis control force limits in dives and pullouts
4.5.9.5.3	Roll axis control force limits in crosswinds
4.5.9.5.6	Roll axis control force limits for failures
4.5.9.5.7	Roll axis control force limits for configuration or control mode change
4.6.7.1	Yaw axis control force limits in rolling maneuvers
4.6.7.4	Yaw axis control force limits in crosswinds
4.6.7.6	Yaw axis control force limits in dives and pullouts
4.6.7.8	Yaw axis control force limits for asymmetric thrust during takeoff
4.6.7.9	Yaw axis control force limits with flight control failures
4.6.7.10	Yaw axis control force limits - control mode change
4.8.3	Control harmony
4.8.4.2.1	Stall approach
4.8.4.2.3	Stall prevention and recovery
4.8.4.3.2	Recovery from post-stall gyrations and spins

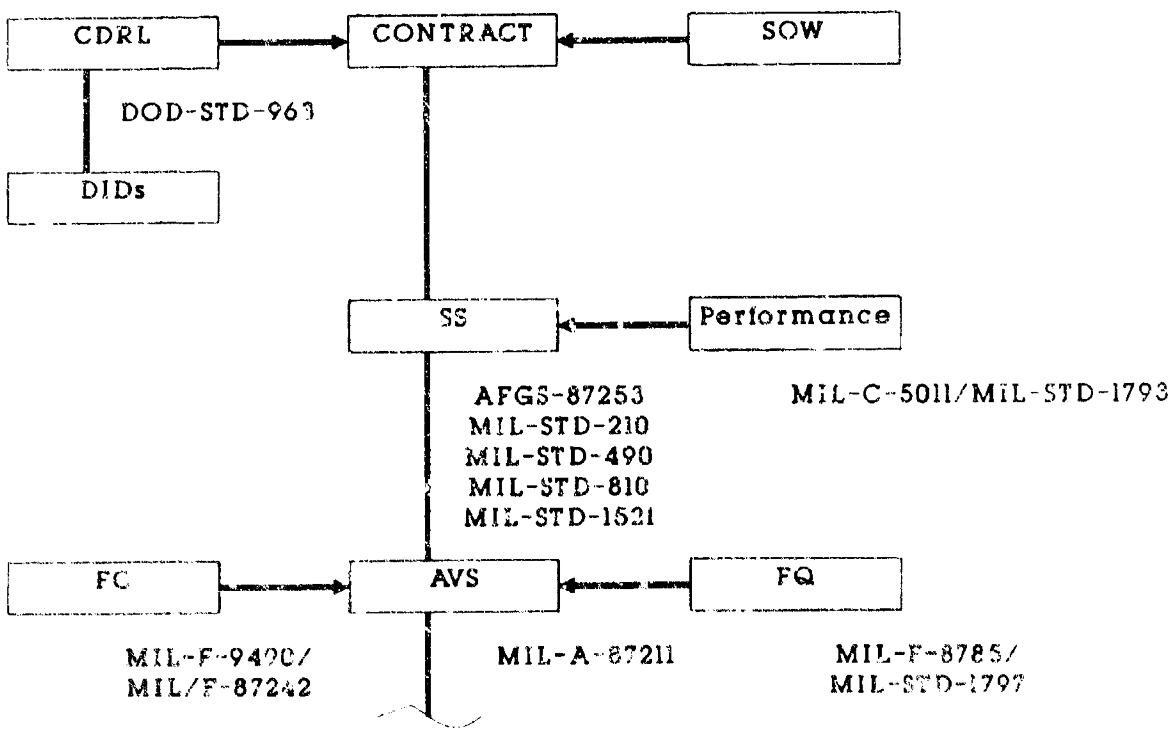

Figure 5. Sample Specification Tree

Specification Compliance

Pilot Opinion		Meets Specification	Doesn't Meet Specification
	Pilots Like It	OK	Prove It
	Pilots Don't Like It	Fix It	Fix It

Figure 6. Possible Outcomes of Flying Qualities Evaluations

FLYING QUALITIES EXPERIENCE ON THE AMX AIRCRAFT

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ABSTRACT

AMX is a subsonic ground attack aircraft with a fly-by-wire "Flight Control System" (F.C.S.) built into a digital flight control computer.
From the Flight Mechanics stand-point it has been designed against the MIL-F8785-C requirement. For some specific flight tasks the need of more demanding requirements has been envisaged.
Modern handling qualities criteria have been applied in the area of longitudinal and lateral-directional precision tracking task and P.I.O. tendencies to cope with operational problems.
High incidence criteria have been used in the design and evaluation of control modifications which improve the flying qualities in the stall and post stall regions.
Comparisons between analytical predictions, manned simulation and in-flight results have been made. Indications of agreement or disagreement with data and new criteria are presented.

SYMBOLS

δ_{st} Longitudinal stick deflection
δ_{stl} Lateral stick deflection
δ_{ped} Pedal deflection
δ_e Elevator deflection
δ_s Stabilizer deflection
δ_a Aileron deflection
δ_{sp} Spoiler deflection
δ_r Rudder deflection
α Angle of attack
γ Flight path angle
θ Pitch attitude
q Pitch rate
N_z Normal load factor
H Altitude
X Downrange distance
ϕ Roll attitude
p Roll rate
r Yaw rate
ω_d Dutch roll frequency
ζ_d Dutch roll damping
ω_ϕ Zero roll frequency
ζ_ϕ Zero roll damping
β Sideslip angle
ψ Nose heading angle
Az_{CCIP} Azimuth CCIP
τ_s Spiral time constant
τ_r Roll time constant
τ_{ep} Roll time delay

INTRODUCTION

Brief History

AMX aircraft (figure 1) is a subsonic, single seat, single engine, dedicated attack aircraft developed within a framework of a joint Italian and Brazilian programme. Both Air-Forces had requirements for an aircraft whose primary mission would be "ground support at and just beyond the forward edge of battle" supporting the land and naval forces and having capability to provide:
- close interdiction
- close air support
- reconnaissance

and possibility of:
- air defense and offense

Therefore the main design aims were about an aircraft capability to operate at low altitude subsonic speed with the following characteristics:
- large external store capability for ground attack
- air to air missile and gun for the air defense
- low sweep, medium thick wing to adopt a sophisticated "High Lift System" for adequate take-off and landing performances.
- single seat, high visibility cockpit
- advanced avionic systems for navigation and weapon aiming with head-up
- high survivability to continue the mission despite considerable battle damage.
- safe return-home capability following total electrical and hydraulic failure.

Flight control system description

The AMX flight control system, therefore, has been designed as a "fail operative" system with a mechanical "back-up".
A fly-by-wire F.C.S. provides three axes control, trim and compensation employing a dual-dual digital computer, in addition to analog motion dampers for all three axes. The hydraulic system is a dual-source, dual-redundant throughout. Normal functioning is still ensured after the first electrical and/or hydraulic failure whereas after the second electrical failure the controls are provided by hydraulically assisted back-up linkage on ailerons and elevator. These revert to manual following the second hydraulic failure, degrading the flying qualities to level 3 for the longitudinal and lateral stick forces in cruise as well as in landing flight phases.
Figure 2 shows the layout of the aerodynamic control surfaces assisted by mechanical (hydraulic or manual) and electrical control lines, while figure 3 shows a schematic of these connections.

Flight simulator

The AIT Flight simulator has played a very important role since the very beginning in the development of AMX aircraft. It continues to support flight testing on the instrumented prototypes relevant to handling and manoeuvrability of aircraft modifications. At the moment an activity is in progress to investigate on the twin-seat aircraft behaviour at high incidence angle, in the post-stall region.

Part of an expanding simulation complex (figure 4), the AMX simulator is a fixed base, fully instrumented cockpit with a real "head up display", in an inflatable dome with 10m diameter. A "control loading system" is available for a quick change of stick characteristics and a "G-suit, G-seat" system is available to improve the usable cue environments.

Handling Qualities Requirements

The aircraft is basically stable and not completely "fly-by-wire" dependant. From the flight-mechanics standpoints it has therefore been designed using fundamentally MIL-F8785-C requirement as a design criteria and generally good results have been achieved.
For some specific tasks the MIL Spec. turned out to be insufficient to fit the flying characteristics, so the need for more demanding requirements arose.
New criteria, in time and frequency domain, developed for highly unstable aircraft have been a good aid to evaluate the handling qualities in critical flight phases with the pilot in the loop. In particular they have been used, both for longitudinal and lateral control, for the landing task and to predict behaviour during precise tracking in weapon aiming or in flight formation.
A good help has been found also with some new criteria developed for high angle of attack and used to predict and to improve the aircraft manoeuvrability near the stall incidence angle and the aircraft departure and spin susceptibility in the post-stall area.

LANDING TASK

One of the most difficult tasks, which often involves serious safety problems, certainly is the landing task. In particular the performance landing on a short runway with a precision touchdown point tends to drive up pilot workload with a very tight loop closure. For this reason, AMX has been evaluated against some of the more recent landing criteria developed for "fly-by-wire" aircraft.

Longitudinal control in landing

The longitudinal control in landing is a very complex multitask that involves pitch control using the stick, and speed control using thrust and aerodynamic drag. The criteria for this task (Ref.1) usually consider only the short period step response at constant speed with flight path control as a primary task. For a conventional aircraft without a "lift control" it is achieved by the pitch control which cannot be decoupled from the flight path.
In such a context, the time delay t_γ between the step control input and the flight path angle response, could be equally or more valuable as a metric than the short period criteria, to which it is directly related.
Further parameters (defined in ref.1) affecting the pilot rating during the flare manoeuvre depend on the pitch acceleration and pitch rate response.
Suggested landing optimum level 1 values for these parameters are:

flight path angle delay	$t_\gamma < 1.5sec$
time at pitch rate peak	$0.5 < t_q < 1.1sec$
time at pitch accel. peak	$t_q < 0.25sec$
pitch rate ratio	$q_{max}/q < 3$
pitch attitude dropback	$DB/q < 1.25sec$

All these figures have generally been considered as guide-lines but the transient step response alone does not completely address the more complex landing problems like overcontrol or PIO conditions approaching the touch-down point.
In Ref.1 the time response criterion is integrated with a frequency response criterion,

reflecting the past experiences of its author. It's based on the assumption that, in the frequency range used by the pilot for the pitch aircraft control (up to 0.25-0.5 Hz) the phase lag shall be less than 120 deg. Furthermore in this range the response must be neither too sluggish nor too abrupt.
At higher frequency instead, the pitch control does not particularly influenced the aircraft response, nevertheless limits on the amplitude attenuation and on phase rate level are to be applied in order to avoid PIO conditions.
The fundamental behaviour observed in landing PIO is the unconscious stick activity called "stick pumping" that the pilot exerts during the runway approach a few second before the touch-down.
This activity occurs at a frequency where the pitch acceleration is nearly in phase with the stick deflection input and at an amplitude of about $6.5 deg/sec^2$

These time and frequency criteria have been proven a good guide-line to evaluate the landing controllability in the frequency range of aircraft control.
Figure 5 shows a classical time and frequency response against the above-mentioned criteria, for a typical landing condition in "High-Lift" configuration, low speed, gear down.
The time response parameters reflect the optimum range values and they are not particularly affected by the landing conditions. The frequency response fulfils the expected limits with a pilot gain 0.9 and a cross-over frequency at 120 deg phase lag of about 0.3 Hz.
The pumping frequency is about 1.1 Hz with a corresponding amplitude of:

$$\left(A\left(\frac{\theta}{\delta}\right)\right)_{\omega=1.1Hz} = 5.6 sec^{-2}$$

The stick deflection to reach the threshold pilot sensibility to the pitch acceleration of $6 deg/sec^2$ is:

$$\delta_{min} = \frac{\theta_{min}}{A} = \frac{6\ deg\,sec^{-2}}{5.6\ sec^{-2}} = 1.07 deg$$

Flight test data, shown in figure 6, confirm this figures. The pumping activity just a few seconds before the touch-down is performed by the pilot with a frequency and stick amplitude according to prediction and no tendency to overcontrol or divergent oscillation are noticeable.

Lateral control in landing

For the lateral control in landing flight phase, shortage of available data have not yet allowed to draw a specific criterion even if the basic rules of a quite large attenuation at 180 deg phase lag (15-20 db) and a crossover frequency higher than 1.0 Hz are to be considered useful to prevent PIO conditions. For lateral control no unconscious pilot activity, like the longitudinal stick pumping, is performed as a PIO catalyst during the landing approach.

The AMX lateral control is considered by the pilots satisfactory too, and adequate to allow a performance landing with a precise contact point.

During the flight testing different F.C.S. standards with various level of roll control sensitivity have been tested. Some occasional tendency to lateral divergent oscillation approaching the runway has been experienced particularly for short field landings where the pilot workload is somewhat increased.

Figure 7 refers to an AMX landing in the above-mentioned condition. The initial large lateral stick deflection, used to counteract an unexpected lateral atmospheric disturbances, demands large surface deflection reaching the actuator rate limit. The rate limit decreases considerably the stability margin of the open loop pilot-aircraft bank attitude control.

Figure 8 presents the lateral time and frequency response of a typical landing condition in "High lift" configuration, low speed, gear down. No criticalities come up from the linear analysis, but a different situation is highlighted if the control system non-linearities are considered.

Figure 9 shows a frequency response comparison between linear, non-linear small oscillation and non-linear large oscillation systems. The non-linear small oscillation is very similar to the linear response up to the higher frequencies where, due to a small dead-zone on spoilers deflection, a small loss of gain margin and phase rate is evident. For large stick inputs, instead, a large non-linearity due to the actuator rate saturation, considerably reduces the phase and gain margin ,thereby entering a PIO condition.

From the latter case emerges that: the pilot, using a "high gain" for different emergency situation, low visibility or unfavourable atmospheric condition, can reach a PIO condition due to large amplitude lateral stick motion. This could be promptly recoverable within few cycles if the stick deflection is kept to a minimum, thereby reducing the spoiler demand and avoiding the actuator rate limiting.

Generally, lateral stick activities similar to longitudinal stick pumping is not often seen in the landing phase. However, the flight test time-histories of figure 10 show a rare example of small lateral excitation corresponding to cross-over frequency of 1.1 Hz at 180 deg of phase lag (figure 9) without any tendencies toward divergent oscillations.

TRACKING CHARACTERISTICS

Both longitudinal and lateral-directional flying qualities of a combat aircraft are very important for a successful mission.
The tracking characteristics, infact, are highlighted in the most critical phases of an operative mission such as ground attack with either gun or weapon aiming mode, or during in-flight refuelling or close flight formation. However, it is during these phases when continuous small control inputs are necessary to reach the final or a constant line-up.

For a such important tasks, as a result of a specific investigation, some time and frequency criteria have been considered to evaluate the AMX "fine tracking characteristics.

Lateral-directional tracking requirement

The primary lateral-directional control task is the control of the bank angle by use of lateral stick. The equivalent transfer function relating the dynamics of this task can be obtained by reducing the high order system over the frequency range from 0.1 rad/sec to 10 rad/sec based on the principle of matching the bank angle to lateral control and the dutch roll to directional control (Ref.2):

$$\frac{p}{F_{as}} = \frac{L_{F_{as}} s(s^2 + 2\zeta_\phi \omega_\phi s + \omega_\phi^2) e^{-\tau_{ep} s}}{(s + 1/\tau_s)(s + 1/\tau_r)(s^2 + 2\zeta_d \omega_d s + \omega_d^2)}$$

It is evident that, when the complex dipole cancels out ($\omega_\phi = \omega_d; \zeta_\phi = \zeta_d$) the roll rate response is not contaminated by sideslip excursion in the dutch-roll mode and the major consequence is its non-oscillatory behaviour.

When dipole cancellation does not occur lateral-directional precision tasks both in the open and closed loop control are severely affected. A potential methodology that can be applied in this case is the Northrop criterion (Ref.3). To cancel the complex roots the criterion uses the magnitude ratio ω_ϕ/ω_d and the real axis location of the zero with respect to the dutch-roll pole $\zeta_\phi \omega_\phi / \zeta_d \omega_d$.

The cancellation depends mainly on the values of ω_ϕ and ω_d and to a lesser extent on ζ_ϕ and ζ_d. Hence the importance of ω_ϕ/ω_d as a parameter which determines proverse ($\omega_\phi/\omega_d >$ 1.) or adverse ($\omega_\phi/\omega_d <$ 1.) yaw tendency during the roll control.

All the interactions caused by this quadratic pair are lumped under the general heading of ω_ϕ/ω_d and $\zeta_\phi \omega_\phi / \zeta_d \omega_d$ effects, however several others parameters play an important role in the totality of effects, such as $1/\tau_r$, $1/\tau_s$, τ_{ep}, $|\phi/\beta|_d$.

For this reason the application of the requirement implies quite a number of guidelines which must be considered. The roll, spiral and dutch roll mode MIL requirements should first be met as well roll time delay; moreover, small to medium values of $|\phi/\beta|_d$ are preferred.

In Ref.4 correlation of pilot rating with the parameter ω_ϕ/ω_d exhibits different trends as a function of $|\phi/\beta|_d$, especially with low ζ_ϕ and ζ_d leading to:

$\omega_\phi/\omega_d = 1.0$ for $|\phi/\beta|_d$ small
$0.75 < \omega_\phi/\omega_d < 1.0$ for $|\phi/\beta|_d$ medium to large

For large ζ_ϕ and ζ_d as such as for highly augmented aircraft meeting level 1 requirements. $\omega_\phi = \omega_d$ is generally preferred.

A limited fixed base simulation of the lateral directional tracking criterion has been carried out using an AMX aircraft. Several F.C.S. configurations have been considered, the nominal along with the degraded states. These configurations, all for the same flight condition (one of the most critical) have been reported in figure 11.

The simulation activity has been performed using Aeritalia's fixed base simulator. Formation flight has been simulated using a computer generated image of the lead aircraft flying in the same direction. The pilot was asked to maintain the fixed vector displayed on the H.U.D. exactly on the nozzle of the model in level flight or in a 45 deg bank turn manoeuvre. Of course during the whole manoeuvre (30 sec) the yaw control was free and minimum use of longitudinal control was recommended.

The average and integral errors on lateral and longitudinal motions were monitored to indicate the quality of the tracking task and the stick activity provided a measure of the pilot workload. Both these parameters were used as a comparison term among the various cases to establish a correlation with the analytical prediction (lateral directional tracking criterion).

The pilot comments for the different condition were:

1	FULL F.C.S.	Not extremely easy to control in roll due to the sluggish roll response but acceptable.
2	C/F OFF	Difficult to control for the roll and yaw oscillation developed during the task (cross-feed off).
3	R/D OFF	Easier than 1 because the faster roll response and the possibility to quicker stop the bank angle (roll damper off).
4	Y/D OFF	Very difficult to perform the tracking task because of the divergent oscillations (yaw damper off).
5	R/D+Y/D OFF	The same as case 4
6	C/F+R/D OFF	Yaw oscillation,the roll control seems easier than case 2
7	C/F+Y/D OFF	Strong yaw oscillations, similar to case 6
8	C/F+R/D+Y/D OFF	More difficult than case 6 because of the higher oscillation in roll and yaw.
9	G1=2/3*G	Easier than case 1 (reduced gain aileron/spoiler).

The average error of the different F.C.S cases was compared in figure 12 and in general a good correlation of error levels with pilots comment was found.

The nominal condition (full F.C.S.) has been found ightly difficult to control due to the sluggish roll response even if the roll time constant meets the level 1 MIL Spec. A better performance has been found for conditions 3 and 9, infact with R/D off lower roll time constant leads to an improvement for the roll control according to the pilot opinion.

The worst cases were conditions 4 and 5 because of the low damping (level 3) and $\omega_\phi > \omega_d$ leading to pilot induced oscillation. Points 6,7 with $\omega_\phi < \omega_d$ were considered conditions quite difficult to control but they were found to satisfy level 2 of handling qualities unlike the boundaries in the criterio...

The performance for conditions of point 8 were bordering on level 3, but still preferable to cases 4 and 5.

Case 9 with reduced gain aileron/spoilers results in a good control for the following reason:

$$C_n\delta_a / C_l\delta_a < 0. \quad \text{for } \delta_a \quad \text{(adverse)}$$
$$C_n\delta_a / C_l\delta_a > 0. \quad \text{for } \delta_{sp} \quad \text{(proverse)}$$

The reduction of spoilers deflection leads to a decrease in proverse yaw, approaching the best condition of $\omega_\phi/\omega_d=1$.

A general agreement has been found between the pilot opinion and the analytical predictions based on the lateral-directional tracking criterion. The left hand limits of the above criterion seems to better define the tracking difficulty, while, according to our investigation, the exact position of the right hand limits is disputable.

<u>Extension of the tracking criterion to the weapon aiming task</u>

The above criterion has been used to predict the tracking characteristics in the following tasks:

- In-flight refuelling
- close flight formation
- gun aiming
- weapon aiming

For each task, application of the criterion predicted tracking characteristics in the level 1 region for all the considered flight conditions and external store configurations.

Analytical predictions were in good agreement with the pilot rating and the post-flight analysis for all the task except the last one.
For this case, significant discrepancies were noted for certain kind of external stores and pilot rating of 4 (Cooper Harper scale) was achieved against a predicted optimum condition.

Following a careful analysis of the flight test results a parameter has been focused which bridges the gap between the analytical predictions and the flight results.

Whereas gun aiming can be related mostly to the heading control, the precise tracking during weapon aiming with the "continuous computed impact point" (CCIP) is in a good approximation a bank angle control.
The steering line displayed on the HUD, representing the lateral displacement of the bomb impact point, is, infact, correlated to the bank angle as shown in figure 13

The ratio of altitude release to the downrange distance of the bomb, (H/X), is a key factor in establishing the relationship between the ϕ/ϕ_c and $\Delta_{CCIP}/\Delta_{CCIP_c}$ transfer functions (figure 14).

Figure 14 shows the frequency response for the above transfer function for two different kind of bombs. h/X ratio depends on the balistic characteristics of the bombs and the flight conditions of the release. As the ratio approaches to 1.0 the aiming task appears to the pilot as difficult as the lateral control task. In case of low speed, high dive angle or for delayed bombs, (i.e. $H/X>1$), the frequency response curve moves toward the closed loop resonance area and lateral PIO conditions are achieved.

Figure 15 illustrates this concept in the time-domain through a 6 d.o.f. non-linear simulation in which the weapon aiming algorithm is accounted for. An ideal pilot (a pure unity gain) acts on the lateral control in the attempt to minimize the error between the reference value for the azimuth (1.0 degree) and the actual one, settled initially to 0.0.

With $H/X=1.0$ a good acquire is reached without overshoot or oscillations. Reducing the downrange distance of the bomb, X, with exaggerated bomb drag to have $H/X=5.6$ (degradation of open loop gain of approximately 15 db) the steering line control becomes extremely difficult and, due to the increase of the system gain, continuous lateral control oscillation with low damping is induced by the pilot.

On the basis of the above consideration on the diagram of figure 16 the line $H/X=1.0$ reflects the conditions for which the weapon aiming task is very similar to the aircraft lateral controllability. The area standing on the right side of the line $H/X=1.0$ indicates conditions of an easy task, whereas proceeding on the left side a more and more difficult task and even PIO occurrence are to be expected.

Flight tests confirm these tendencies, figure 17 shows a different target tracking behaviour for two different flight conditions corresponding to the points A and B on figure 16. The point A represents a normal ground attack condition and a good tracking is achieved with a small pilot workload. The point B, instead, in the area of reduced controllability, refers to a test outside the bomb delivery envelope with the pipper elevation out of the HUD displayed field. In the latter case the pilot is able to stabilise the steering line on the target with enormous workload.

Once the relationship has been established between the A/C lateral controllability and the quality of the weapon aiming task, it is legimate to extend the tracking criterion to those weapon aiming conditions for which the spurious effects related to the bomb characteristics (H/X) are minimized (i.e. the conditions related to the $H/X=1.0$ line).

The validity of such an assumption has been verified through the analysis of the flight records, the still-photos of the HUD videorecorder and the pilot ratings.

The relative error between the target and the CCIP steering line during precise tracking has been evaluated against the level 1 or 2 "Median Lateral Error" criterion as well as the "Lateral Error Cumulative Distribution Function" requirement, according to which the level 1 or level 2 median error shall be 50% or greater.

It has been found that, at different flight conditions and with different kinds of bombs, all spreaded around the $H/X=1.0$ line, were appreciated by the pilots and resulted in agreement with the "median lateral error" (figure 18).

In this way, from all the conditions which fulfilled the "fine lateral tracking criterion" it is possible to pick-up those which are expected to be satisfactory also for the weapon aiming, $(H/X=1.0)$. On the other hand the unsatisfactory aiming situation can be improved by acting on the release conditions in such a way that $H/X<1$.

HIGH INCIDENCE BEHAVIOUR

The Northrop and Weissman departure criteria have been applied to predict the departure conditions and to estimate the effectiveness of the corrective actions undertaken both via aerodynamic changes and via FCS modifications.

The considered parameters are:

$$Cn_{\beta_{dyn}} = Cn_\beta * \cos\alpha - \frac{Iz}{Ix} * Cl_\beta * \sin\alpha$$

which measures the yaw departure susceptibility when the incidence angle is increased as result of the longitudinal control application.

$$LCDP = Cn_\beta - Cl_\beta * \frac{Cn_{\delta a} + \frac{K_{sp}}{K_a} * Cn_{\delta sp} + \frac{K_r}{K_a} * Cn_{\delta r}}{Cl_{\delta a} + \frac{K_{sp}}{K_a} * Cl_{\delta sp} + \frac{K_r}{K_a} * Cl_{\delta r}}$$

The lateral control departure parameter predicts the roll reversal when the rolling moment due to the adverse yaw overcomes the lateral control power.

Roll Performance at high angle of attack

The roll control of the AMX has been designed to meet the requirement of the MIL-F-8785C expressed in terms of time-to-bank defined for 1 "G" flight condition and for 80% of the minimum and maximum operational load factor.

Beyond the above normal load factor limits, no specification is given for the roll performances, while from an operational standpoint it is desirable to maintain a suitable lateral manoeuvrability.

In the α range approaching the stall, AMX exhibited unsatisfactory roll performances due to a rapid change of lateral and directional stability, along with the loss of lateral control effectiveness. Albeit the MIL requirement was fully satisfied the roll capability resulted as being a limiting factor for the maximum angle of attack allowed.

The phenomena, increasing the incidence angle, were at first evident as a roll hesitation and then as a roll reversal.

Figure 19 shows the in-flight results relevant to an objectionable situation. With a sustained full lateral stick, at an angle of attack near to the stall a large sideslip build-up generates a roll acceleration inversion with a consequent roll-reversal.

Predictions with the above mentioned departure criteria, reported in figure 20, show that the aircraft without crossfeed exhibit a rapid loss of LCDP which becomes negative at α-values lower than the α-stall by some degrees. The point I, in figure 20, is the corresponding point for the in-flight manoeuvre of figure 19.

To improve the roll performances within this area, up to the stall, a roll-to-yaw crossfeed has been designed which cause the rudder to be deflected as a function of the lateral and longitudinal stick displacement. The advantages associated with this solution are shown in figure 20: the $LCDP - vs - Cn_{\beta_{dyn}}$ curve, in the α range up to the stall moves towards the positive value of LCDP.

Flight tests confirmed the predicted improvement for the roll characteristics up to the stall angle as can be seen in the "time-histories" presented in figure 21 related to the points II and III on the previous analytical criteria. The roll manoeuvre performed at the α-stall (point III) is still acceptable, even if the low LCDP contributes to generate continuous roll oscillation.

Improvements in terms of roll rate versus angle of attack are shown in figure 22.

Stall behaviour

The approach to the stall and post-stall flight trials has been carried out through a gradual work-up according to the following steps:

- increasing the nose-up trim setting up to its full authority
- moving the CG rearward
- increasing the energy entry
- sustaining the stall up to 15 sec

The aircraft motion beyond stall is characterized by a bounded wing rock which develops into a slow nose slice (yaw rate 2-3 deg/sec) when the stick is held hard back.

The wing rock is a classical and regular roll oscillation developed as a divergent dutch-roll with frequency proportional to the square root of l_β and amplitude limited by the non-linear roll damping characteristics. Time-histories of figure 23 refer to a flight test record reflecting the conditions of the point IV in the afore-mentioned criteria.

Stalls aggravated with clinical manoeuvres, such as full back stick-full lateral stick, showed, for the clean configuration, a moderate rolling departure. Figure 24 shows the flight recorded time-histories of such a behaviour matched with a 6 d.o.f. simulation program. Moderate angular rates are reached after departure and due to good longitudinal control effectiveness a very fast recovery is possible after controls release.
This is in a good agreement with the departure criteria that for the point V predicts acceptable stall with mild rolling departure and low spin susceptibility.

Other conditions, like flap setting or store configuration have been evaluated against the departure criteria and a good agreement with the prediction has always been found. For example an additional increase in departures resistance is predictable and was experienced with manoeuvre flap setting.

For this reason, to improve the combat capability while reducing the probability of departure and the pilot workload, especially with underfuselage load or asymmetric store, an automatic deployment of manoeuvre devices will be shortly introduced.

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Ref.2 J.Hodgkinson, W.J.LaManna "Equivalent System Approaches to Handling Qualities Analysis and Design Problems of Augmented Aircraft" AIAA Atmospheric Flight Conference, Hollywood, FL, August 1977

Ref.3 J.T.Gallagher, W.E.Nelson Jr. "Flying Qualities of the Northrop YF-17 Fighter Prototypes" Business Aircraft Meeting, Wichita March 1977

Ref.4 Anon., "Flying Qualities Requirement for United States Navy and Air Force Aircraft", AGARD-CP, October 1961

Ref.5 A.M.Skow, A.Titiriga Jr. "A Survey of Analytical and Experimental Techniques to Predict Aircraft Dynamic Characteristics At High Angle of Attack" AGARD-CP-235, Athene, May 1978

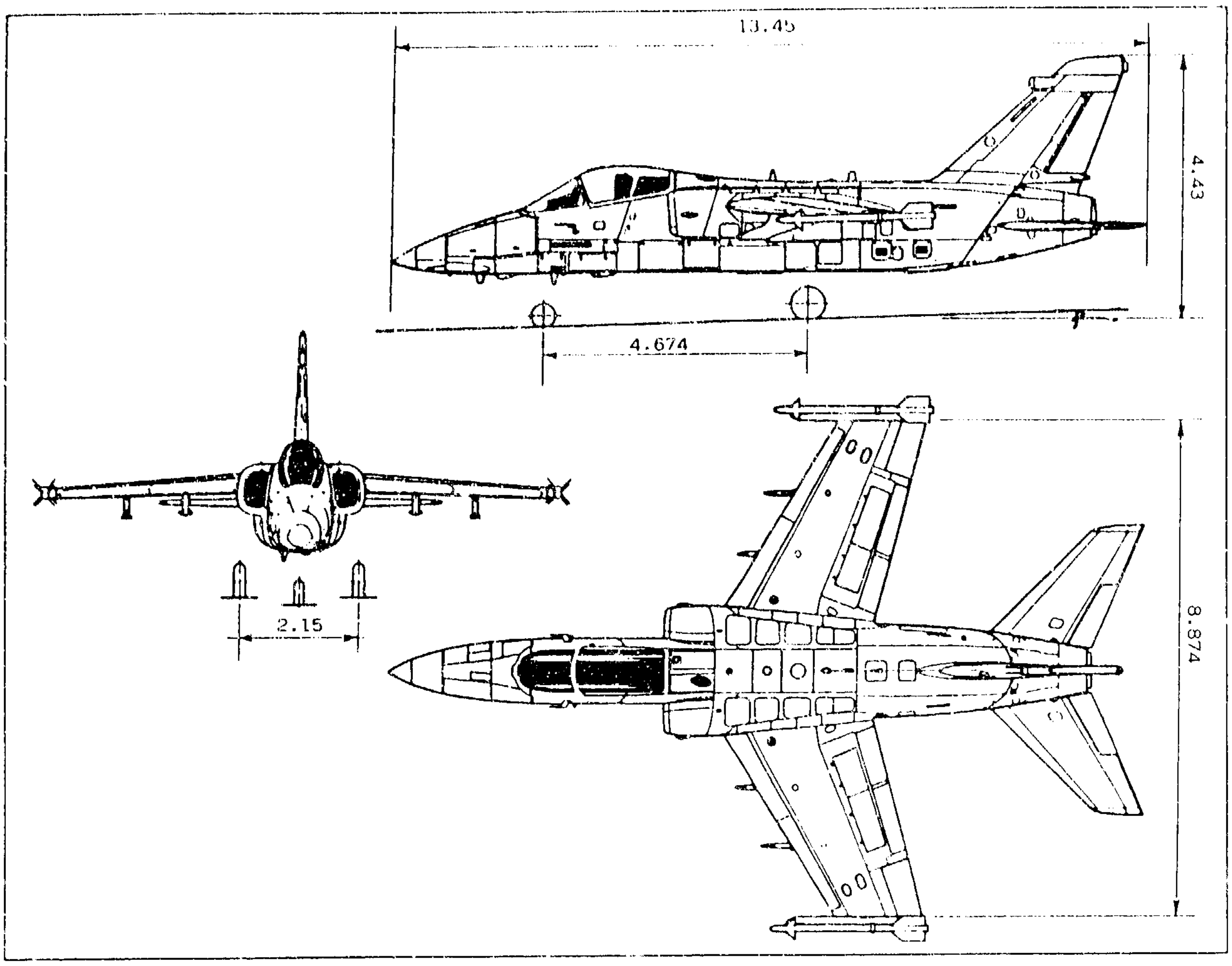


Figure 1 AMX THREE VIEW

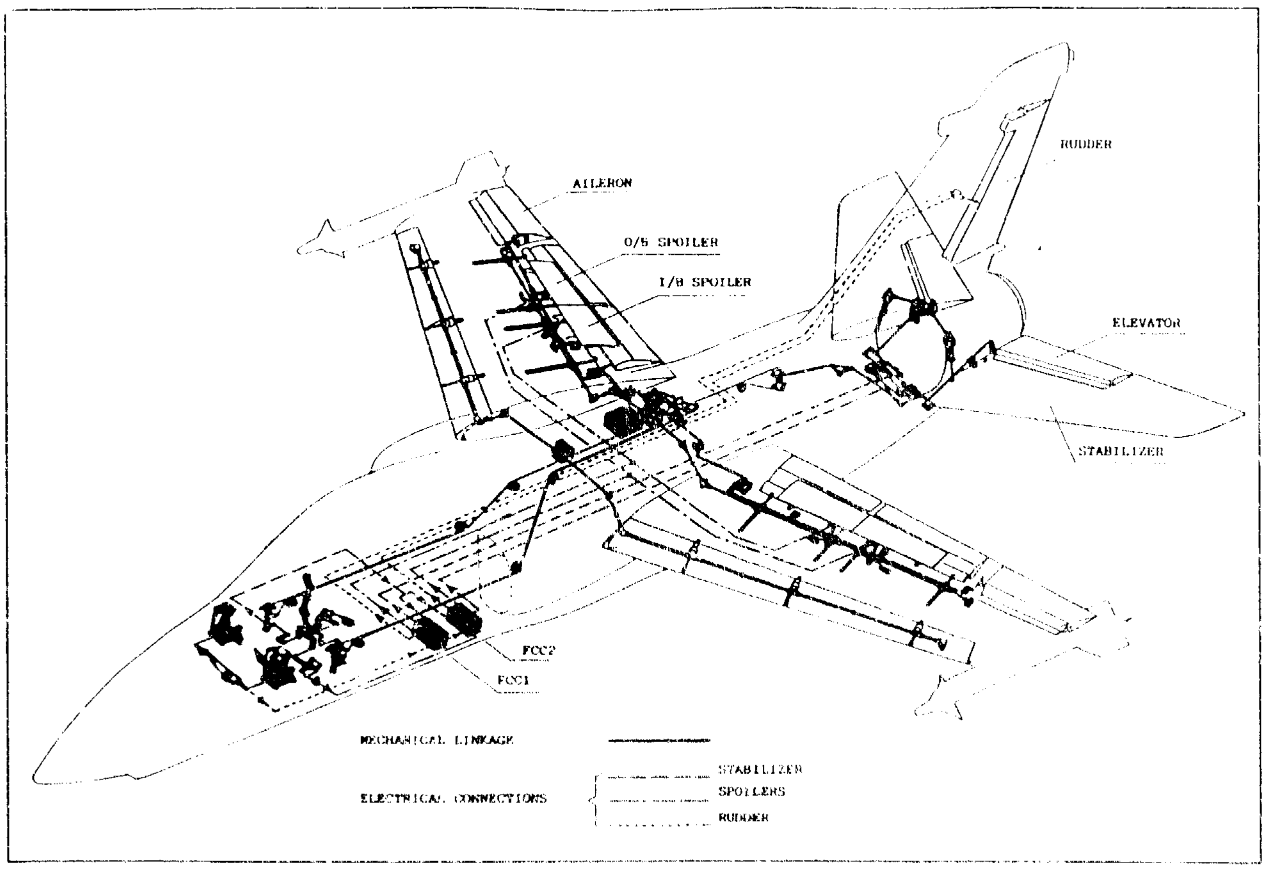


Figure 2 AMX-FLIGHT CONTROL SYSTEM LAYOUT

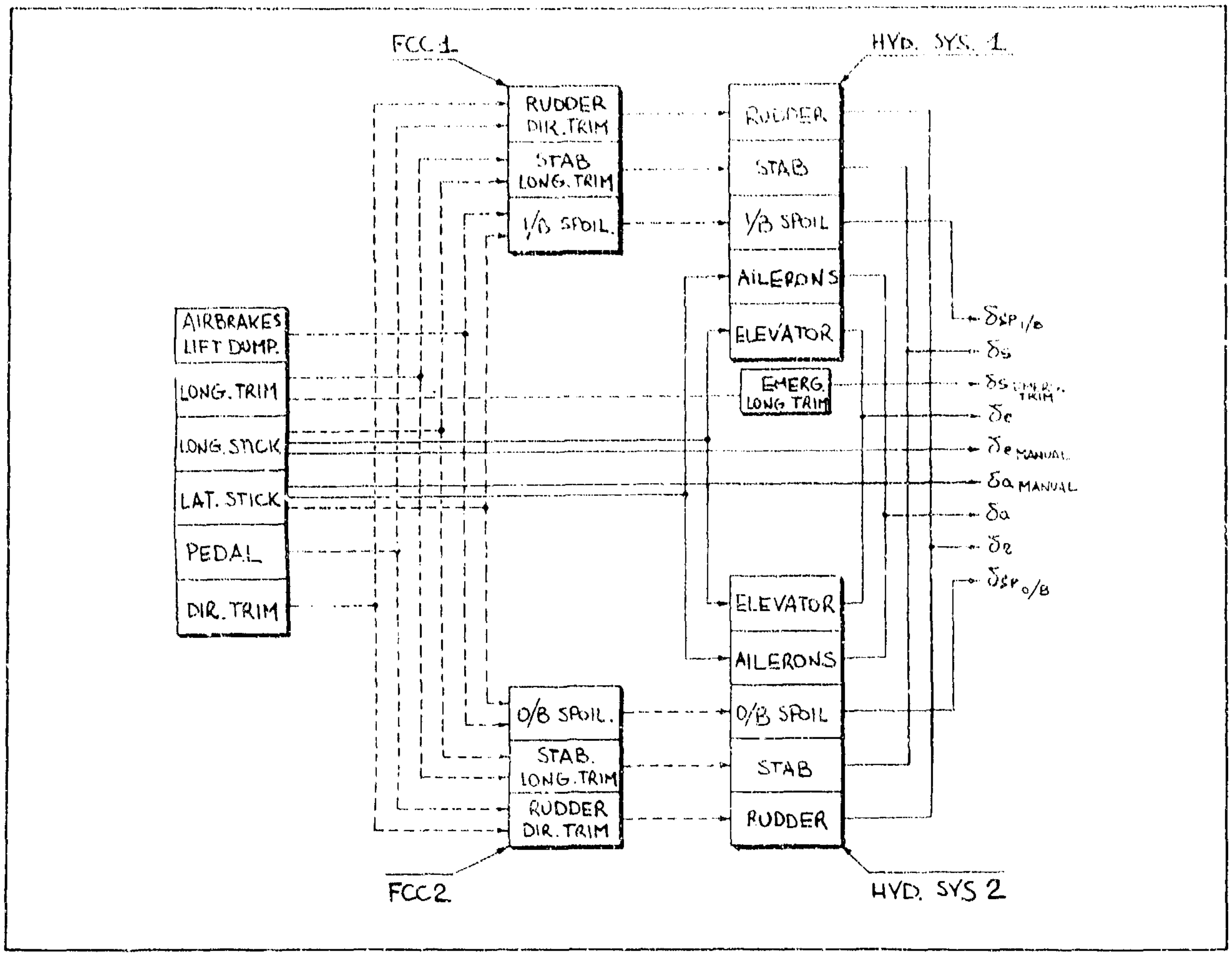


Figure 3 AMX-ELECTRICAL,HYDRAULIC AND MANUAL CONTROL CONNECTIONS

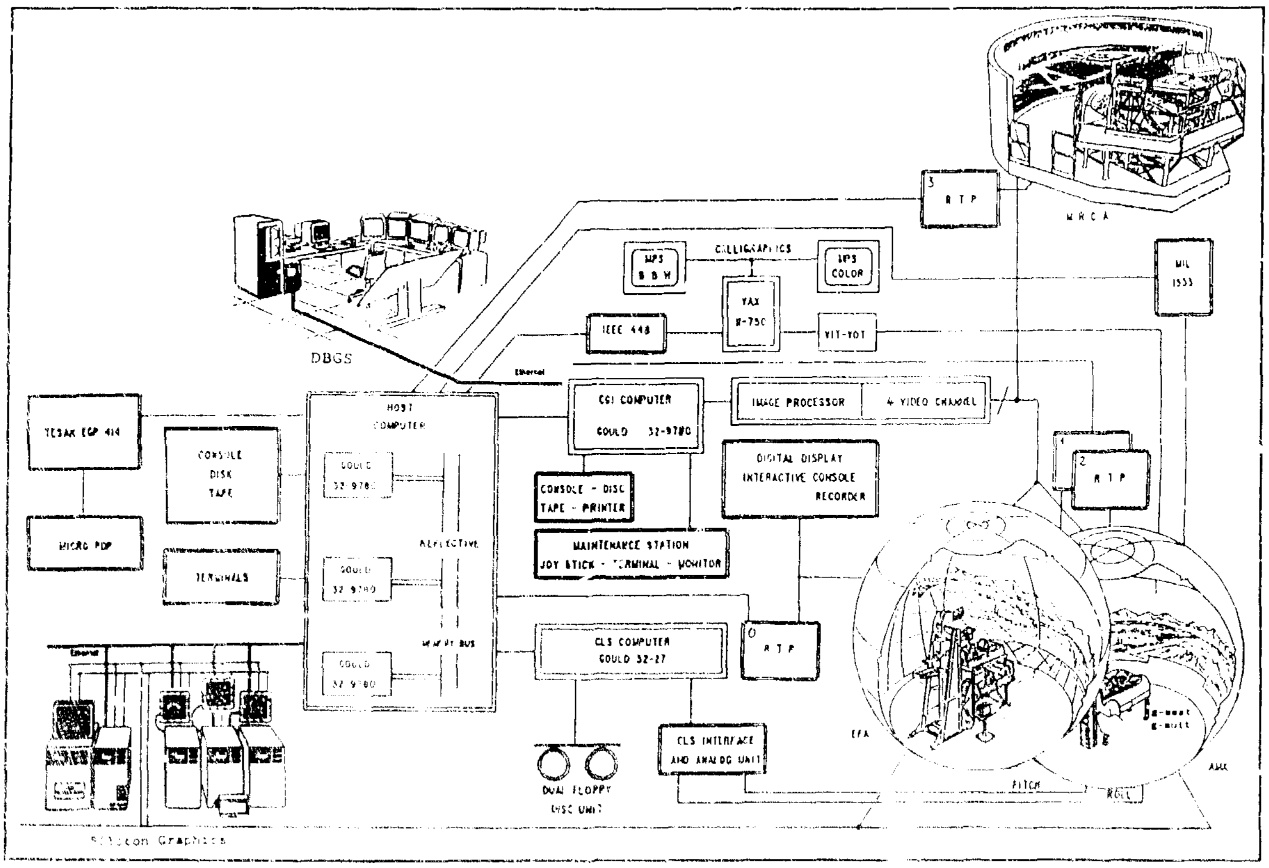


Figure 4 AIT-FLIGHT SIMULATION FACILI IES

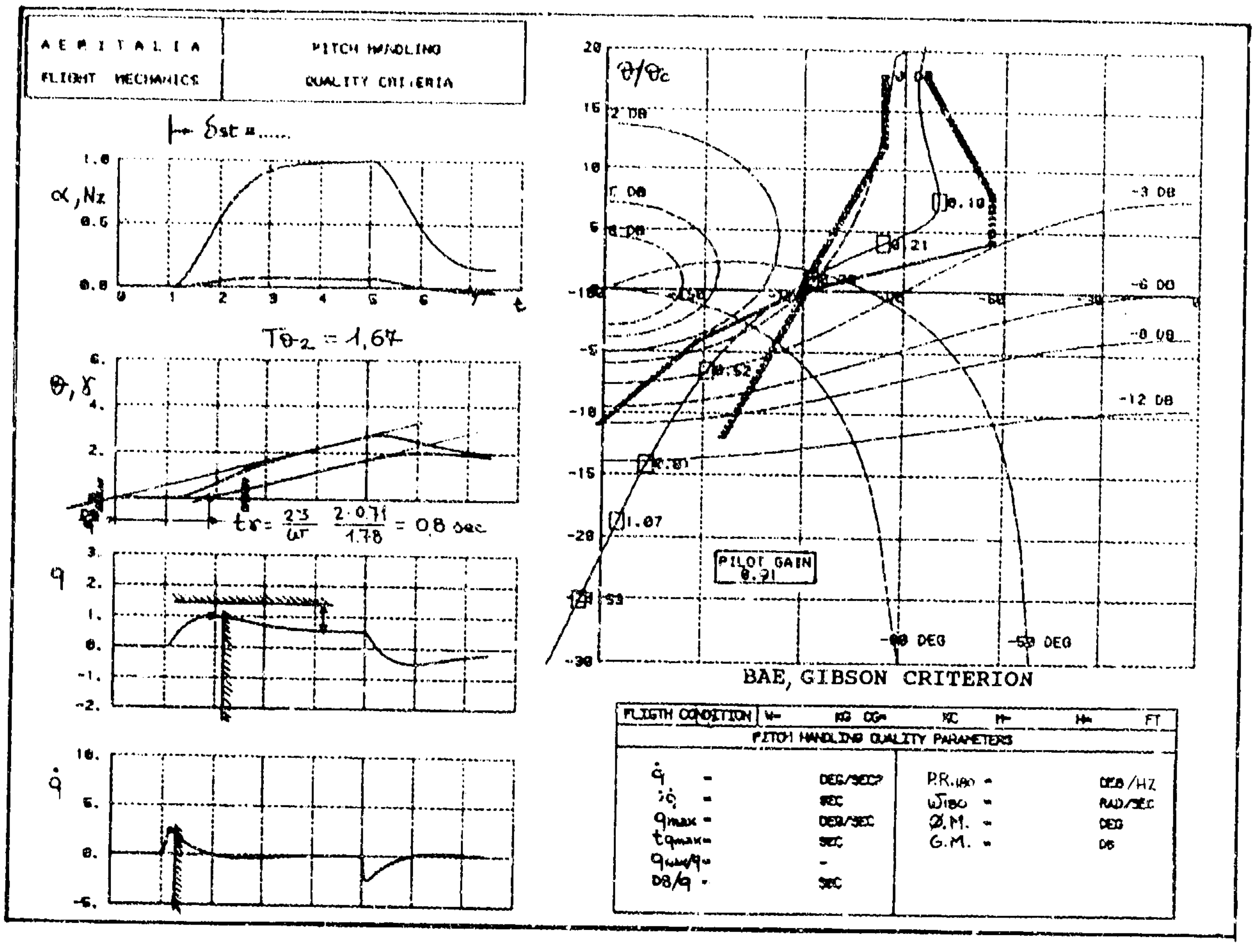


Figure 5 AMX-LANDING APPROACH PITCH HANDLING QUALITIES

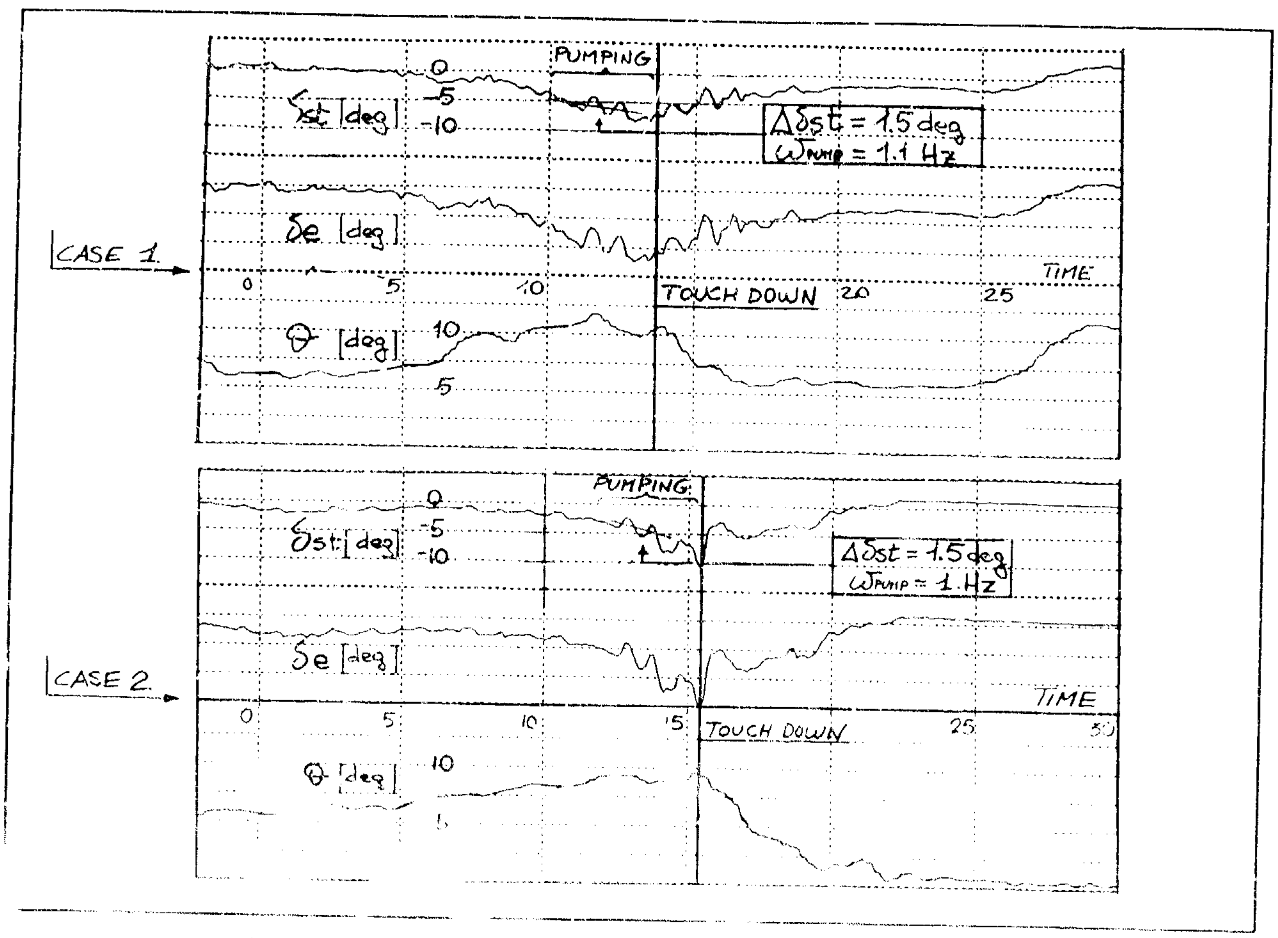


Figure 6 AMX-LANDING APPROACH FLIGHT TEST TIME-HISTORIES

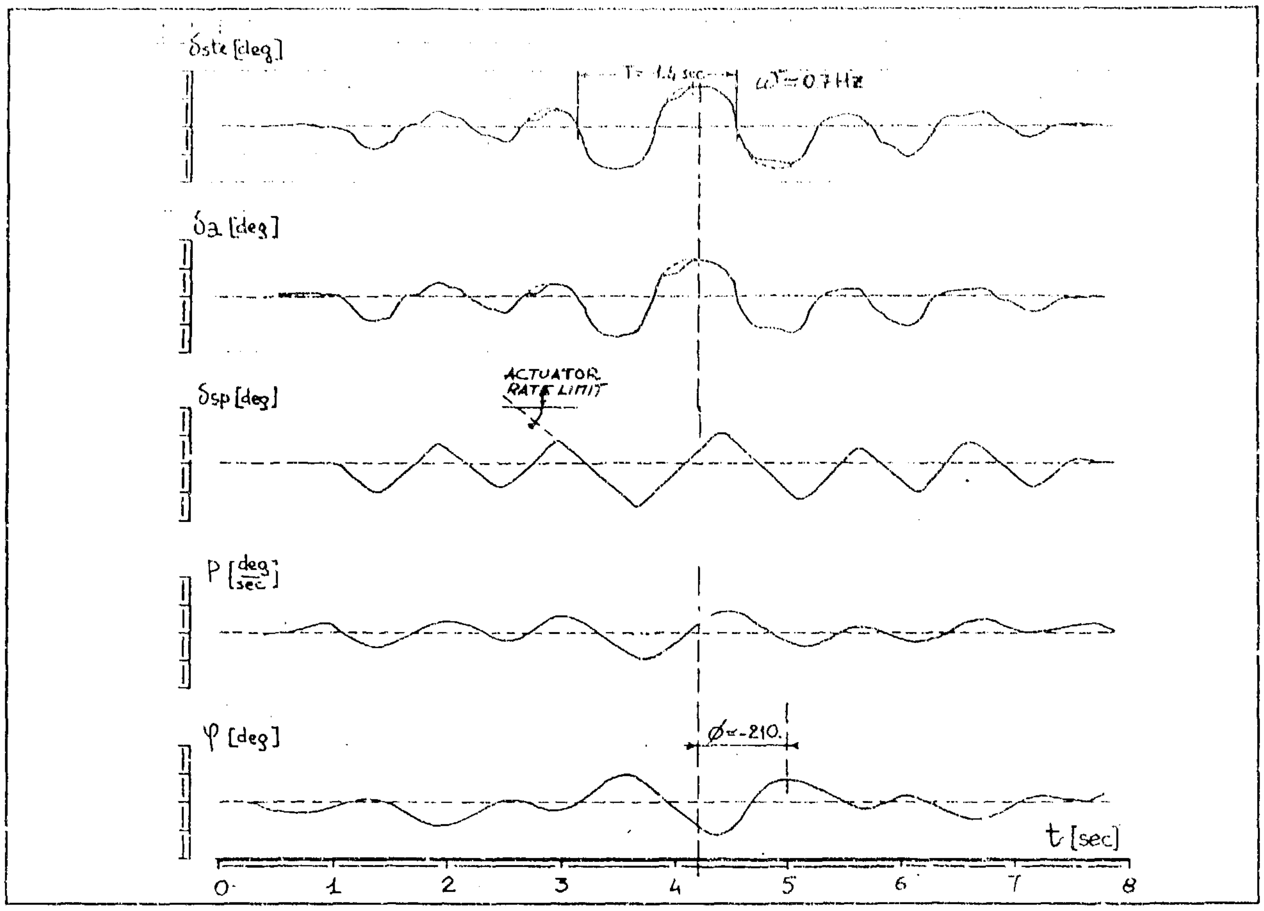


Figure 7 AMX-LANDING APPROACH LATERAL PIO WITH A NON-STANDARD FCS

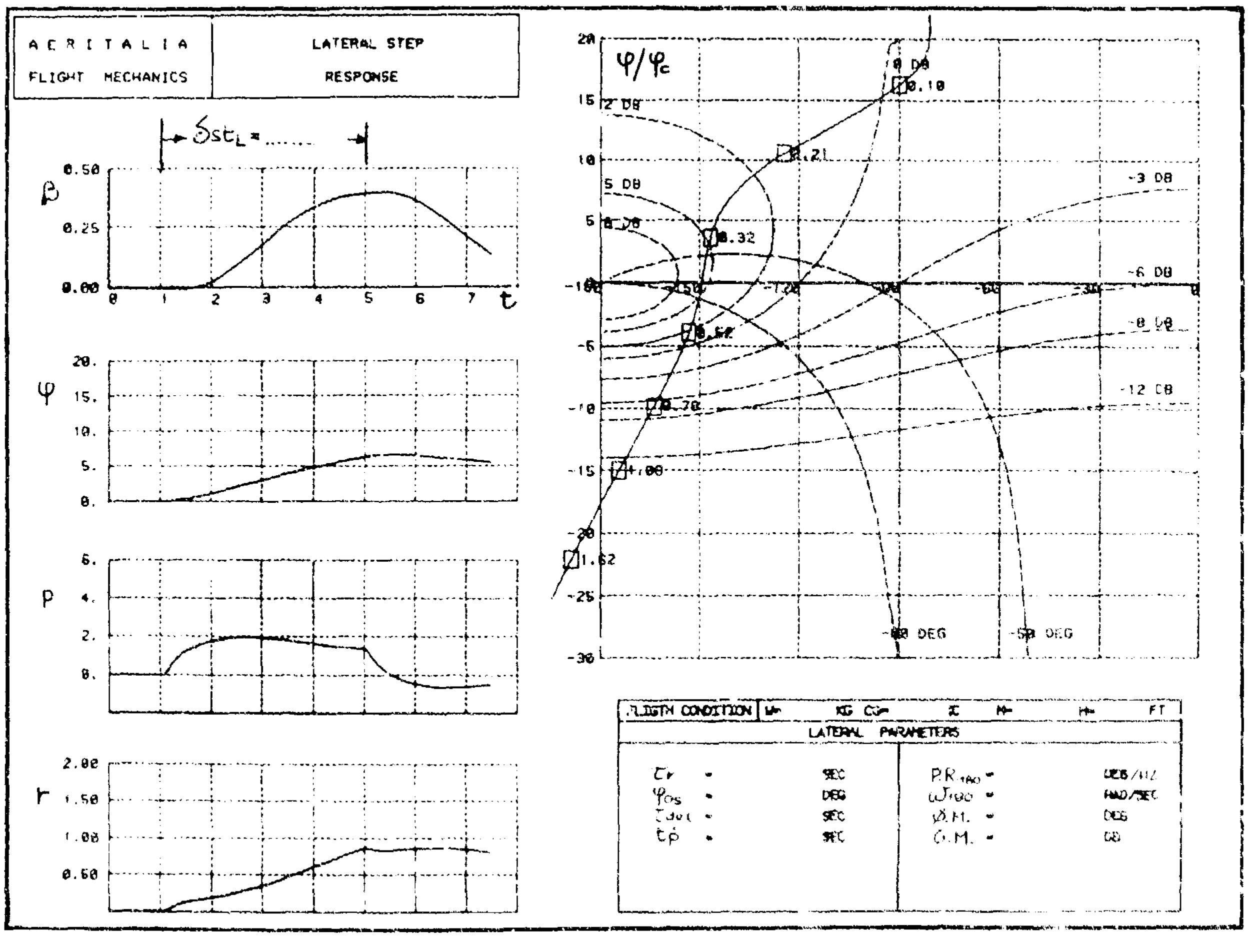


Figure 8 AMX-LANDING APPROACH LATERAL HANDLING QUALITIES

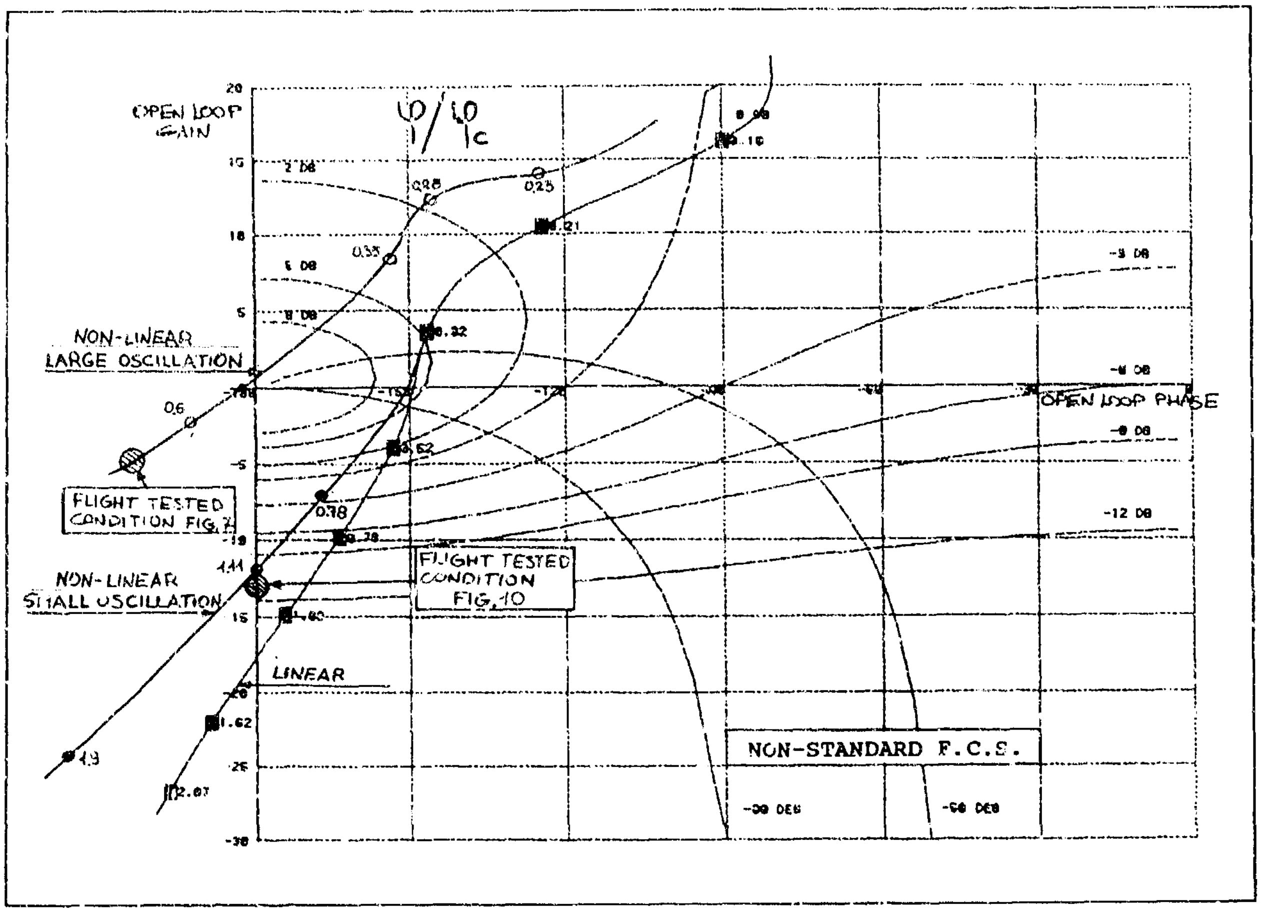


Figure 9 AMX-LANDING APPROACH LATERAL FREQUENCY RESPONSE

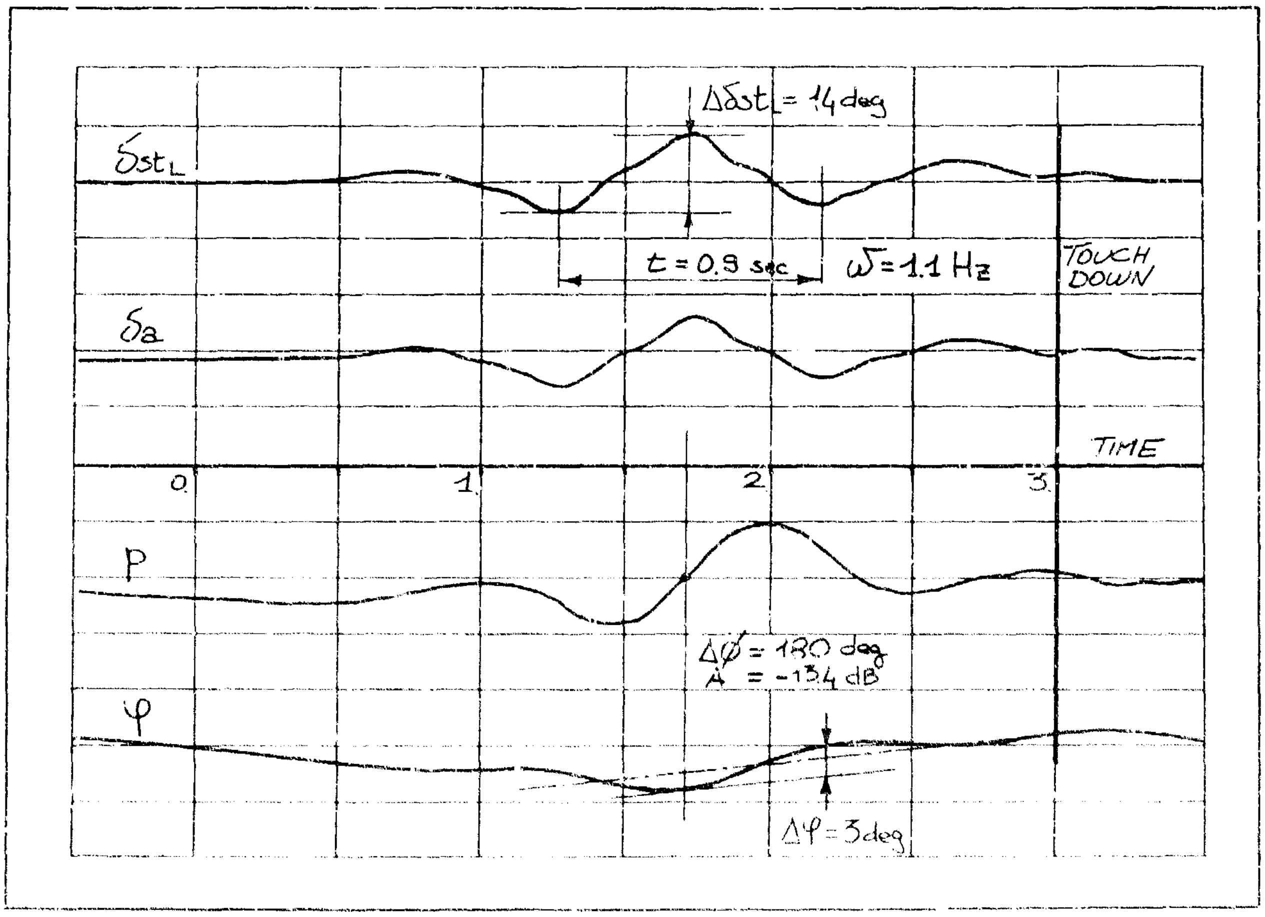


Figure 10 AMX-LANDING APPROACH SMALL LATERAL EXCITATION

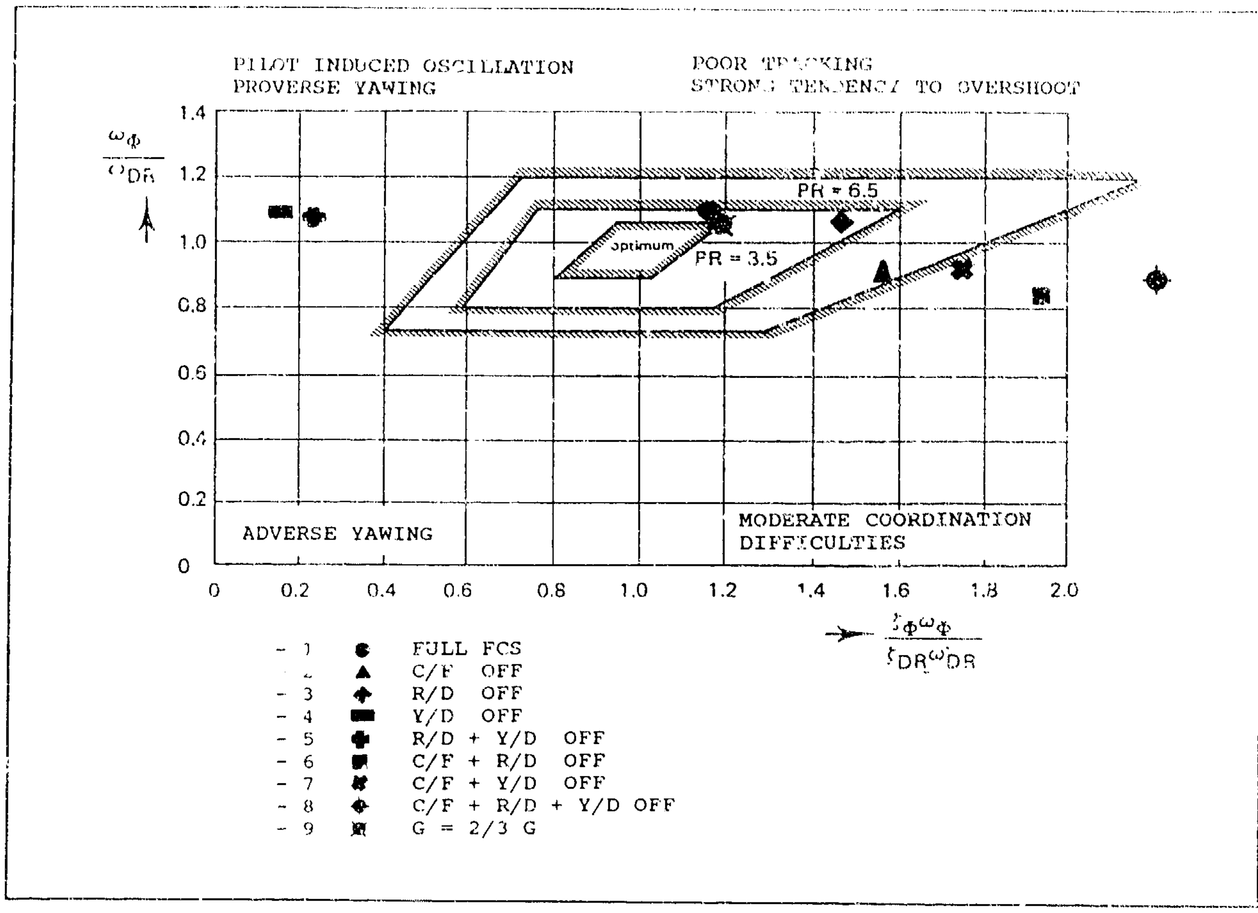
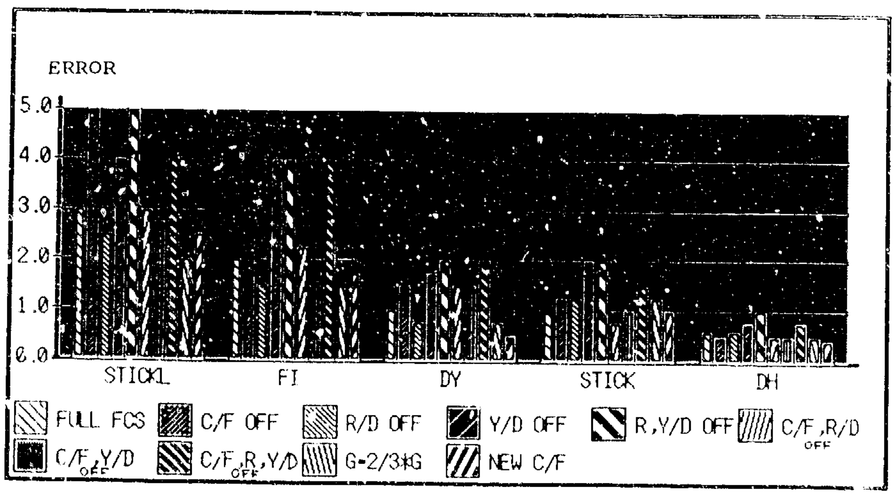


Figure 11 AMX - LATERAL DIRECTIONAL TRACKING PARAMETER



LIST OF SIMBOLS

F.C.S. : flight control system
C/F : cross-feed
R/D : roll-damper
Y/D : yaw-damper
G=2/3*G: reduced gain aileron/spoiler

Figure 12 AMX - CLOSE FLIGHT FORMATION TRACKING ERROR

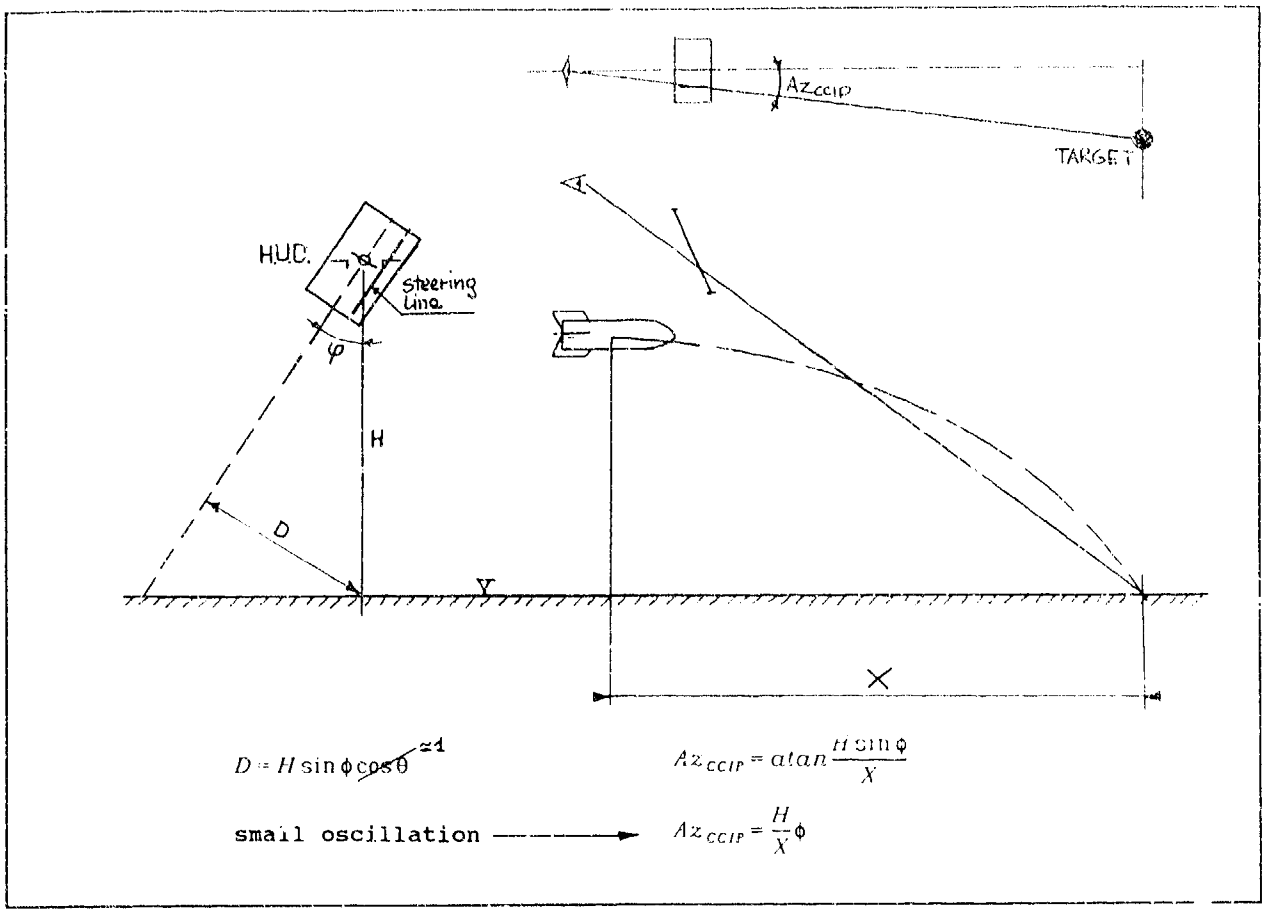


Figure 13 RELATIONSHIP BETWEEN CCIP AZIMUTH AND BANK ANGLE

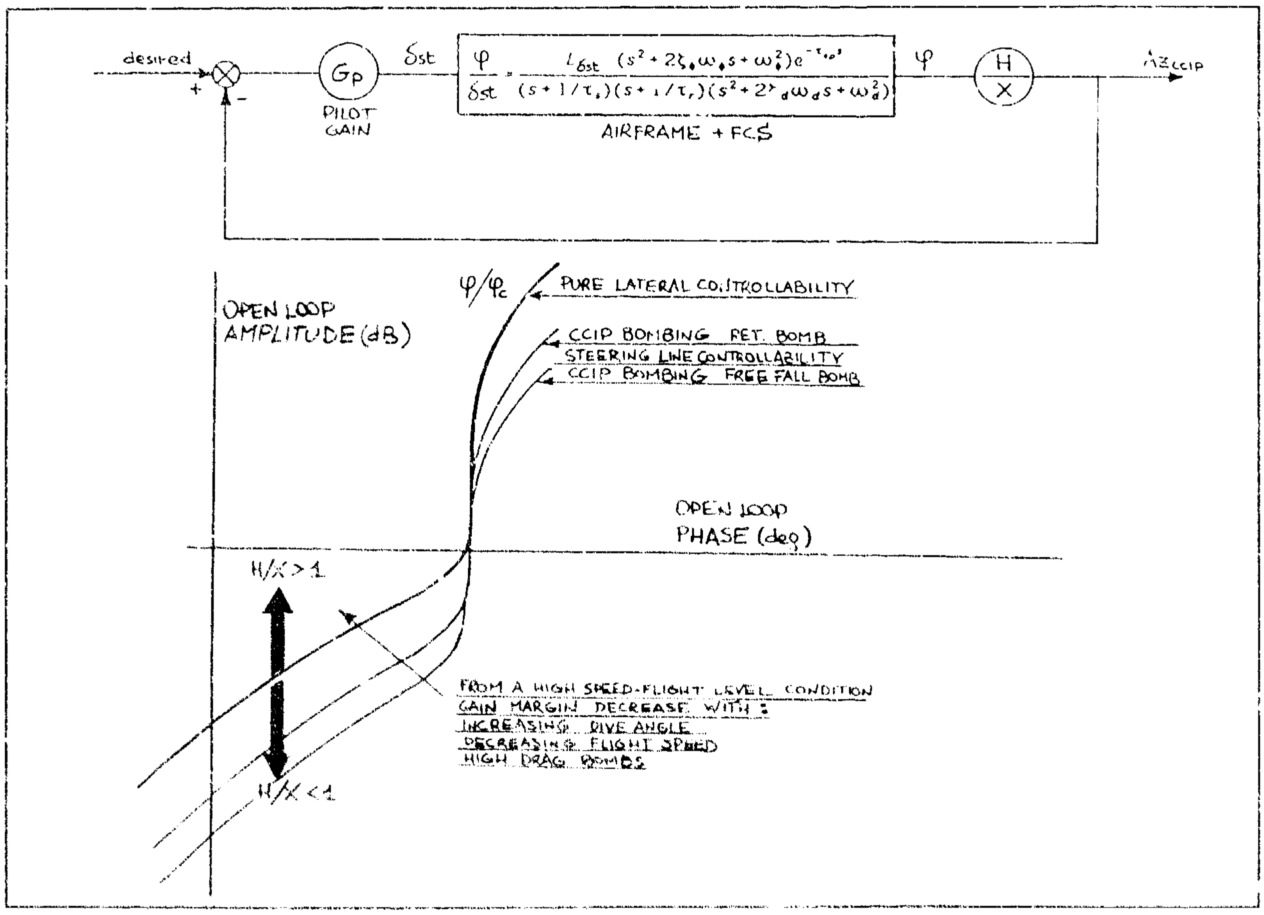


Figure 14 AMX - CCIP STEERING LINE CONTROLLABILITY

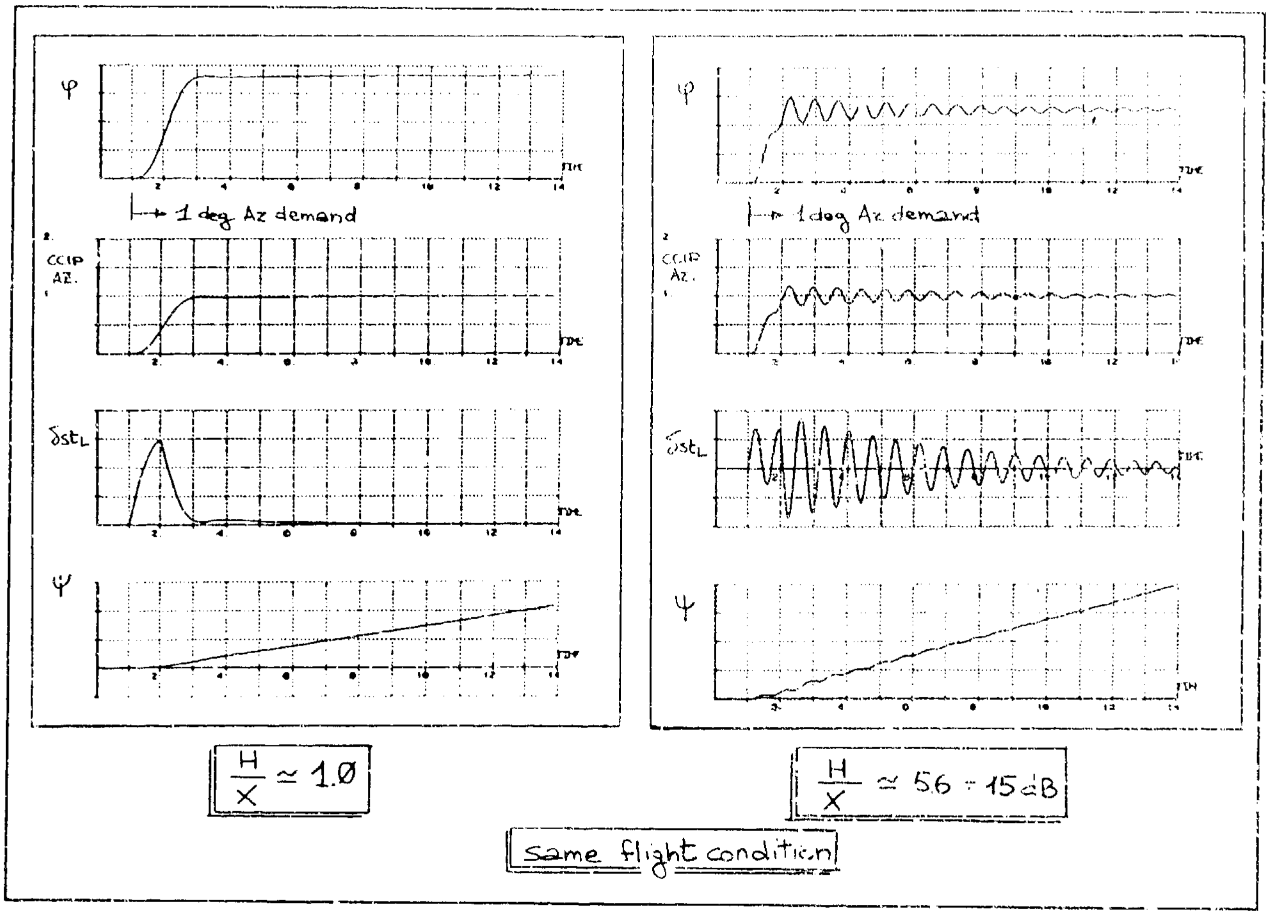


Figure 15 AMX - CCIP AZIMUTH ACQUIRE WITH DIFFERENT H/X RATIO

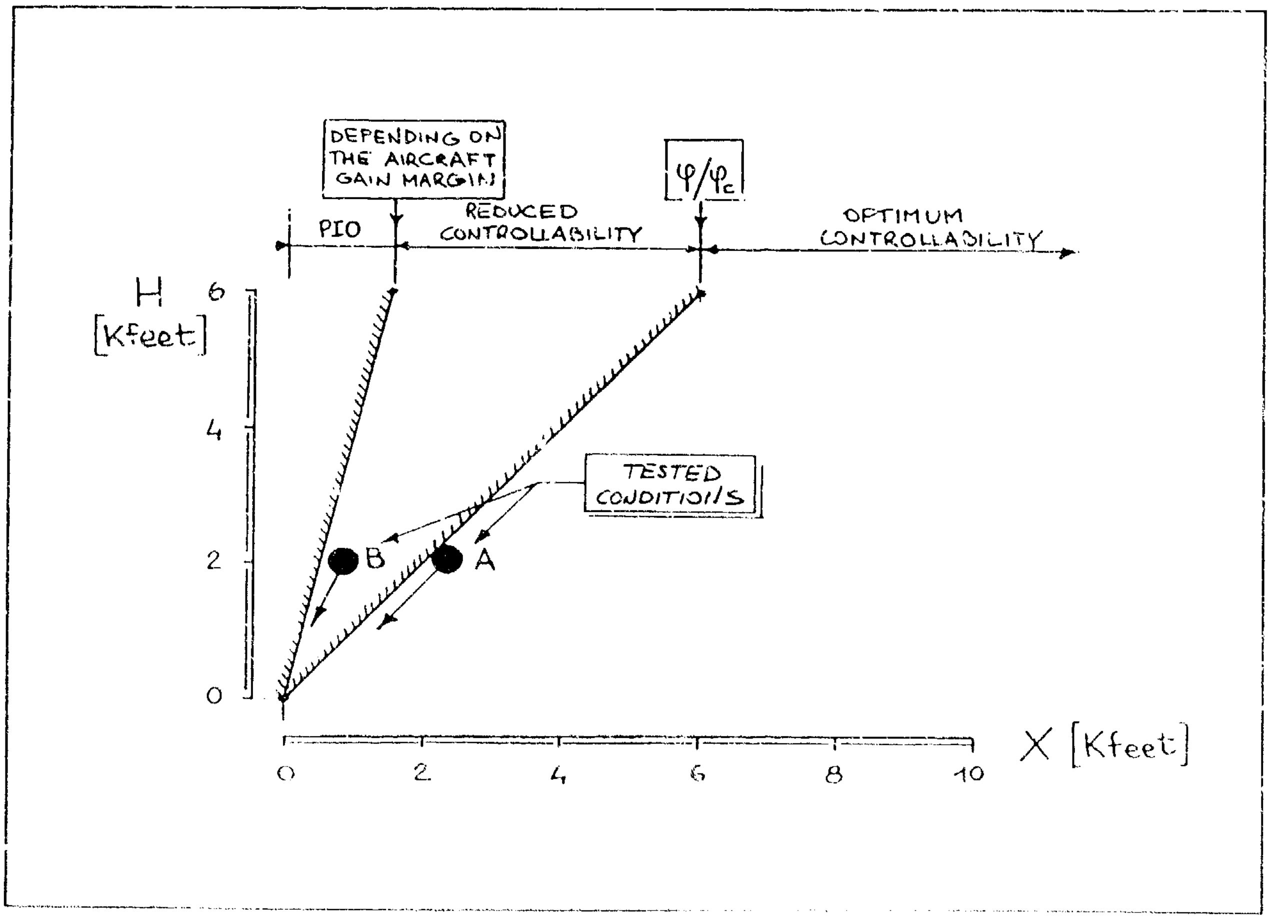


Figure 16 LIMITS ON ALTITUDE DELIVERY H VERSUS DOWNRANGE DISTANCE X FOR GOOD CCIP STEERING LINE CONTROLLABILITY

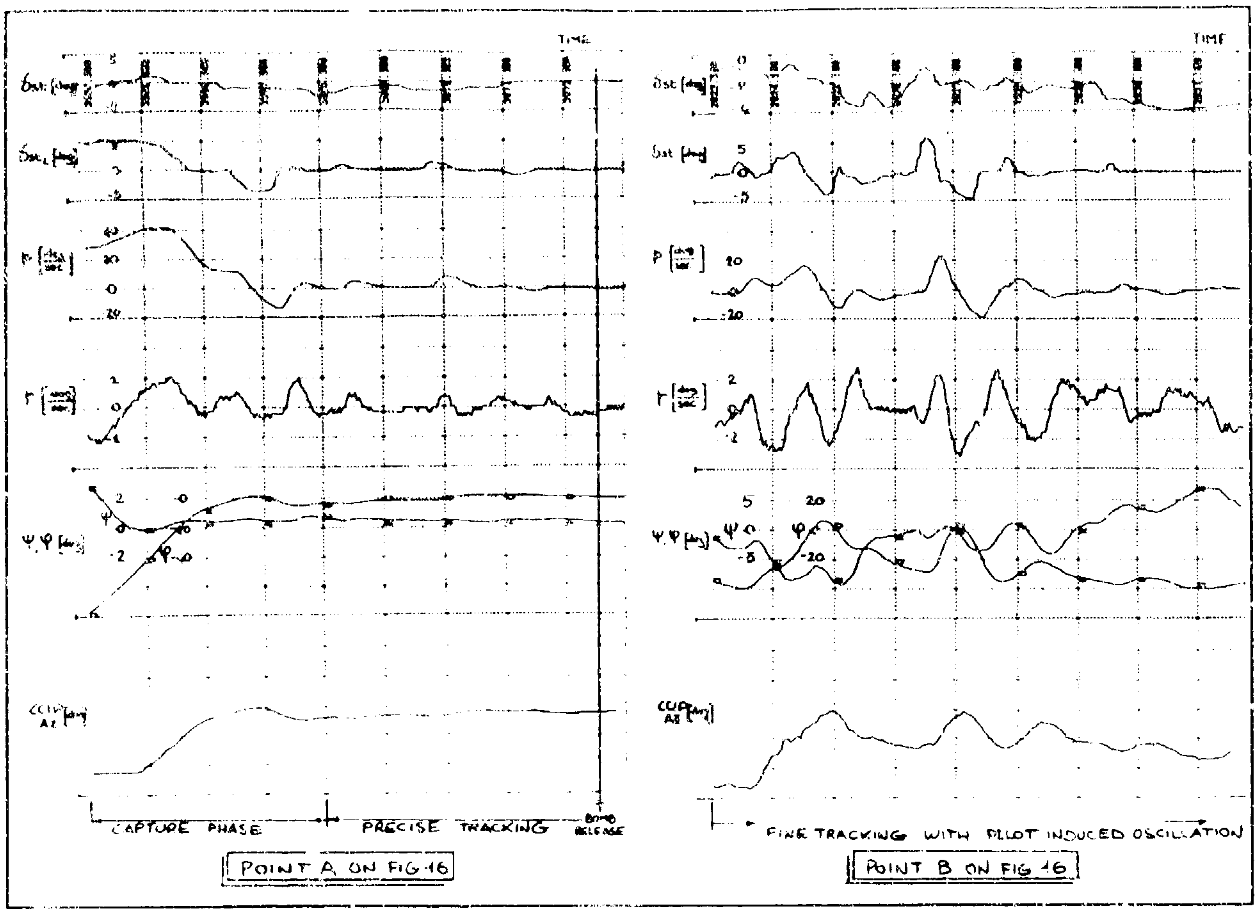
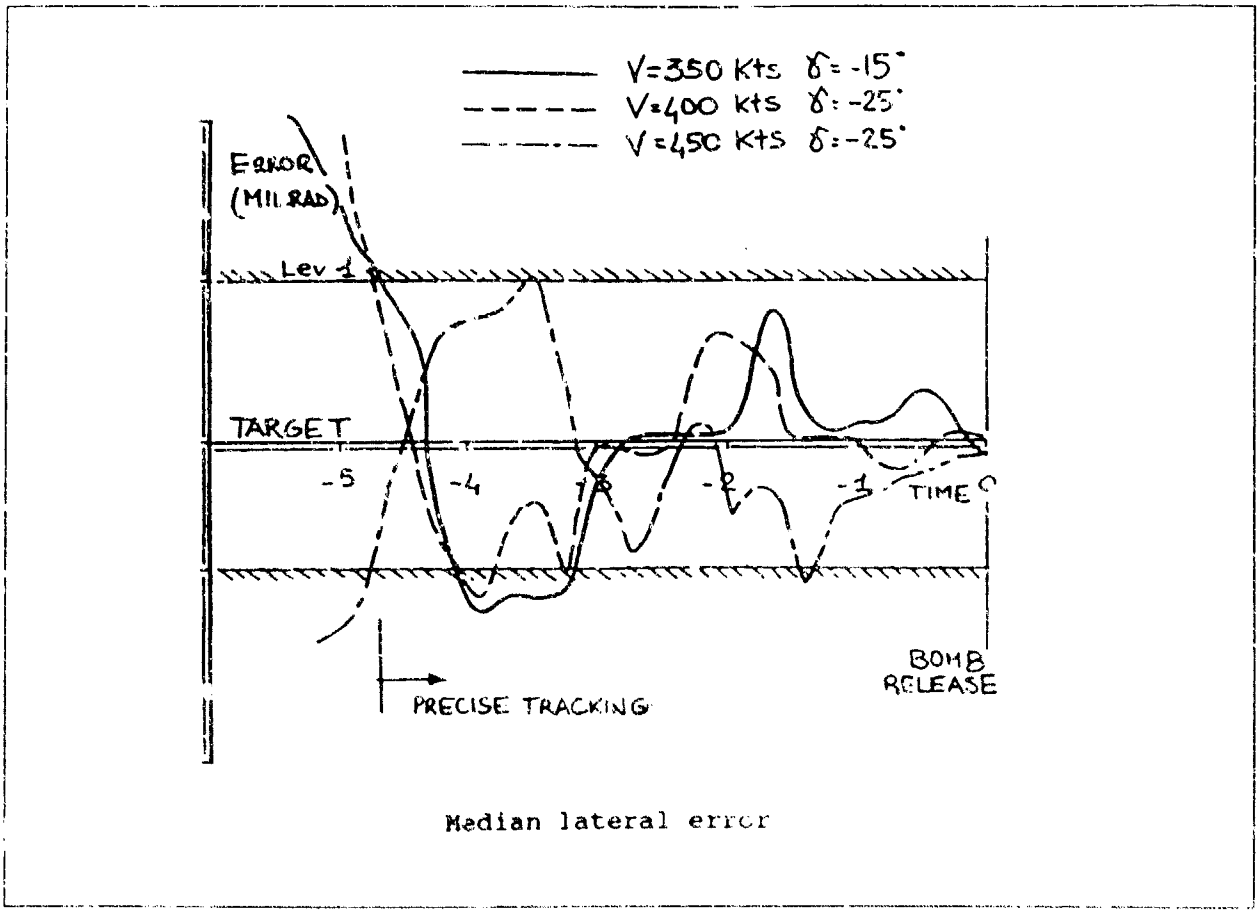


Figure 17 AMX - IN-FLIGHT TEST OF DIFFERENT TARGET TRACKING CONDITIONS



Median lateral error

Figure 18 FLIGHT TEST PRECISE TARGET TRACKING RESULTS

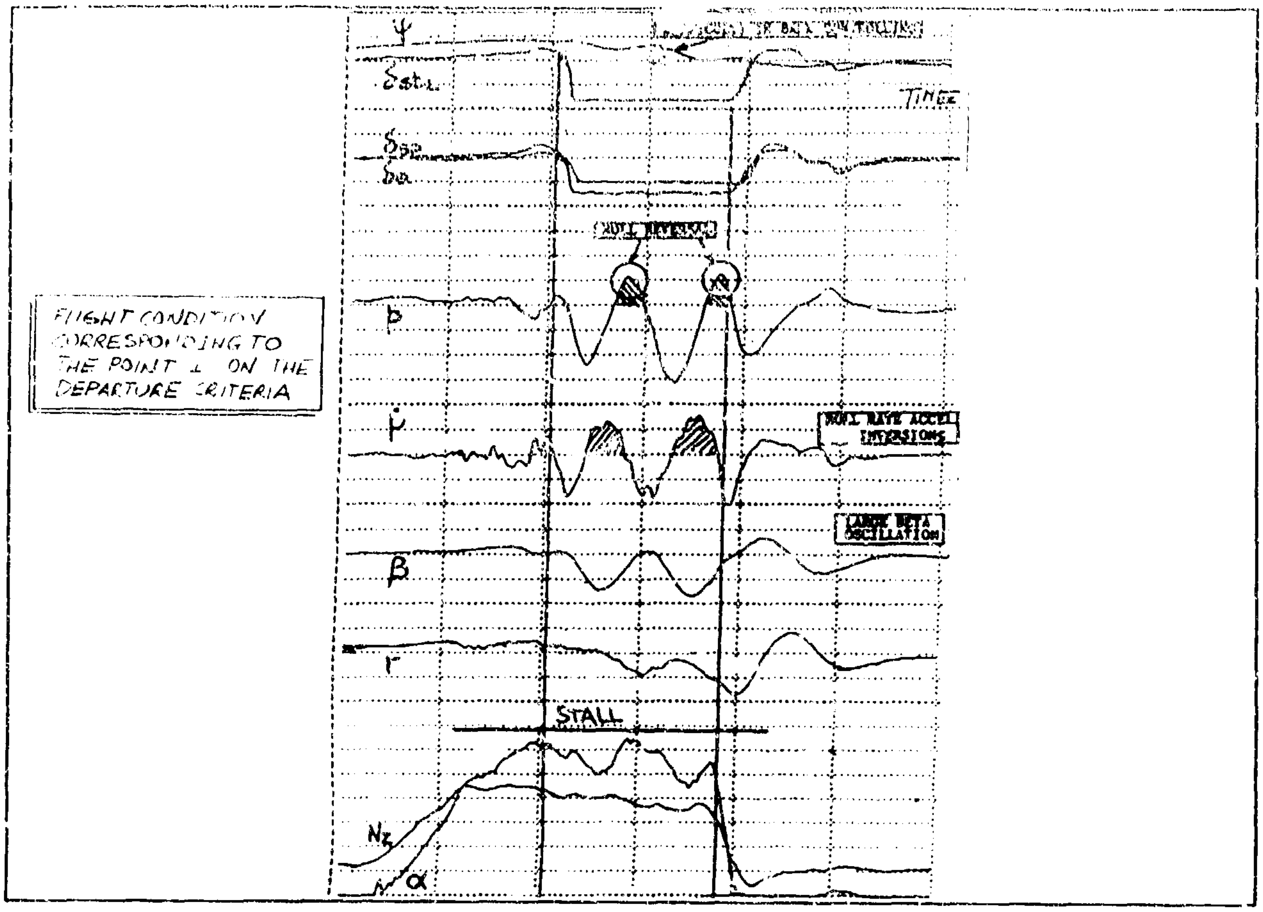


Figure 19 AMX - HIGH INCIDENCE ROLL MANOEUVRE WITHOUT ROLL TO YAW CROSSFEED

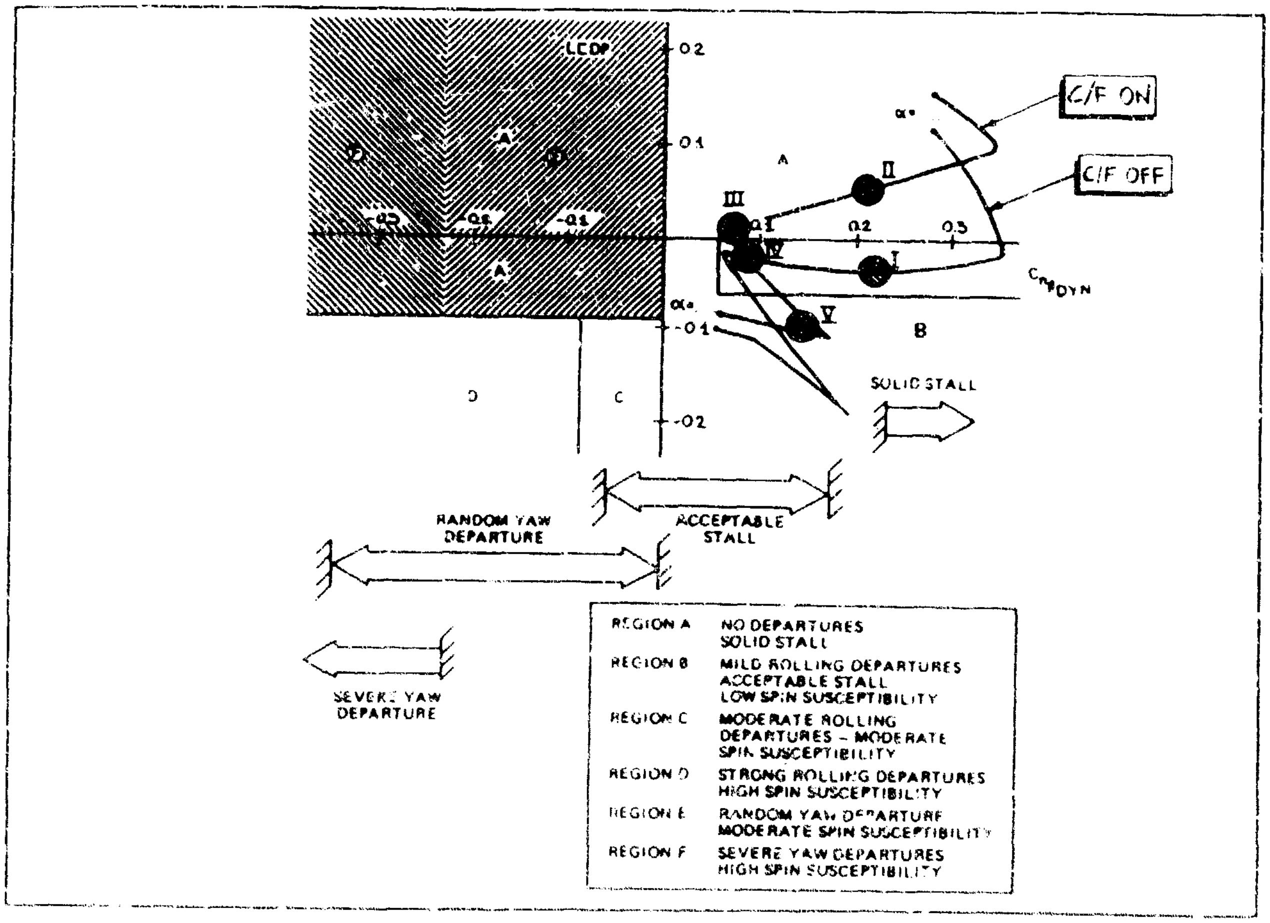


Figure 20 AMX - DEPARTURE PREDICTIONS WITH NORTHROP-WEISSMAN CRITERIA

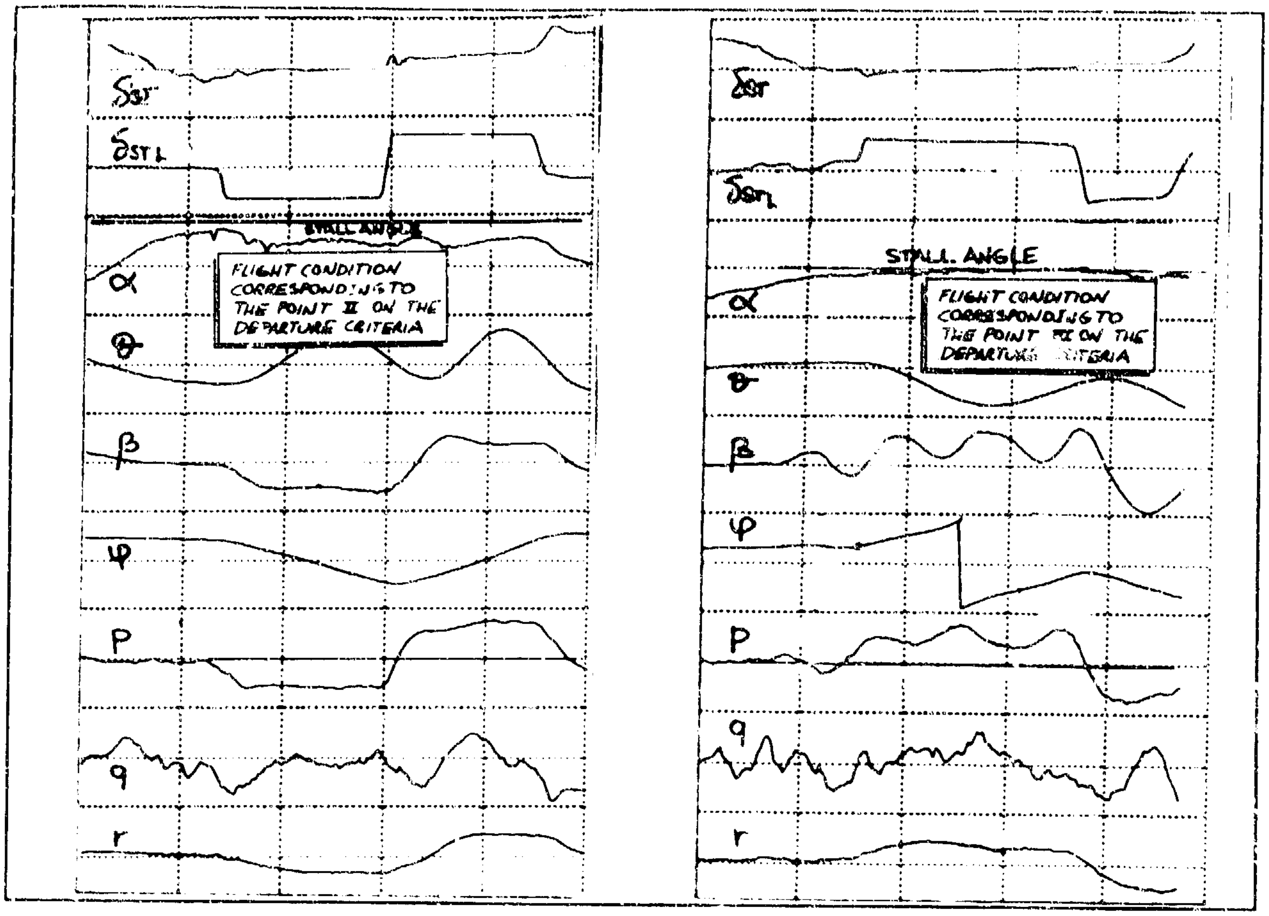


Figure 21 AMX - ROLL MANOEUVRES CLOSE TO THE STALL ANGLE WITH A ROLL-TO-YAW CROSSFEED

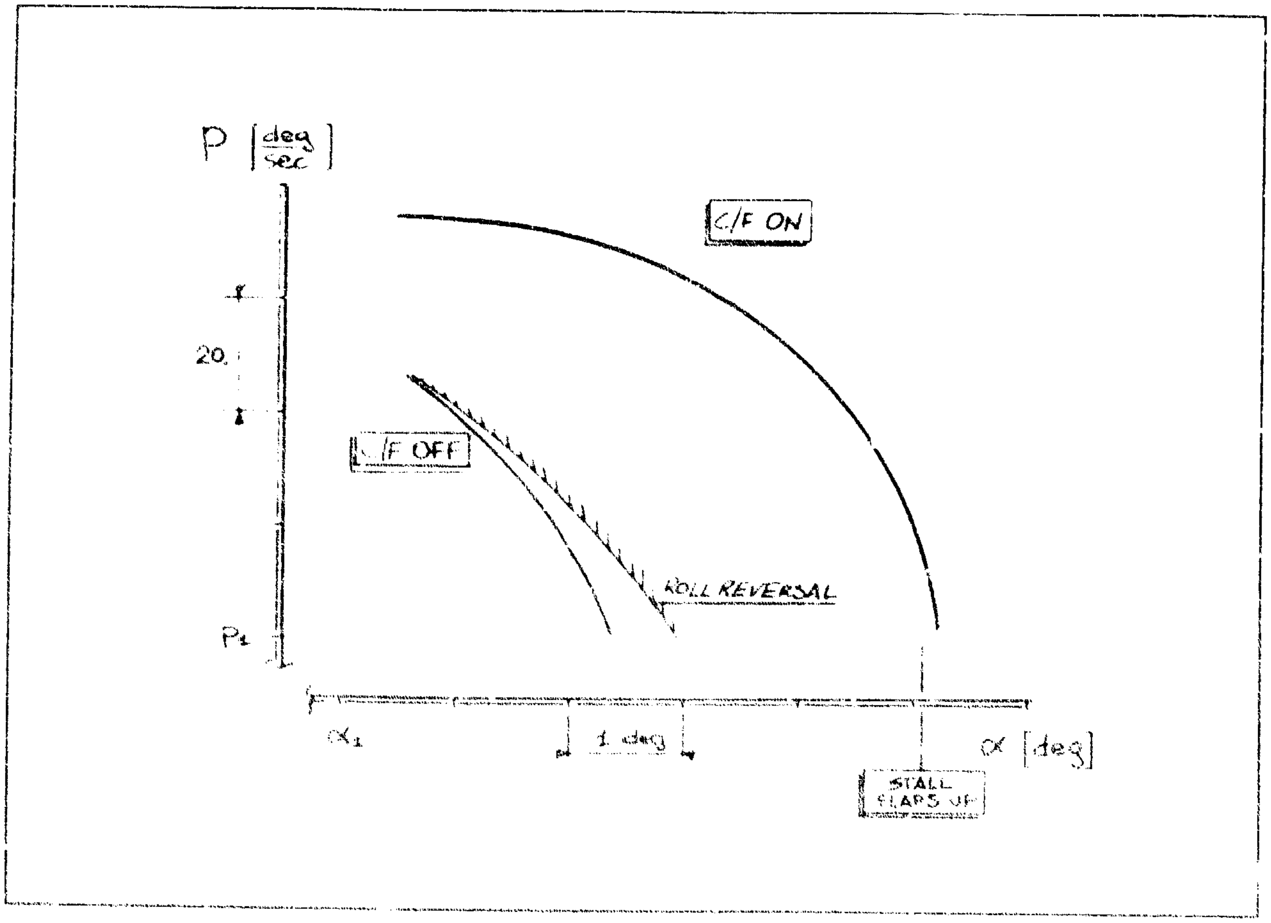


Figure 22 AMX - ROLL RATE IMPROVEMENTS AT HIGH INCIDENCE

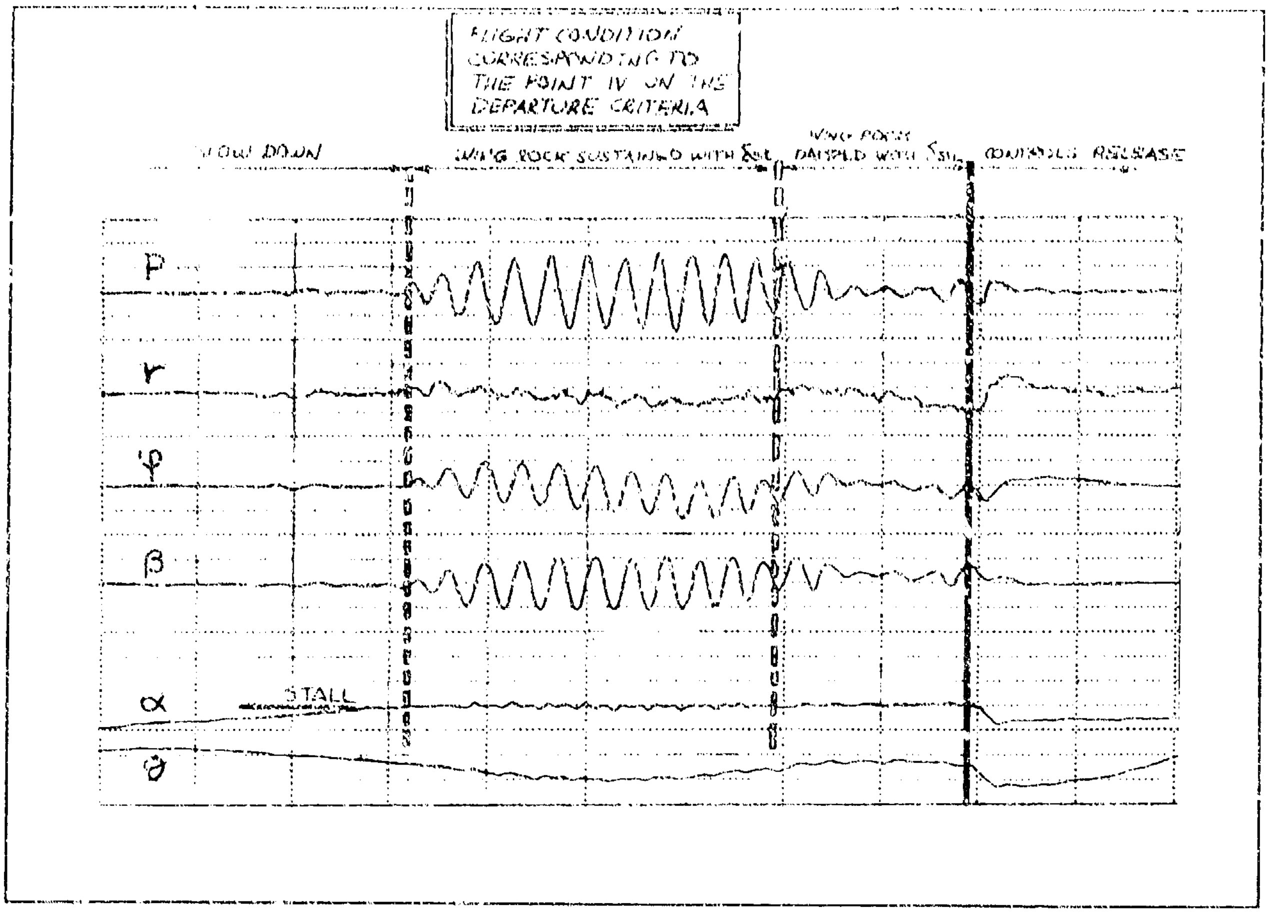


Figure 23 AMX - CLASSICAL STALL WITH A BOUNDED WING ROCK

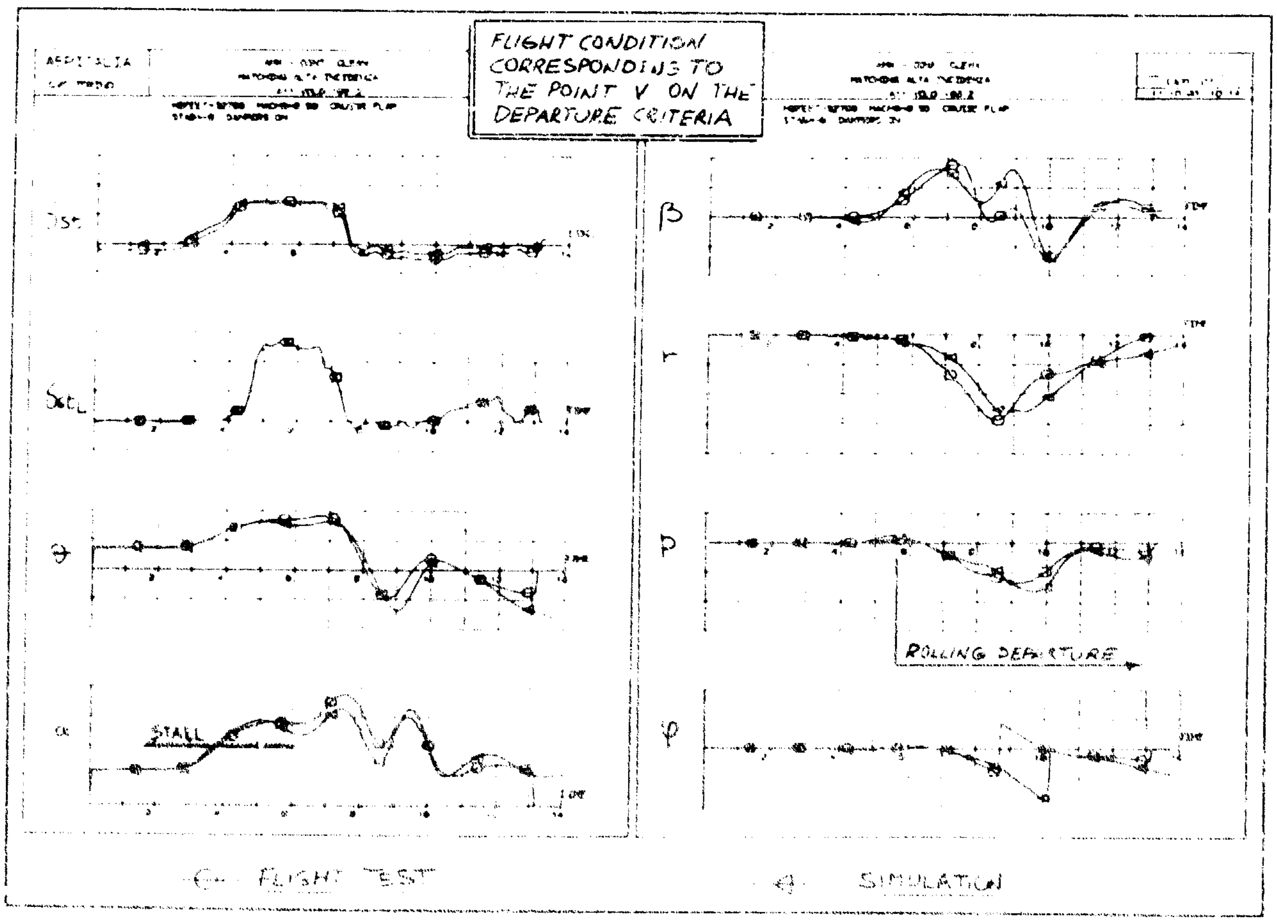


Figure 24 AMX - CLASSICAL ROLLING DEPARTURE AFTER A STALL AGGRAVATED BY LATERAL CONTROL

THE DEVELOPMENT OF ALTERNATE CRITERIA FOR FBW HANDLING QUALITIES

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SUMMARY

Provision of robust FCS and structural mode stability margins and carefree handling in highly unstable combat aircraft with a wide range of store loading requires new methods for handling qualities optimisation. The possibilities for new control modes and task-tailored handling have been greatly enhanced by modern controls. This has led to development of many alternate criteria which were tried and tested in two digital FBW research aircraft, the EAP and Jaguar FBW, and have been further developed for use in EFA. They cover the field of flight path and attitude bandwidth, tracking precision, pitch and roll acceleration dynamics and sensitivity, PIO prevention, and enhanced lateral-directional damping. Derived as design guidelines, with the facility to design for optimum rather than merely acceptable handling, more research is needed into formal boundaries for Levels 1, 2 and 3 specifications. This paper reviews the criteria and illustrates some of them by example.

1. INTRODUCTION

It is widely acknowledged that formal handling qualities criteria, developed directly from the flight experience of the early decades and formalised initially in the 1940's, are inadequate for the needs of modern control systems. While this was evident in the many handling problems of recent years, ranging from a nuisance to serious difficulties, it is also the result of the greatly increased possibilities for new control modes, and for optimised task tailored handling. The classical aircraft handling of the past was determined by the basic aerodynamics. Later the skills of the control designer were added to replicate the natural-seeming qualities of stick feel and to augment the damping to a limited extent. Simple modal parameters such as frequency and damping, although describing only the flight path behaviour, sufficed to quantify in general terms the response characteristics both to pilot inputs and to external disturbances. The handling of the superaugmented unstable combat aircraft of today, and increasingly also with future airliners, is determined almost completely by the flight control system. The resulting behavioural differences can be profound.

This has begun to be addressed in Ref.1 in the form of alternate criteria offered for consideration in the design process. Some of these and others are discussed in Refs.2 to 4, which describe how handling qualities can be identified from elements of transient and frequency responses independently of the FCS complexity. They can therefore be derived from the 8785 database or other sources of experimental data, requiring no transformation into simplified forms. The approach followed at Warton in their development has always been to establish what response characteristics were favoured by pilots and to evolve methods of description and measurement which can be directly associated with pilots' comments and their perception of task performance. This work has been greatly assisted by flight experience of two highly unstable research aircraft with advanced digital FBW, the FBW Jaguar (5) and the EAP (6). While the criteria used for design and flight clearance of the FBW Jaguar were informal, those for the EAP were comprehensive and completely replaced sources such as Ref.7 in formal clearance processes. Both demonstrated excellent handling and complete absence of PIO, and the criteria methodology has become the standard for the EFA aircraft when formal requirements seem less appropriate.

The high order of some control systems has often been blamed for poor handling. However, it is convenient to consider the "orderedness" of handling qualities to be related to the phase ranges in which they are primarily generated, rather than to the order of the FCS. The design problem then reduces to the elimination of excessive and PIO-prone high order effects arising from phase lags greater than the classical norm, and secondly to the provision within the classical response phase lag range of low order qualities optimum for given tasks. These may differ significantly from the classical forms.

2. HIGH ORDER PILOT INDUCED OSCILLATIONS

"Low order PIO" associated with pitch bobble, overcontrol due to insufficient pilot gain adaptation, improper pilot closure of altitude loops, etc. is found in classical responses and is not specifically a FBW problem. Bobweight PIO, although not low order, is also not relevant here. This discussion refers only to PIO caused by gain/phase deficiencies in FBW control law design, which can be considered to be the principal high order problem area.

High order pitch and roll PIO is quite clearly an unstable attitude loop closure, with a "synchronous" pilot acting as a simple gain on attitude error. This model simplifies the task of identifying and eliminating the attitude dynamics which provoke a PIO. While a somewhat non-linear behaviour is often seen in flight records, with the stick being switched when the attitude rate changes sign, sometimes with a stick dither while the pilot waits for the next switching point, it is unnecessary to model this. These records also suggest that stick position control is dominant and the feel dynamics should not be included in the analysis. In this loop closure model, instability occurs at the frequency where the attitude phase lag is 180 degrees or slightly more. Factors which have been empirically found to be associated with PIO tendencies are the gain, the frequency and the rate of increase of lag with frequency at this point. The latter, more simply known as the phase rate, is found to correlate strongly with PIO, as discussed in Refs.3 and 4.

A highly effective PIO metric, phase rate is readily plotted in design work and is easily found graphically from older data. Phase slope is a similar metric, given in Ref.1, which has been developed further in Ref.8 to account for the effects of response bandwidth. The typical PIO combination of low frequency and high attitude gain, marked by a high phase rate, permits large amplitude oscillations in attitude to be generated within the control power pitch acceleration limits. The frequency tends to be high when the gain and phase rate are low and therefore not PIO prone, but can also be high with poor dynamics. This may reduce the possible pitch amplitude, but it is no protection at high airspeed, where high loads are easily generated. The destruction of an F.4 at 800 knots on the third cycle of a PIO in under three seconds is witness to this.

The frequency plays a significant part in landing PIO, which is thought to grow out of the stick pumping usually exercised by pilots in the few seconds before touchdown. The Bihrle theory is that pilots excite a pitch acceleration of about +6 deg/sec² to test the response. This is done subconsciously and at the frequency where it is closely in phase with the stick input, conventionally well above the short period. The attitude lags by 180 degrees and so resembles a PIO, but this is not observed by the pilot normally because at typical frequencies for low order aircraft the attitude response is small. With increasing high order lags this frequency reduces until the attitude for the nominal acceleration is large enough to become obvious to the pilot. A ready-made PIO may then occur because the pre-existing control oscillation, harmless as a subconscious input to the aircraft pitch acceleration, essentially an open loop, becomes unstable as a pilot gain of lbs/degree in the closed attitude loop.

A PIO frequency gain of less than 0.1 degrees/lb seems to be associated with the absence of PIO, but a higher gain may be very satisfactory given good dynamics. With a low gain the amplitude limit imposed by the maximum possible stick inputs does not permit a large PIO oscillation to develop whatever the dynamics. With poor dynamics, the higher the PIO gain the more easily a PIO will be triggered. Pilots may use the full stick travel when trying to regain control regardless of the forces involved. It is especially important to assess the consequences of such inputs by calculation and in simulation, and it must not be assumed that "pilots would never fly like that".

The Calspan NT.33 has provided most of the data used to develop an understanding of the problem. Two examples from Refs.9 and 10 in Figs.1 and 2 illustrate some task related differences and the simplicity of analysis. Both have exceptionally poor attitude frequency response shapes, with a rapid increase of phase lag and relatively high gain towards and at the 180 degree lag region. The high phase rates are associated with severe PIO. Pitch acceleration is also shown, shifted by 180 degrees phase angle so as to position its frequency points directly above those of attitude. The pitch acceleration is exactly in phase with the stick when attitude lags it by 180 degrees, and its gain at this PIO frequency is of great importance in determining the nature of the PIO.

In the Fig.1 example, the pilot could not track because this always resulted in a PIO, which diverged when he tightened his gain, earning ratings of 8 and 9. The response shape invokes both closed loop droop and resonance, and despite the heavy stick force per g the uncompensated pilot attitude gain for instability is only 3 lb per degree. Because of the large lag built into the control law, pilot compensation is very difficult to achieve. With a high feel stiffness and an attitude gain of 7 degrees per inch of stick at the PIO frequency, it is not surprising that an oscillation of about +35 mils is the best the pilot could achieve instead of a satisfactory 2 mils median tracking error. The PIO trigger is the act of tracking initiated by the pilot, a conscious decision which can equally be reversed. The threshold pitch acceleration of about 6 deg/sec² is excited by inputs of only 1 lb or 0.05 inch, and as the PIO incurred inputs some six times larger the oscillation could be tied positively to the stick. The pilot was able to stop the PIO simply by abandoning the task.

In the example of Fig.2, the pilot could complete the circuit but was unable to land because of unstoppable PIO in the flare, earning a rating of 10. The record shows stick pumping as predicted, although the pitch acceleration amplitude is less than the nominal. This may be due to the combined low frequency and response gain requiring a stick force of +6 lbs for the threshold acceleration, rather high for a subconscious activity. Pilots do exhibit considerable variability, however, from slight to highly active pumping. At 10 to 15 seconds the PIO is triggered, possibly after increasing the pumping to generate a more positive "feel" for the response. The sudden reduction in amplitude indicates the start of the unsuccessful attempt to get it under control. The increased frequency when switching from pumping to PIO seems to be typical.

The landing PIO has a negligible direct effect on the flight path task in hand, and the pumping which may be the trigger is not part of the task. Large stick forces and displacements have to be applied before there is much sense of a directly commanded response because the pitch acceleration gain is so low. In one such case a pilot believed that the motion consisted of some initial turbulence response followed by a pitch up which did not immediately respond to his application of full forward stick, completely unaware that he was in a classic PIO driven entirely by his control inputs. Given that the aircraft is about to contact the ground in a very unsafe attitude and that it has apparently gone out of control, freezing the stick and overshooting is not usually seen as an option.

Although the pitch acceleration is in phase with the stick during the example PIO's, a serious deficiency is evident in the rapid increase of phase lag at higher frequencies. In the classic response, acceleration phase angles decrease from a lead at low frequencies to zero at high frequencies. This results in an immediacy of response to stick inputs, effected physically by the direct equivalence of the control surface and stick motions. This is not affected significantly by the addition of actuation lags of any reasonable quality or by the presence of feedback paths. With increasing lag in this forward path, high order handling problems may be expected as the immediacy of response to the stick is lost. The LAHOS results showed that a single 0.25 second lag stick filter added to a low order FCS could be catastrophic although in another configuration a 0.5 second lag altered the pilot comments but not the rating. With the proper attention given to the forward path design, both the FBW Jaguar and the EAP with typically 36th order pitch FCS have demonstrated excellent handling free of any PIO tendencies.

Hence the common attribution of high order problems to the complexity of the FCS design is misplaced. In simple terms, the absence or otherwise of such problems can be qualitatively identified by the extent to which a stick signal is adulterated by lags in the forward path to the control surface. It is never necessary to accept their presence as a consequence of feedback design as they can always be circumvented. A quantitative measure of acceptability is the pitch acceleration peak rise time, which should never exceed about 0.25 seconds or ideally much less. Because of their nature, handling difficulties from this source rapidly turn into closed loop problems which should be analysed in the frequency plane as discussed above.

3. TASK RELATED HANDLING QUALITIES

The tasks performed by an aircraft require it either to be directed along a given flight path or to be pointed in a given direction, usually at different times but sometimes both together. For most of the flight time the primary task is control of the flight path, but in a general sense the pilot effects manoeuvres by control of attitude. This may dominate the perception of the general handling as well as being crucial for precision pointing. The measure of the short term pitch response is usually the manoeuvre margin, with the related short period frequency and damping which essentially quantify the dynamics of the angle of attack mode. This determines only how rapidly the flight path angle rate commences, with a magnitude proportional to the g and the inverse of true speed. The pitch rate is equal first to the transient angle of attack rate and then to the steady flight path angle rate, in proportions fixed by frequency, damping, wing loading, lift slope and true speed. No formal specification exists to define this complex attitude response.

At the end of the 1940's, the handling of some 60 types had been analysed by NACA and the data formed the basis for new military handling quality specifications, which included the short period response among many other equally important metrics. As discussed in Ref.4, such aircraft, generally with modest wing loadings and high lift slopes, did not require large changes in angle of attack for manoeuvre. The corresponding attitude transients were moderate and good tracking performance was often achieved, as seen in many WW2 gun camera films. Later types were faster, heavier, eventually supersonic, flew to much greater altitudes, and lacked natural feel. Because of the degraded handling they required stability augmentation. Research to define satisfactory qualities led to the still current specification of short period frequency limits related to the ratio of g to angle of attack. This reflects the natural behaviour of aircraft with constant manoeuvre margin. The frequency is related to equivalent airspeed and the pitching moment to trim a given g increment is constant. This results in the fixed upper and lower CAP limits of initial pitching acceleration per g.

The specification therefore includes a partial element of attitude sluggishness or oversensitivity, but only in the first instant of the angle of attack response. The subsequent attitude time history is undefined. Refs.2 and 3 show how this can be precisely defined by the parameters of attitude dropback or overshoot and the pitch rate overshoot characteristics. These can vary widely for fixed points within the standard specification, which gives no design guidance for precision attitude control tasks. The pilot ratings in past research, often inconsistent when related to short period metrics alone, frequently become well correlated when attitude parameters are accounted for. Any one of the many formats in which handling data have been presented in past literature can be augmented to include such effects, which usually illuminate the reasons for the choice of handling boundaries in terms which identify the physical response seen or felt by the pilot.

The freedom to obtain desired handling qualities by feedback adjustments may be severely constrained in a highly unstable and flexible aircraft by the difficulty in providing adequate stability margins in both flight response and structural modes. For any level of stability, a powerful and flexible technique is command filtering. By this means it is possible to perform all or most of the compensation otherwise left to the pilot, who is then able to act as a simple gain controller and to devote full attention to developing a small time delay for compensatory tracking. The compensated aircraft response qualities can be related to standard low order characteristics after the adverse high order effects discussed above are eliminated separately.

3.1 Attitude Tracking

For pure compensatory target tracking, in which only the error is displayed, the optimum attitude model K/S is well known. In real tracking a mixture of pursuit and compensatory tracking will be employed, but even so the best results will be obtained with a response closely resembling K/S. This is often described verbally as "The nose follows the stick", associated with nominally zero attitude dropback, though small values of dropback or overshoot can be acceptable. The pitch rate overshoot needed in the initial transient to achieve the nominal K/S response is not to be confused with low damping. However, too much will produce excessive attitude dropback, which will also be step-like if the transient is very rapid. There will be a confusing mixture of apparent attitude overshoot followed by dropback if it is slow to settle. With a fast response, zero pitch rate overshoot with a corresponding attitude overshoot can give satisfactory tracking.

Typical problems are illustrated in Fig.3 by examples from Refs.11. It had been intended to find out if stick feel changes could degrade the handling of a case on the Level 1 upper frequency limit to Level 2 in precision tracking tasks. However, Level 1 ratings could not be achieved at all, even after frequency reduction. The transient attitude response can be derived from the basic modal parameters given in such reports. A step control input is assumed which is removed at the instant when a target attitude is reached. This shows that the large attitude dropback and pitch rate overshoot for both frequencies ensured that the exercise could not succeed.

In Ref.12 the superaugmented FCS was configured to produce a close low order match of a selected moderate frequency and CAP. General flight tasks received Level 1 ratings, but fine tracking was rated Level 2 because of excessive pitch bobble. This was shown in the attitude response for the nominal 20,000 ft/0.7M case by the large 0.4 second dropback and significant pitch rate overshoot. The rating could not be improved by any acceptable variation in frequency or damping. Level 1 ratings were achieved by adding a lag-lead stick path filter to replace the numerator zero by an effective $1/T\theta_2$ value which reduced the dropback to 0.17 seconds. This leaves the attitude response in the conventional low order form but the flight path response contains the additional lag-lead. For other designs a non-classical form is is equally possible in the attitude response.

The effects of attitude bandwidth can be observed in the step response initial transient, eg the time at the peak pitch rate and the time to settle to steady rate. It is better to measure this closed loop parameter in the frequency response. The definition here is that of the Bandwidth Criterion in Ref.1, the frequency where the attitude lag is 135 degrees. This loosely resembling the 45 degree phase margin point, and it can be considered as the effective boundary between the low frequency dynamics used by the pilot for task performance and the higher frequency region whose effects are an intrusive nuisance if not adequately suppressed. Refs 2 and 3 used the 120 degree lag frequency for this.

Fig.4 shows three typical response shapes for a hypothetical configuration used in a tracking simulation. Shape A has a rapid short period flight path response giving good attitude bandwidth, but it has a large dropback of [illegible]6 seconds. Shape B achieves zero dropback by reduced frequency and a stick filter to modify the apparent $1/T\theta_2$ with a greatly reduced bandwidth. Shape C retains the original frequency and achieves zero dropback entirely by a stick lag-lead filter, with only slight bandwidth reduction. In the task of repeated target re-acquisition and fine tracking, the relative tracking efficiencies of A, B and C were 1.0, 1.035 and 1.3 respectively. A relative value of 1.7 was achieved by a different configuration with smaller $T\theta_2$ and even higher bandwidth, with near zero dropback.

Such results show that for excellent fine tracking performance, high bandwidth is not sufficient and near zero dropback is necessary but not sufficient. The best results are obtained with high bandwidth and near zero dropback combined, which will generally require a non-classical response. Specification of a maximum bandwidth seems unnecessary. The maxima in the Bandwidth Criterion and the Northrop Criterion are due to the classical response which results in excessive attitude dropback. The ideal but unrealisable K/S has infinite bandwidth with zero dropback. As shown in Ref.4, the natural bandwidth variation is proportional to airspeed, and this should be reflected in specification of a minimum in the same way as the short period frequency.

<u>3.2 Flight Path Tasks</u>

$T\theta_2$ is often referred to as the flight path lag, arising from the path to attitude relationship. It does not appear in the path to control input transfer function and is more properly known as attitude lead. Quickened velocity vector and climb-dive HUD information allow the pilot to control flight path directly. While the attitude characteristics have an important part to play in the predictability of the response by giving the pilot phase advanced cues to the future flight path, control of the path as the primary task is effected through the flight path angle time delay determined by the short period frequency and damping (Ref.2). It increases inversely with speed and represents a constant distance at a given altitude. In closed loop flight path control, the path angle bandwidth, defined as the 135 degree phase lag frequency, is of significance.

Air-to-air combat requires an agile path response but does not depend on following a uniquely prescribed path. The path delay distance can be as little as 50 metres or less for the highest possible agility, where only the problem of pilot incapacity through rate of g onset might set a minimum limit. A curiosity of the non-classical response filtered to zero dropback for optimum pitch tracking is that, regardless of how high the attitude bandwidth is made, the path time delay is a constant $T\theta_2$ seconds and the path bandwidth approaches but can never exceed $1/T\theta_2$ rad/sec. The effect is always a slower g response, termed "g creep" in Ref.12, but which can still give satisfactory target acquisition especially at low altitudes. This may not be so at high wing loadings or at higher altitudes because of increased $T\theta_2$. The use of direct lift can enhance the path response but few configurations can generate enough to match more than a small proportion of the lift due to pitch rotation. The design flexibility inherent in stick command shaping techniques permits optimisation of either the flight path or attitude response, or of a fixed compromise between them, or of an amplitude dependent response graduated from precision attitude to rapid flight path.

In terrain following a prescribed path is computed but it does not require great agility. Pulling up well in advance of an obstacle for minimum clearance in level flight is not much affected by flight path delay. Pilot confidence is enhanced by smooth, precise attitude handling when flying at extremely low altitudes, even for heavily loaded types where this could give a one second or 300 metre path delay. In the experiment of Ref.13 the terrain following display was quite insensitive to short period dynamics. The satisfactory boundary, which could be interpreted as a 0.6 second or 200 metre path delay limit, seems to have been selected by routine manoeuvres and was probably influenced by the sluggish attitude overshoot, Ref.3.

Flight refuelling is a demanding task, requiring exact alignment of the flight path with the mean path of the drogue or flying-boom tanker. However, the tests in Ref.9 showed very clearly that although the absence of high order attitude control problems is essential, excellent attitude tracking characteristics are not required. The best configurations had small path delays of about 0.3 seconds (equivalent to 50 metres distance) or less, with a large attitude bobble which was not too intrusive because of the very small control inputs used. Cases 1B and 2D achieved a pilot rating of 1 with dropback and pitch rate overshoot values of about 0.5/2.0. Case 2A was rated 2.5 but tended to unpleasant oversensitivity with values of 0.73/3.4. The attitude tracking ratings were 4/5/6/7 (1B), 2/2.5/3 (2D), and 3/4 (2A). The NT.33 with its quite small 0.8 second value of $T\theta_2$ was not fully representative of modern higher wing loading configurations at refuelling altitudes, where for similar path delays the attitude transients are larger. The data from Ref.14 indicate that the path delay should not exceed about 0.7 seconds. If a flight refuel FCS mode is required, it should probably respect such values for both path and attitude, as far as is possible. The data available from more recent types are insufficient to generate a formal criterion.

Performance of the approach and landing task depends on control of the flight path, but this is a function of many parameters such as front- or back-side drag, thrust response, phugoid and speed stability, time and distance available, ground effect, etc, as well as the basic short term response which underpins all the others. The most significant open loop parameter is the flight path delay, which should not exceed about 1.5 seconds generally or 1.0 second for precision path control. Attitude tracking performance is not such a strong factor, although as in flight refuelling there are limits to the attitude parameters for satisfactory predictability. Dropback and pitch rate overshoot ratio greater than about 1.0 second and 2.5 respectively are likely to appear too aggressive. Although zero or positive dropback will be usual when the path delay is short, Level 1 has been obtained with attitude overshoot up to about 0.3 seconds, in one example with pilot ratings of 1 or 2 with excellent control of the nose in the flare. Some of the flight path bandwidth research data, most of which relates to this task, is discussed below.

4.0 FURTHER ASPECTS OF BANDWIDTH

Height error tracking involves the double integration K/S^2 plus the short period lag. Improperly used in single loop control, it can lead to severe PIO in flight refuelling, in formation or in flight very close to the ground. The path angle involves a K/S-like integration plus the short period lag. This produces the characteristic step time response of a ramping path angle following an apparent time delay inversely proportional to speed. The constant distance this approximates could be considered as a constant bandwidth in the flight path profile, as noted in Ref.3, but as speed decreases the path delay time increases excessively. This becomes a major factor in predictability which sets a limit on the response time.

The landing flare is a task where open loop control is typically employed, for which time domain criteria are appropriate. A precision or instrument approach requires closed loop path control, in which the frequency bandwidth is significant. The relationship between this and path delay is shown in Fig.5, though this is valid only for the classical second order short period. With increasing natural frequency, the bandwidth tends towards the inverse of the path time delay. In a well damped response, typical of highly augmented systems, a path time delay limit of 1.0 second for precision approaches is closely equivalent to the minimum bandwidth of 0.8 rad/sec given in Ref.14. Similarly for more routine approach tasks a path delay limit of 1.5 seconds is nearly equivalent to the minimum bandwidth of 0.6 rad/sec given in Ref.16. A 0.7 second path delay limit for flight refuelling is equivalent to a minimum bandwidth of 1.15 rad/sec.

These limits exclude a substantial portion of the Cat.C Level 1 envelope. The need for a wider exclusion is highlighted by many examples from the low frequency/low damping area. One is the carrier approach Case K with 1 rad/sec and 0.79 damping ratio shown in Fig.9 of Ref.3 (from Eney, Navy NF-8D), where the rating was an average 7.3 and was 8 for the GCA task. This case comfortably meets the usual Cat.C criteria, though the flight path bandwidth is slightly less than the suggested optimum at 0.7 rad/sec. However, the attitude time response is very sluggish with the pitch rate peak occurring after 2 seconds and reaching steady conditions only after 8 seconds. The attitude response does not affect the functional performance of a flight path control task directly, but it can influence the pilot's perception of it to modify the achieved performance.

The pilot derives a cue to the future flight path from the initial attitude transient, which is mostly the angle of attack increment. The measure of this increment is $T\theta_2$, the angle of attack per unit steady pitch rate. It provides the attitude phase lead relative to flight path angle, and determines the attitude dropback or overshoot, the pitch rate overshoot ratio, and the time to the first pitch rate peak for a given frequency and damping. The cue will be unsatisfactory if it is too small, sluggish, large or rapid. Time response and frequency bandwidth criteria which reflect this are discussed in Refs.2 and 3.

Attitude lead is not always necessary, and there are many examples which show that flight path can be directly controlled. Given a high enough natural frequency, a pure direct lift mode would provide a very satisfactory attitude response even at constant angle of attack. For example, at 8 rad/sec and a damping ratio of 1.0, both bandwidths would be 3.3 rad/sec, and the flight path time delay and the attitude overshoot would be 0.25 seconds, settling to steady rates in less than a second. Many powered aircraft from Cessnas to jets can use thrust for short term control of flight path, as well as in the long term which all aircraft must do. V/STOL aircraft such as the Harrier can only use direct lift for powered approaches, usually at constant attitude. Even in the simple sailplane, pilots use the airbrakes for accurate high bandwidth control of flight path directly through drag modulation, at constant speed and without explicit pitch control except for the flare, estimating the path angle deviations from relative movements of the aiming point.

Many HUDs show velocity vector or climb-dive angle quickened to eliminate the usual lag, effectively giving a bandwidth similar to attitude and enabling the pilot to control the path with great ease and accuracy by simply flying the symbol onto the desired aiming point. This is not always available, of course, and there are many situations where the pilot does not have specific flight path guidance. For good handling in the general case, it is recommended that the bandwidth guidelines should always be followed for aerodynamic lift configurations. An upper limit on bandwidth, or lower limit on path delay, will result from excessively abrupt and large attitude time response dropback, as noted in 3.2, but it is not possible to define unique limits. However, attitude time responses can be adjusted by stick filtering to maintain constant values of path delay and dropback while increasing the path and attitude frequency bandwidths substantially, impossible with classical response relationships. Unlike optimisation for pitch tracking, the optimum path time delay is usually much less than $T\theta_2$ and the path bandwidth is not constrained to $1/T\theta_2$ rad/sec.

While improper superaugmentation design can lead to a K/S^2-like path response with poor control qualities (Ref.14), a similar effect is achieved by low bandwidth in a classical response (Ref.4). Path bandwidth is determined by the combined effect of the short period frequency and damping. These are specified separately in Ref.1 and can be transformed into path delay boundaries as shown in Fig.6. The limits suggested above are superimposed and show the large areas of permitted but inadequate response in the Ref.1 requirements. For the reasons found in Ref.11, the lower boundary is unlikely to give good handling due to severe attitude bobble at all speeds. The upper boundary may not be satisfactory at higher speeds for some task requirements. Additional "best response" boundaries can be inserted as required. Used with other criteria discussed above, this format enables highly augmented systems to be assessed against the Ref.1 requirements without potentially invalid identification of frequency and damping.

Although flight path control is a function of the g response, for the landing approach the optimum control sensitivity is not closely related to stick force per g. Refs.2 and 13 indicate a correlation with attitude gain at the bandwidth frequency. Pilots were found to select a high attitude gain for a low bandwidth decreasing to a low gain as bandwidth increased. This seems to relate to their desire to overdrive a sluggish response and to constrain an abrupt one. However, pitch acceleration gain sensitivity has long been known to be significant though no criterion has become widely used. The influence of this gain on the nature and violence of PIO has been considered above. Ref.4 notes the fact that the LAHOS results indicate a correlation of high frequency acceleration gain with the attitude bandwidth as good as that of the attitude gain. These results fair smoothly in to the lower bandwidth end of a similar correlation from Ref.9 data. It can be observed that stick force per g should be proportional to CAP in order to maintain satisfactory pitch acceleration sensitivity in up-and-away flight.

5.0 MANOEUVRE DEMAND EFFECTS

Ways in which the manoeuvre demand structure affects handling and therefore task performance are discussed in Ref.3. Pitch instability can be corrected by angle of attack, g or attitude feedback, the latter being implemented by integral pitch rate in practice. The dynamic response to control input and the steady manoeuvre of these systems can be made identical by command filtering and scheduling. The superior attitude disturbance rejection of integral pitch rate systems in turbulence is greatly favoured by pilots. If the heave response must be reduced for pilot comfort or weapon delivery accuracy, direct lift gust alleviation added to a pitch rate demand system is optimum. For accurate control of manoeuvre limits, integral angle of attack or g demand is most suitable, and the attitude stability and self trimming of integral pitch rate demand is ideal in level, climbing or diving flight. The best attributes of all three can be blended into an optimum system.

While the performance of most tasks is predominantly influenced by short period fixed speed handling, the three manoeuvre demand concepts differ in their long term responses. This is most significant in low speed flight where the speed, flight path and static stabilities have important effects. Many criteria have been proposed for the landing flare, though there is no final consensus so far. The take off and landing rotation, inhibition of integrators on the ground, ratio of static to manouevre stability, phugoid effects, etc., must be carefully managed to avoid unexpected handling problems. The integral pitch rate system has proved to be highly compatible with Cat. C task elements, with excellent pilot ratings when enhanced by short period path response shaping and speed feedback for static stability.

6. LATERAL-DIRECTIONAL HANDLING

Highly augmented aircraft with manoeuvre limiting or carefree handling facilities will usually have full authority augmentation in the lateral-directional axes also. It is less common for advanced augmentation to be employed in these axes, as instability is unusual except for limiting conditions. These include roll damping reversal beyond the stall, or directional instability beyond or near the stall or at very high Mach number.

The directional axis augmentation will usually take a traditional form. In the lateral axis conventional roll rate damping is coupled with enhanced command authority augmentation. An internal loop is seldom used as completely adequate control of the roll performance is obtainable without it for most configurations. There has been little fundamental need for high order handling metrics, reflected in the relative scarcity of research data. However, the lateral handling qualities of some modern aircraft have given rise to difficulties ranging from slight to serious. The causes and solutions are similar to those of the pitch axis, and metrics to describe the handling by similar methods can be derived by considering how the pilot perceives the handling responses.

6.1. Roll Attitude PIO

The standard low order PIO hypothesis is that of closed loop attitude instability due to the roll numerator and the dutch roll mode when the latter is poorly damped. It is not certain how much support is available in flight records for this hypothesis. Many have shown the lateral control loop to be closed around the heading angle. This is consistent with pilots' greater aversion to disturbances in yaw than in roll, and a preference for correcting heading errors by lateral rather than by directional control. In either case the same mechanism is involved and the same solution is needed to match the numerator and dutch roll frequency and damping. It is not a problem for highly augmented aircraft except for remote failure states.

The pure roll attitude PIO which has occurred in some modern types is identical in principle to the high order pitch attitude PIO discussed above. The pilot can immediately synchronise with the oscillation, fixing it at the frequency where the attitude and lateral control input are 180 degrees out of phase. This is possible because of additional lag and increased gain at this crossover frequency introduced into the response by the control laws. Such PIO was not experienced in past aircraft because the attitude lag did not exceed 180 degrees, the direct path to the control surfaces giving unlagged roll acceleration at high frequencies. Routinely achievable actuator lags are not normally sufficient to cause closed loop handling problems, especially when coupled with non-linear command to alleviate overcontrol tendencies at low or zero roll rate in high roll performance aircraft.

Fig.7 shows some typical basic Nichol's plots of pure or modified roll mode time responses. The nominal input amplitude is the full stick input giving a roll rate of 75 degrees per second, representing a landing approach case or the initial slope of a non-linear command in a high speed case. Linear equations normally used only for small amplitude can be used here, even though inaccurate for actual full stick inputs, because such plots are an excellent guide to the relative sensitivity at small inputs where closed loop control problems are usually initiated.

Response A has a relatively short augmented roll mode of 0.25 seconds. Even after some typical actuator lag is added the frequency at 180 degree lag is high with low gain, ensuring freedom from closed loop problems. Smaller values than this can easily be achieved by augmentation. A stick lag filter has been used in some command augmentation systems for response attenuation. Response B represents this by adding a 0.3 second stick lag to A. The response at given frequencies is clearly attenuated, but at the critical case for closed loop stability the gain is several times greater and the frequency is reduced, well into the range of potential piloting problems. The alternative solution is a lag-lead filter to increase the apparent piloted roll mode constant. Response C represents this by a 0.55 second roll mode, achieving similar attenuation with much less lag penalty, and maintaining a low order response free from PIO.

A comprehensive database of roll PIO examples does not exist, but the metrics which indicate PIO sensitivity are the same as in the pitch axis. Excessive attitude gain, low frequency and high phase rate at the 180 degree lag point all indicate a problem. A response amplitude of more than ±10 degrees and lag crossover frequency less than 1 Hz should be cause for concern. Less than ±5 degrees at well over 1 Hz should be very satisfactory and it is not difficult to do much better than this. The pitch phase rate criterion seems to apply equally to the roll axis, although the order of magnitude greater accelerations and pilots' readiness to use large inputs in roll suggest that a more stringent application is sensible. Pilots do not tolerate significant roll attitude errors in the landing flare and are likely to exercise tight closed loop control in the presence of turbulence. This can act as the PIO trigger in a similar way to stick pumping in pitch, and probably most roll PIO occurs in this flight phase.

Roll ratcheting is another form of closed loop oscillation found with some augmented controls, but whether the pilot is active or passive seems to be disputed. It can occur in level flight or while rolling, and it seems generally to occur at a frequency of about 2 to 3 Hz. For aircraft with significant lateral stick displacement, it is sufficient to treat the problem as a bobweight loop. The pilot's contribution is confined to the passive effects of arm and hand on the reflected mass, stiffness and damping of the stick and connected control circuit. This can vary widely according to pilot technique. In one example pilots found they could control the presence or absence of mild ratchet by the way the stick was held, but it was readily eliminated by a software change following a bobweight analysis. In the later EAP, the bobweight loop gain margins are 30 or more and the stick is totally inert despite roll acceleration described by one pilot as "spectacular". Ratchet did not occur in earlier aircraft because of the negligible acceleration lag and the high stick free frequency and damping of the unpowered control circuits. These margins have deteriorated in more recent designs.

6.2. Roll Time Response Metrics

The control of roll attitude is mainly open loop. Except as discussed above, precise tracking of a target bank angle is usually a low bandwidth task, eg trimming to level flight for the cruise. As an inner loop in control of heading or of vertical velocity in steep turns, inputs typically take the form of pre-programmed discrete adjustments. Because there is no frequency response lead term corresponding to $T\theta_2$ in the pitch axis to compensate for the roll mode lag, a K/S-like response is not possible and can only be approached by reducing the lag. This is limited in practice by pilots' intolerance of the resulting high roll acceleration, both physical in a real aircraft and to a lesser extent visual in a fixed base simulator.

The standard metrics of roll handling qualities have long been the roll mode damping time constant τ_R, and the time to bank. Usually the roll mode has been measured as the time to reach 63% of the steady roll rate in a step response. This has been a convenient label, but it does not identify the way in which roll damping is perceived. The most meaningful visual measure of damping is the bank attitude overshoot, the extent to which the roll continues before stopping after control removal. All response characteristics of a pure roll mode are completely defined by the time constant. It is equal to the ratio of the steady roll rate to the initial roll acceleration, the ratio of roll attitude overshoot to steady roll rate, and the intercept on the time axis of the attitude. The time taken to reach a steady roll rate is effectively about $5\tau_R$. In these ways the effective roll mode constant of a highly augmented response can be identified.

When FCS lags and delays are added, distortions occur in the acceleration and rate time histories. The attitude overshoot and time axis intercept are good measures of effective roll damping for any order of system if a reasonably steady roll rate can be measured. Fig.8 shows the time responses of the basic cases A, B and C from Fig.7. Although B and C are not identical, they have the same attitude overshoot and time axis intercept, and therefore the same effective roll damping of 0.55 seconds. An effective time delay of 0.07 seconds and equivalent roll mode of about 0.52 seconds could be derived in the rate response of case B. This does not exactly produce the correct attitude response overshoot but it is very close. This is an unsatisfactory method because the small time delay fails to indicate the borderline closed loop behaviour shown very obviously in Fig.7. The acceleration time response shows a substantial lag which is clearly much more indicative of a high order problem than the effective time delay in the rate response.

6.3. Roll Bandwidth

The single degree of freedom roll response is of such a form that the bandwidth, defined as the frequency where the attitude response lag is 135 degrees, is always $1/\tau_R$. Requirements expressed in terms of the time response mode t_R define the roll bandwidth. When higher order effects are introduced, this is no longer automatic. Fig.7 shows that although cases B and C have the same effective roll mode in terms of the bank angle damping, B has a lower bandwidth than C. The difference is visible in Fig.8 as a slower initial growth of roll rate, but neither this nor the bandwidth is sufficient to indicate the closed loop control problems of case B. Its bandwidth is still higher than hat of a 1 second low order roll mode satisfying Cat.A Level 1. No research has been carried out into bandwidth effects as such, so far as is known.

6.4 Dutch Roll

The Cat.A Combat requirement for a minimum frequency of 1 rad/sec and damping of 0.4 is very easily met. High relative damping of 0.7 to 0.9 is often obtainable, which provides excellent behaviour in turbulence but may still be unsatisfactory. Ref.17 describes how gun aiming of the F-20 was degraded by a small nose slice or drift after target acquisition, attributed to the effects of the washout filter producing a residual drift in rudder command. The minimum dutch roll frequency was 2 rad/sec with relative damping of 0.5 to 0.8. After excitation of the dutch roll by lateral control, some seconds then elapsed before the sideslip settled. This was cured by adjustment of the filter and dutch roll frequency.

Even without considering the washout filter effect, the combination of low frequency and high damping gives long settling times. The common use of rudder pedals to point the nose in gun aiming is similar to tracking with a direct side force yaw pointing mode, for which the Bandwidth Criterion in Ref.1 was originated. The Level 1 minimum bandwidth of 1.25 radians per second could well be used to fix the minimum values of dutch roll frequency at higher damping levels. Both considerations require the frequency to be increased with higher damping to compensate for the increased sluggishness, and show that the standard Cat.A limits are inadequate.

7.0 FUTURE RESEARCH NEEDS

A comprehensive range of FCS design criteria for enhanced handling qualities of FBW aircraft has evolved at Warton using the principles discussed above and in the references. Their aim has been to provide the best achievable qualities rather than just acceptable-satisfactory, and by 1986 were already developed sufficiently to achieve pilot ratings in the EAP typically of 1 and 2. Work has continued subsequently to refine the criteria and to further the understanding of the relationship with task performance. Other recent FBW aircraft are showing good qualities also, and it should be expected that the difficulties of the past are largely overcome.

The increasingly high standards now available inevitably lead to changing pilot perceptions of desired handling limits. It is also the case that there may be no counterpart in conventional handling of the past to some of the characteristics widely found on FBW aircraft. An example of this is the combination of extreme attitude stability coupled with responsive control now feasible, and another is the potential to combine high bandwidth, smooth precision attitude tracking and sharp manoeuvres in a single mode. Criteria for optimum handling are not acceptable as specifications, both because it is unnecessarily restrictive and because it may be impossible to achieve under all operational conditions of flight envelope, store carriage, etc. Consequently the boundaries of Level 1 handling are in need of updating and indeed reformulation to account for the alternate criteria methodology which must surely supplant the current formal requirements.

There is a strong need to conduct new simulation in an operationally realistic environment to determine Level boundaries appropriate to FBW aircraft. With the increasingly potent CGI systems now available, this process can be initiated and developed in ground based simulators, but the final validation requires the use of a modern variable stability aircraft. It is to be hoped that such a programme will be undertaken with sufficient funding to provide an in-depth database of new handling information.

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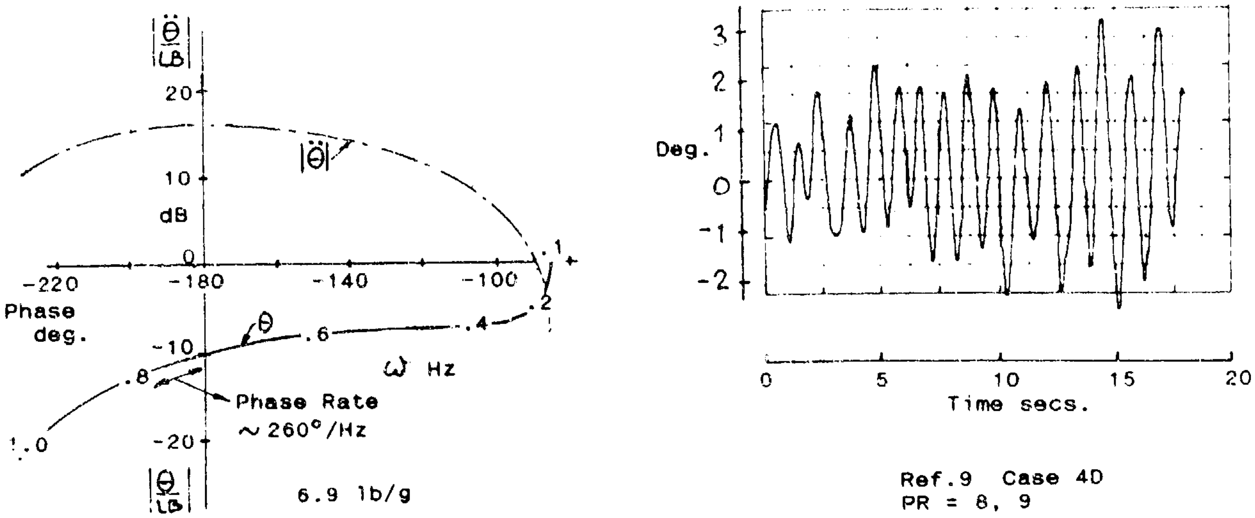


Fig.1 EXAMPLE OF TRACKING PIO

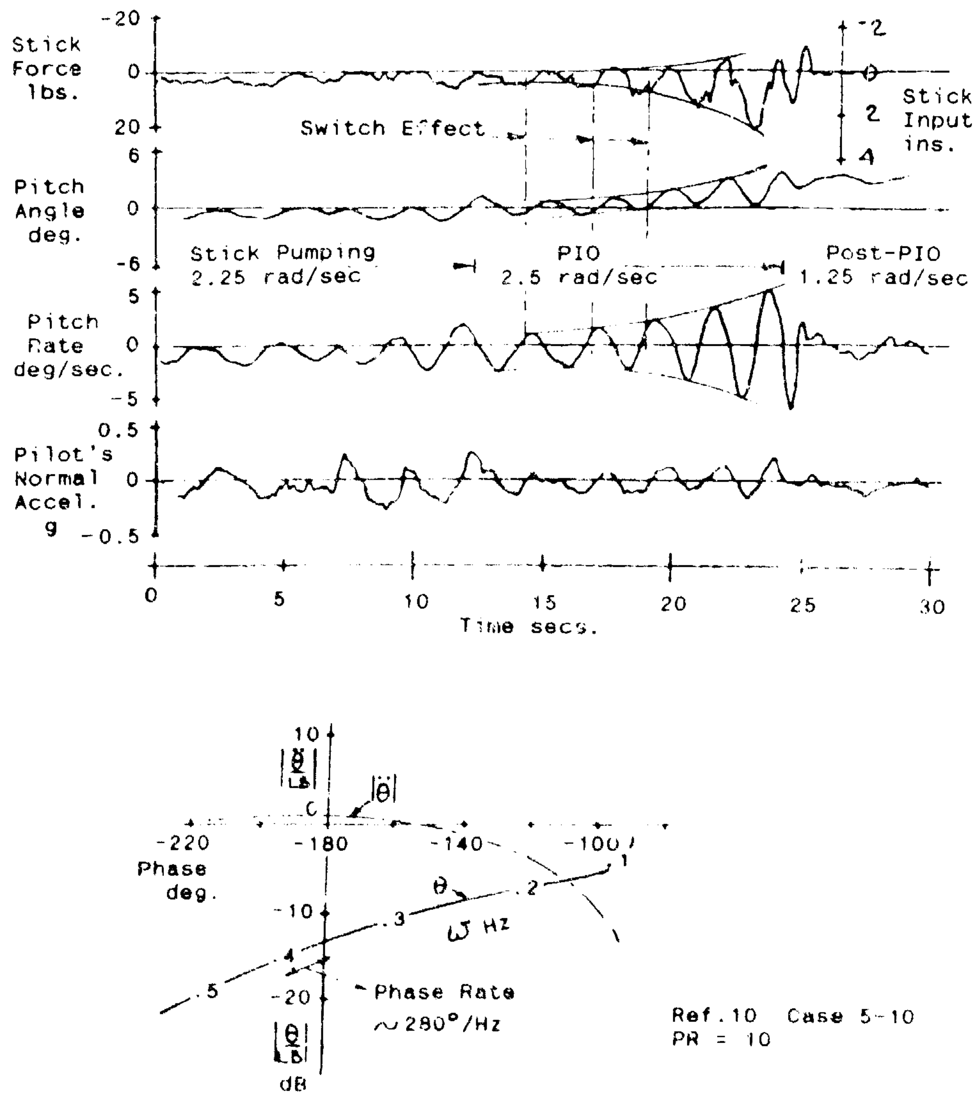


Fig.2 EXAMPLE OF LANDING PIO

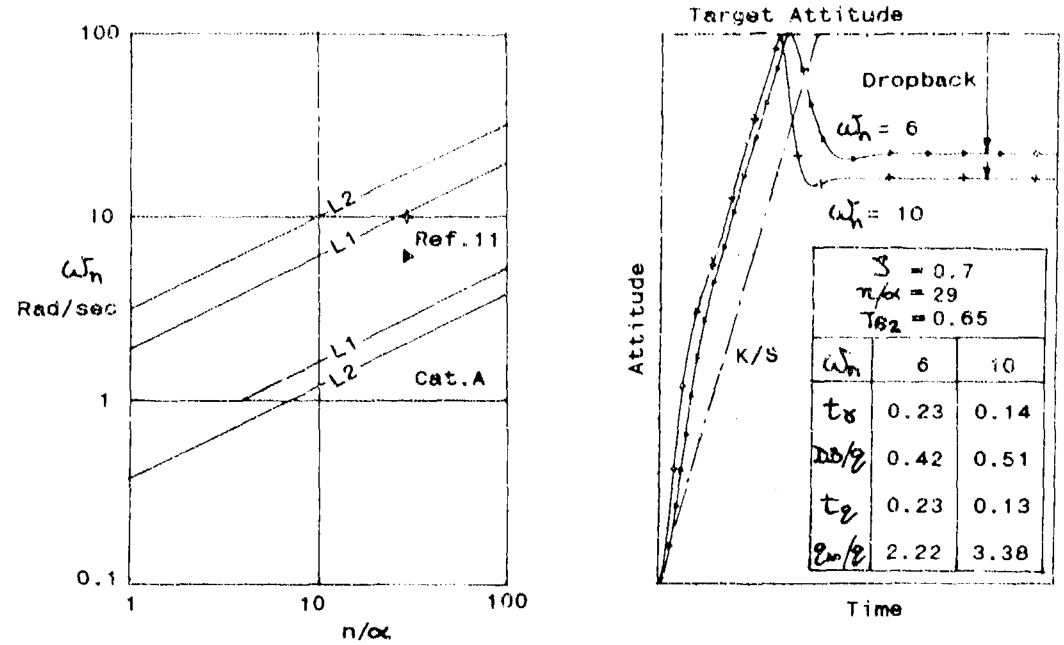
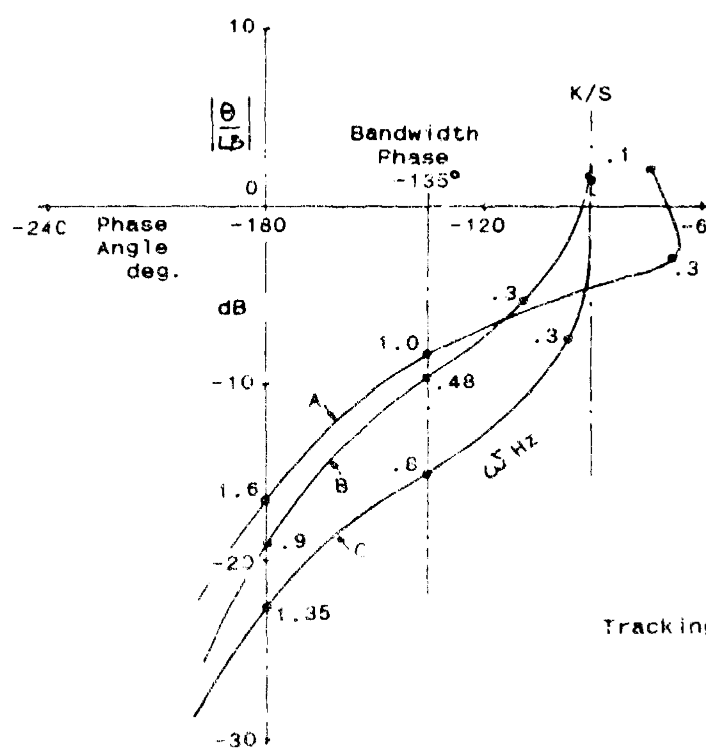


Fig.3 TRACKING PROBLEMS WITHIN LEVEL 1 LIMITS



	DB/q	qm/q	tq	tγ
A	0.6	2.55	0.3	0.3
B	0	1.3	0.75	0.9
C	0	1.3	0.45	0.9

Tracking Task Efficiency:- A = 1.0
B = 1.035
C = 1.3

Fig.4 EFFECTS OF BANDWIDTH AND DROPBACK ON TRACKING

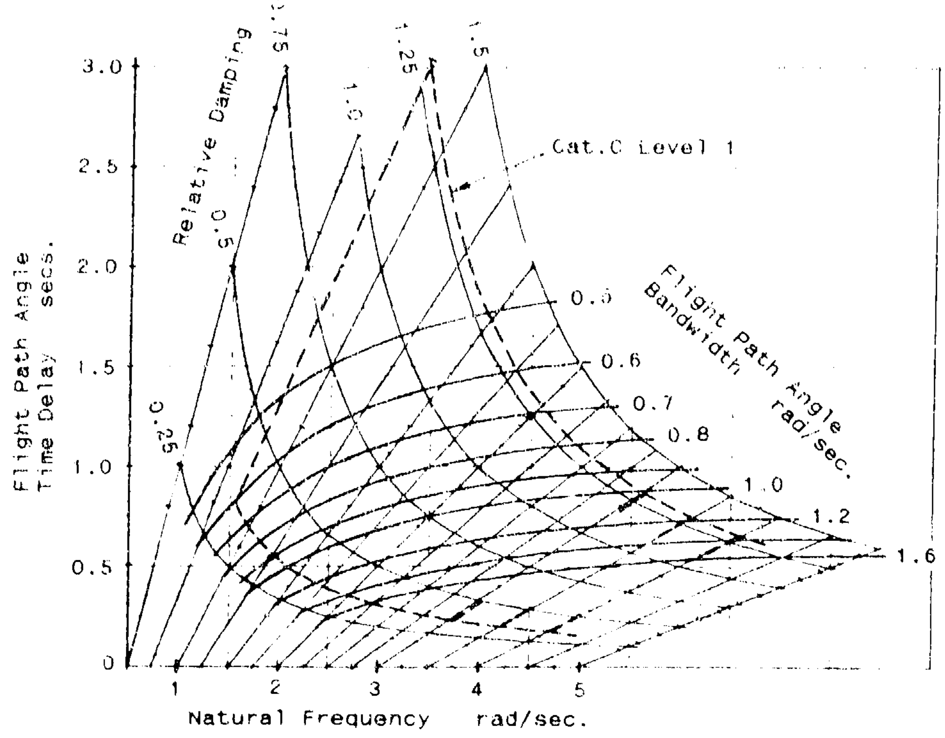


Fig.5 FLIGHT PATH ANGLE BANDWIDTH

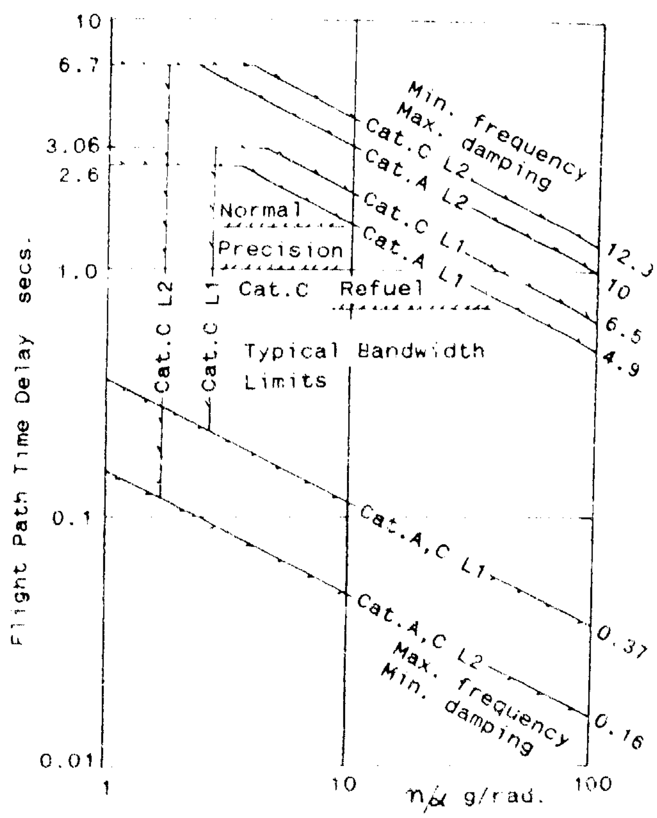
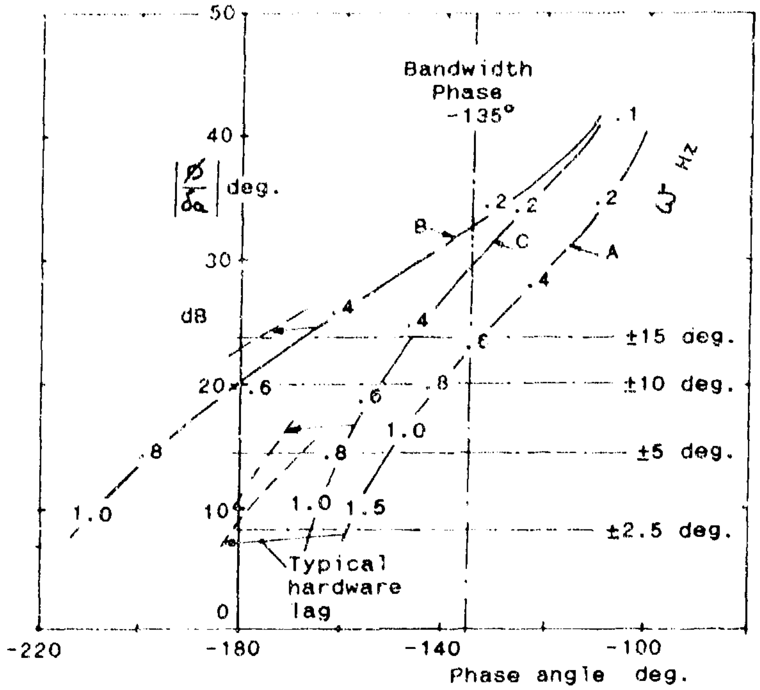


Fig.6 TRANSFORMED FREQUENCY/DAMPING BOUNDARIES



A: p = 75/(1 + 0.25S)

B: p = 75/(1 + 0.25S)(1 + 0.3S)

C: p = 75/(1 + 0.55S)

Fig.7 ROLL FREQUENCY RESPONSE CHARACTERISTICS

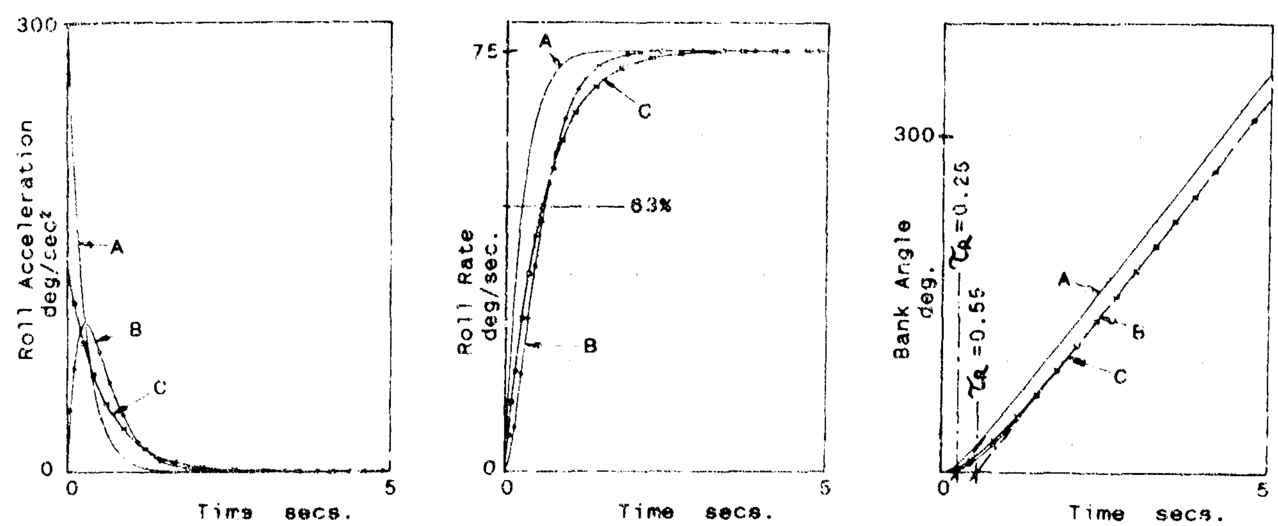


Fig.8 ROLL TIME RESPONSE CHARACTERISTICS

DEVELOPMENT OF MIL-8785C INTO A HANDLING QUALITIES SPECIFICATION FOR A NEW EUROPEAN FIGHTER AIRCRAFT

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1. Introduction

For the development of the Tornado, which started in the sixties, a draft version of MIL 8785B [1] was used as a guideline and specification for the flying qualities the airplane should have. No consideration was given at that time to the fact that requirements in [1], which were based on mathematical models of the airplane, only considered the flight mechanics of the bare airframe. An example of this is shown in figure 1, where a flight test measured transfer function "pitch rate due to elevator" is depicted which has the shape one would expect for a first over second order transfer function with a total phase change of 90 degrees. No hint was given on the method of approach to fly by wire airplanes with a full time full authority flight control system, which after all does change the transfer function of the airplane the pilot has to work with considerably. In figure 2 the transfer function pitch rate due to stick force for the same airplane and flight condition as in figure 1 clearly shows the effect. The total phase change in the frequency range shown is at least 180 degree with no apparent limit value.

In most cases a new airplane is sold on performance promises. The fact that a pilot has to be able to fly an airplane safely and efficiently in order to achieve full performance, especially in a combat airplane, is often forgotten. The Services are not seldom quite disappointed when it becomes clear that it is impossible to achieve the promised performance for reasons of conflicting flying qualities issues which demand e.g. other than performance optimal trim schedules.

This, coupled with the fact that there was not a lot of flying qualities research within the nations participating in the Tornado program, led to the situation that it was more or less only during flight test that we on the user side realized the problems involved with full authority full time flight control systems.

In the meantime, however, some research efforts have been initiated by government as well as industry (mostly government funded), which have provided us with some second thoughts on flying qualities requirements for highly augmented airplanes with a basically unstable pitch axis.

Together with industry, the four nations involved in the European Fighter Aircraft (EFA) Program decided to initiate an effort to generate a flying qualities specification for EFA based on the MIL-F-8785C [2]. Germany undertook the task of providing a draft of what is now the Handling Qualities Definition Document for EFA [3] (HQDD). In subsequent discussions and negotiations chaired by the NATO EFA Management Agency (NEFMA) the draft was turned into the final document which is now part of the EFA development contract.

Some of the issues we discussed and which might be of more general interest will be presented below. Our presentation will contain remarks on

- the equivalent system approach
- high order requirements for the pitch axis
- the carefree handling issue
- roll performance
- small lateral directional inputs
- air combat
- stall and spin

2. Longitudinal Axis

2.1 Equivalent System Approach

Lower order equivalent systems were developed as a methodology in the early seventies as part of a flying qualities assurance program in support of the F 14, prior to its first flight. This methodology was refined to the point where it could be utilized as a flying qualities criterion in Ref. [2].

Guidelines on the use of the equivalent system methodology are to be found in the background information to MIL-F-8785C [4] and in the draft of the MIL-Prime Standard and Handbook [5], where a mismatch envelope for the simultaneous match of pitch attitude and normal acceleration transfer functions (short period approximation), figure 3, is given. The following example indicated to us that we should use those mismatch envelopes with caution. A second order system with a frequency of 3 rad/sec a damping of 0,7 and an assumed n/alpha = 10 was corrupted with a prefilter having two poles and two zeros. The original system was chosen to have level 1 flying qualities.

In figure 4 the pitch acceleration response and the normal acceleration response of the resulting transfer functions are shown. Figure 5 depicts a comparison of the resulting mismatch with the mismatch envelopes, taking the uncorrupted system as the equivalent system. The resulting mismatch is clearly inside the recommended envelope.

With the Control Anticipation Parameter (CAP) computed according to its original definition,

$$CAP = \frac{\ddot{\theta}_O}{\Delta n_{SS}} = \frac{\ddot{\theta}/\delta e\,|_{s\to\infty}}{\Delta n/\delta e\,|_{s\to 0}} = \frac{\ddot{\theta}/\delta e\,|_{max}}{\Delta n/\delta e\,|_{s\to 0}}$$

the evaluation of the systems led to the results given in figure 6, placing the systems with the added dynamics in level 2 and level 3 respectively.

However, since the original system without the added dynamics qualifies as an equivalent system, it can be concluded that the mismatch envelopes as given in [5] allow too great variability in the equivalent systems.

The equivalent normal acceleration transfer function as given in figure 3 refers to the instantaneous center of rotation, a point which is not normally readily known, e.g. in flight test.

Figure 7 shows normal acceleration due to stick force transfer functions as deduced from flight tests for the front seat and the back seat respectively of the Tornado airplane under the same flight conditions. The transfer functions were derived after estimating equivalent derivatives with a MMLE type program and then computing the transfer functions. For the front seat this resulted in a numerator with a complex conjugate pair of zeros in the left half plane whereas for the back seat, a non minimum phase system was the answer.

If e.g. flight test measured transfer functions are used for the equivalent system matching process those differences have to be accounted for.

Therefore we recommend the use of equivalent model structures as given in Annex A, Table 1, which include second and fourth order numerators for the short period and phugoid modes of the acceleration transfer functions. If the transfer function referenced to the instantaneous center of rotation (COR) is still required, it could be computed with the help of the pitch acceleration due to stick force transfer function according to the following relationship:

$$\frac{n_{zCOR}}{F_s} = \frac{n_z}{F_s} + k\,\frac{\dot{q}}{F_s}$$

The arbitrary constant k has to be selected in such a manner that the numerator of n_z/F_s reduces to zero order.

Given a higher order aircraft, five parameters are available to accomplish the match: $M_{\delta e}$, ζ_{sp}, w_{sp}, $1/T_{\theta 2}$ and τ (figure 3) the new one being τ, the delay time. The delay time basically enables us to account for the additional phase shift of higher order systems as is shown in figure 8 where a time delay of .05 sec has been added to the transfer functions of figure 7. For the evaluation of stability the shape of the phase curve down to - 180 degrees is of paramount concern. The frequency range for approximation of the higher order system by the equivalent system should therefore be selected in such a manner as to provide sufficient data points to lead to a good approximation of the phase curve down to - 180 degrees and thus to a good and meaningful representation of the phase by the resulting delay time. Therefore we recommend the following procedure for definition of the frequency range of approximation.

The selected frequency range should contain the characteristic frequencies of all modes of interest. Additionally, an upward and downward margin of one octave should be provided.

Initially, a range from 0.1 to 10 rad/sec is defined, which may be modified according to the following rules if required:

For full longitudinal model with phugoid: Lower limit < 0.5 w_{np}.

For Short-Period model, with the phugoid still showing strong influence at 0.1 rad/sec: The lower limit is the frequency between the peaks of resonance of the phugoid mode and the short-period mode, at which the minimum of the pitch rate amplitude appears.

The upper limit should be > 2 w_{nsP} or > 2 w_{90} (w_{90} = frequency at which the phase angle of the pitch rate response is - 90 degrees), whichever is greater.

It must be noted that the delay time computed with the above approach has to be used for comparison with the equivalent system requirements and not delay times defined by other means as figure 9 indicates. In figure 9 four different rules for computing delay times are compared. Here it is clearly shown that the delay time computed according to the bandwidth criterion still indicates level 1 when the equivalent delay time indicates level 2 and the other two criteria never indicate level 1 at all when compared to the set of limits for time delay as given in [2].

A further variable which influences the quality of the equivalent system obtained is the number and distribution of values used in the approximation process.

It appears reasonable to select values w_k as logarithmic equidistant in order to provide for a uniform distribution of the points of support in the Bode plot. This poses no problem when frequency responses of a mathematical model are to be computed.

In contrast to the above frequencies are linear equidistant when evaluating flight test data by FFT directly. This fact introduces the following disadvantages for an approximation:

(i) the number of frequencies is very high and thus computation time is very extensive,

(ii) most of the values are situated in the upper frequency range so that this part is weighted more intensively.

Therefore, conversion of a data set with a linear distribution to a data set with an approximated logarithmic distribution is recommended. In order to accomplish this, the frequency band is subdivided into intervals of equal size in the logarithmic scale. Should more than one point lie inside such intervals, the original data are replaced by one averaged data point. In this way, a shorter data set is obtained.

The influence of the distribution of frequency points used can be identified in figure 10. A 4th-order system was used to generate three data sets with different distributions of frequencies. From these data sets, second order models (with time delay) have been approximated.

Using 1,000 linearly distributed values, a poor fit was achieved (Run A). The same data set was then converted to obtain a nearly logarithmic distribution at high frequencies. Using this converted data set, better curve fits and results were achieved (Run B). Finally, a logarithmically equally spaced data set was generated. From this set, a good curve fit was obtained (Run C) (except at very high frequencies) and the eigenvalues of the LOES are very close to those of the dominant HOS-mode.

With a logarithmic-equidistant distribution, the number of values may be kept relatively low without degrading the quality of the approximation. When all poles and zeros of the system are sufficiently damped, a density of 20 points per decade may be recommended. Systems with weaker damping, however, will require a substantially higher number of points, in order to provide coverage of all the resonance peaks and the steeper phase drop, and exact determination of the damping ratio.

The above experiences and lessons learned were drafted into a recommendation of how to apply the equivalent systems approach which then became part of the HQDD for EFA [3] and is attached to this paper at Annex A. It also contains guidelines for the use of equivalent models in the lateral directional axes.

2.2 High order requirements

Besides a detailed look at the application of the equivalent systems approach we also screened some of the higher order criteria and eventually developed one for EFA.

The criteria we started with were

- The Neal-Smith criterion in its original form [12]
- the Neal-Smith criterion as proposed in [5]
- the Neal-Smith criterion with some changes in its boundaries proposed by one of us (Mr. Marchand), and a reduced bandwidth for level 2 (2.5 rad/sec) and level 3 (1.5 rad/sec) figure 11
- the Bandwidth criterion developed by Mr. Roeger from MBB [6]
- the Nichols-plot criterion developed by Mr. Diederich from MBB [6]

The two last named criteria were among others checked against pilot ratings in a simulation study which Dornier undertook under government contract in 1985 and 1986. The most promising criterion turned out to be the Diederich criterion. The application of the Roeger criterion posed some difficulties in principle because of the way stick force gradients had to be considered for its use.

However, some modifications to the boundaries of the Diederich criterion proved to be necessary and were proposed by Dornier and DLR. Dornier combined the Diederich criterion with a criterion proposed by Gibson [13] which was also formulated in the frequency domain and presented in a Nichols plot. DLR proposed some modifications to the boundaries on the right hand side.

The resulting Nichols plot criterion, figure 12, defines limits for the normalized open loop transfer function pitch attitude, θ, due to stick deflection, δ_{es}, or due to stick force in a Nichols diagram. Normalizing means in this context that the transfer function under test has to be shifted up or down by varying the gain until it runs through 0 db at -110 deg phase lag.

Because the Nichols diagram contains no constraints for the frequency range allowed, figure 13 gives the required bandwidth for the flying qualities levels L1, L2, L3 for flight phases A, B and C.

The criterion was designed for the evaluation of closed loop flying qualities involving small stick inputs, i. e. it is applicable to essentially linear conditions only. Regions of high angle of attack may have to be excluded.

The boundaries identified by asterisks (*) in figure 12 are applicable only where provision is made for precision attitude control for fine tracking at small stick inputs. In this case the boundaries identified by the asterisk in figure 13 need not be observed for stick inputs of less than 10 mm for center stick controllers.

For the boundaries identified by a double asterisk (**), additional criteria apply to the not normalized transfer functions pitch attitude due to stick deflection (or due to stick force). At the frequency where phase lag of pitch attitude to cockpit control displacement (or force) is 180 deg, the following applies to levels 1, 2 and 3:

The rate of change of phase lag shall be less than 16 deg/rad/sec (100 deg/Hz) or if greater, then the phase rate at 190 and 200 degrees phase lag shall be significantly less than 16 deg/rad/sec (100 deg/Hz).

The amplitude shall be less than a maximum of 0.03 deg/mm or 0.022 deg/N (0.1 deg/lb) for a phase rate of 16 deg/rad/sec (100 deg/Hz), increasing to 0.05 deg/mm or 0.036 deg/N (0.16 deg/lb) for a phase rate of 11 deg/rad/sec (70 deg/Hz) or less if omega 180 > = 1.0 Hz.

The absolute limits stated above regarding stick displacement or force gradients are mainly based on experience with EAP and fly by wire Jaguar. They should really be rechecked for control stick designs vastly different from those used in the two airplanes mentioned.

DLR compared three versions ofthe Neal Smith criteria as given in figure 11, the equivalent systems approach and the above described Nichols plot criterion with the Neal Smith database [12] finding good correlation for the Nichols plot criterion just described (figure 14). In addition the combined criterion was checked by Gibson (British Aerospace) against his flying qualities database collected mainly from the fly by wire Jaguar and the experimental aircraft programmes (EAP). In the course of joint discussions Dornier, DLR and British Aerospace developed the final version of the criterion, described above, which now serves as one of the design guidelines for the development of the longitudinal flying qualities of the European Fighter Aircraft (EFA) [3].

The following limited guidance is offered for application of the criterion:

During the design phase of an aircraft project, the transfer function pitch attitude due to stick deflection is readily available as an equation and can therefore easily be compared to the criterion and the additional features, e. g. phase rate between -150 deg and -200 deg phase can be computed as local gradients. For flight test derived transfer functions more care is needed in the region of the -180 deg phase and suitable mean values of the phase rate have to be derived because of the occasional poor quality of flight test data especially near and beyond the -180 deg phase.

If the right hand side level 1 limit above 0 db is violated excessive droop back leading to pitch bubble is indicated whereas violation of the left hand limits points to sluggish aircraft behavior resulting in overshoots. Infringement of the left hand limits of Level 1 below 0 db hints at the fact that the design may be pilot induced oscillation prone. The criterion should at present be limited in its application to judging the precision tracking behavior of combat aircraft in flight conditions where essentially linear behavior can be assumed. Feasibility in the high angle of attack region will be demonstrated by the X-31A program. The original Diederich criterion was used in the design of this experimental aircraft up to high angles of attack.

3. Lateral directional axes

3.1 Roll performance

Figure 15, which is essentially taken from [7], depicts the problem. With the control power available by aerodynamic means, roll performance will deteriorate at the high angles of attack readily attainable by modern fighter airplane designs. Here, the deteriorating yawing power is the key contributing factor when considering rolls around the velocity vector. A survey of the literature and roll performance data available to us led to the conclusion that combat roll performance requirements as stated in [2] have to be waived above a certain angle of attack. For the reduction of the requirements for the time to bank through 30 deg, 50 deg and 90 deg, we adopted the following equation:

$$t_{Bank} = t_{combat} + K(\alpha - \alpha_{limit})$$

where
- t_{Bank} is the resulting time to bank allowed
- t_{combat} is the time to bank as given in [2] for the combat flight phase
- k is a suitably selected constant
- α is the angle of attack attained
- α_{limit} is the angle of attack above which he relationships may be applied.

However, it would be preferred not to need such a lax. on. But this may have to wait until thrust vectoring has become a standard design feature for fighter airplanes.

3.2 Additional roll rate requirements for small inputs

Modification of the requirement "additional roll rate requirement for small inputs" (see [2]) was a consequence of the experience gained from the Tornado Trials Program.

The Mil Specification of this paragraph read:

"The value of the parameter p_{osc}/p_{av} following a yaw-control-free step roll-control command shall be within the limits as shown in figure 16 for Levels 1 and 2. This requirement applies for step roll-control commands up to the magnitude which causes a 60-degree bank angle change in 1.7 T_d seconds".

Dutch Roll Motion provides the phase angle between the rate of roll and the angle of sideslip, which in turn is a crucial factor in deciding which x-axis is to be used. In the Background Information to MIL-F-8785B [11] an explanation is provided that this angle for the linearised system of equations of motion for the lateral directional axis with coefficients of conventional aircraft is to be found in one of the following areas:

$90^\circ < \text{Phase}\ (p/\beta) < 180^\circ$ or
$270^\circ < \text{Phase}\ (p/\beta) < 360^\circ$.

When deviations from these values occur they are so minimal that configurations can still be clearly allocated to the one or the other axis.

In the case of modern fighter aircraft with complicated control systems values lying outside the normal areas can, however, be registered. In the course of the official trials of the Tornado, mathematical models were established from flight trial data by means of system identification. The phase angle pattern p to β as ascertained in one particular test series is shown at figure 17. The values lay in the region of 180° and thus bordered on the area in which the phase angles were expected. . % of the evaluation values were even found close to 225°, with the result that small changes in the parameters allowed the phase to wander under or over this limit.

Now, exceeding a limit in such a fashion after small changes have been made means that each time a change is made the second x-axis must be used. An evaluation which had initially provided a value in the area for Level 1, subsequently intimated after an abrupt transition that the characteristics would not even be sufficient for Level 2 (see figure 18).

In order to obviate such discrepancies, the hypotheses for the derivation of the criterion were ascertained and included in the flight characteristic specifications for EFA. The requirements are as follows:

$\text{abs}\ (L'_r * Y_p) \ll \text{abs}\ (L'_p)$
$\text{abs}\ (L'_r * Y_\beta) \ll \text{abs}\ (L'_\beta)$
$\text{abs}\ (4 * g * L'_r) \ll \text{abs}\ (V * L'^2_p)$
$\text{abs}\ (L'_r) \ll \text{abs}\ (L'_\beta)$.

If any one of these conditions is not fulfilled, then the requirement discussed here for the underlying parametric representation is no longer applicable.

This limitation of the scope of application will obviate application of the criterion based on false premises. In addition there will be no absurd abrupt transitions in the evaluation due to small changes in parameter.

4. Miscellaneous

4.1 Carefree Handling

When industry tries to sell you a new fighter design, "Carefree Handling" is one of the colorful terms thrown at you to make you feel good about the product you are going to order.

Therefore the questions which arise are: what could this term mean, what would be technically feasible, how would it support a pilot in accomplishing his task.

As simulations have shown, a feature which protects the pilot from exceeding vital airplane limits frees his mind from the task of watching e.g. angle of attack, normal acceleration etc.. He would then be in a position to direct more of his attention to the combat task at hand and would be less concerned with flying the airplane. This in turn would make him more successful against an otherwise equal opponent. Therefore limit setting systems make sense, especially if the airplane cannot be designed inherently carefree from its aerodynamic design point of view. It is exactly this, however, which would solve the problem of the angle of attack, but not of normal acceleration i.e. structural limits.

The minimum components a "Carefree Feature" should be able to offer should be, in our opinion:

- Automatic prevention of stall departure, autorotation and spin. This would have to include incidence and sideslip control and limiting for both positive and negative values.
- Normal acceleration control and limiting for positive and negative values.
- The capability for safe but aggressive lateral/directional maneuvering inside the full angle of attack/normal acceleration envelope.
- Unrestricted use of the throttle and of other means which decelerate or accelerate the airplane.
- Automatic scheduling of limits according to configuration and loading conditions.

Together with the desire to exploit the aerodynamic/flight mechanic possibilities of the basic airframe design to its full extent, i.e. to provide the pilot with an envelope which gives him superior combat capabilities, the above will prove to be quite a design challenge.

4.2 Stall and Spin

As a "Carefree Handling" feature should include angle of attack limiting functions, then definition of stall angle of attack as a basis for defining important airplane speeds and speed limitations would have to be reconsidered. Stall angle of attack may not be determined by an air flow related phenomenon such as a normal acceleration break but simply by some suitable limit set by some limiting function inside the flight control system.

This, in turn, would change emphasis of flight test of stall and spin. The characteristics of stall departure and post stall gyrations are not of prime concern anymore. Of prime concern, however, are the following questions:

> How suitable are the flight control system limiting functions set and implemented, given the task of the airplane.
>
> Can they easily be defeated by any pilot action?

A thorough flight test of departure and post stall gyrations would become necessary in order to define the changes, including changes to airplane configuration, which would make it safe to fly only if a limiting feature proved to be incapable of doing its job of protecting pilot and airplane and at the same time providing a sufficient envelope for the task.

Therefore requirements for stall and spin were formulated for EFA to allow for a stepwise approach. Further testing depending on results of steps completed. It may well be that the airplane will never be spun if the only way of achieving this is to switch off the flight control system completely. However, this approach places a heavy burden on wind tunnel testing and simulation.

4.3 The Air-to-Air Combat Flight Phase

Any requirement drawn up is in effect the application of past experience in an attempt to forecast the future. However, there is a way out for handling qualities requirements for piloted airplanes. All requirements defined in terms of envelopes for open or closed loop system parameters have been developed by matching pilot opinion to those system parameters. Where no numeral can be defined, pilot opinion itself may be used and this is done frequently. Based on experience gained at the Air Force Flight Test Center with the handling qualities during tracking (HQDT) approaches [8] and our own experience using this technique [9], [10], we introduced the following as a requirement.

During air-to-air combat, the pilot will perform a series of gross turning, rolling and pull-up maneuvers, target acquisition maneuvers, and precise target tracking. To help ensure that the EFA has satisfactory Handling Qualities during the acquisition and precise tracking segments of the air-to-air combat engagement, pilot opinion ratings must clearly indicate Level 1 Handling Qualities. Such ratings shall be achieved within the primary air combat flight envelope.

We are now preparing a simulation at IABG in their dual dome combat simulator where we will more closely define and match to the European Fighter Aircraft the methods (pilot briefing papers, combat maneuvers, pilot questionnaires and debriefing procedures) to be used during simulation and flight test to prove compliance with the above requirement.

5. Summary

MIL-F-8785C [2] was used in a sense which in the US was introduced with the development of the MIL-standard and handbook concept, namely, to provide a framework for the development of flying qualities for airplanes. In our case [2] was stripped of everything not related to a fighter type airplane and amended where we believed it was necessary. From the many details which had to be considered, our paper concentrated on areas of the HQDD [3] where we felt it necessary to give better guidance for the application of the criteria (equivalent system approach, small roll step inputs), where new criteria had to be introduced (high order system criterion, carefree handling, pilot evaluation in simulated air-to-air combat) and where the emphasis of existing ones had to be changed or amended (stall and spin, roll performance).

We feel that the concentrated efforts of industry and government have produced a set of flying qualities criteria which will pave the way to excellent flying qualities of the European Fighter Aircraft.

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ANNEX A

GUIDELINES FOR THE DETERMINATION OF EQUIVALENT MODELS

A.1 General Procedure

A.1.1 General Approach

For the determination of equivalent models, one or simultaneously two frequency responses of the high order system are approximated by complex functions of the formula

$$F_{ik} = F_i(\omega_k) = \frac{Z_i(\omega_k)}{N(\omega_k)} e^{\ldots}$$

where Z_i and N are polynomials of the formula

$$Z_{ik} = Z_i(\omega_k) = \sum_{v=0}^{m_i} b_{vi}(j\omega_k)^v$$

$$N_k = N(\omega_k) = \sum_{v=0}^{n} a_v(j\omega_k)^v$$

(with i = 1, 2). The coefficients a_v of the denominator polynomial, together with coefficients b_{vi} (i = 1, 2) of the numerator polynomials and the dead times τ_i (i = 1, 2) are determined in such a way that a cost function is minimized with given numerator degrees m_i (i = 1, 2) and a given denominator degree n. To obtain unambiguous solutions, one of the coefficients for each frequency response must be given beforehand, which is most easily done by specifying

$$a_0 = 1$$

A.1.2 Definition of the Cost Function for One Frequency Response

The definition must ensure that the result of the approximation is independent of the standardization of the frequency responses. In the following, two types of the cost function are stated which fulfill this definition (for the approximation of one frequency response to begin with).

a) Gain Phase Error Function

$$C = \frac{1}{N} \sum_{k=1}^{N} [\Delta G_k^2 + f \Delta P_i^2]$$

where

C	= Cost of function or mismatch
N	= Number of discrete frequencies w_k
G_k	= Deviation of the amplitude at w_k (in dB)
ΔP_k	= Deviation of the phase at w_k (in degrees)
f	= Weighting factor between amplitude and phase

The factor f is generally stated with 0.017 or 0.020, but can be varied if required.

b) Maximum Likelihood Error Function

Based on the assumption that the real and imaginary parts of the approximation error are statistically independent Gaussian random variables with constant relative standard deviation, the ML error function reads as follows:

$$C = \frac{1}{N} \sum_k \frac{|\Delta F_k|^2}{|F_k|^2}$$

where

ΔF_k	= Complex value of deviation at w_k
F_k	= Value of frequency response at w_k

The ML error function can be transformed by approximation into an amplitude-phase error function for small errors with a resultant factor f = 0.023 between phase and amplitude terms.

A.1.3 Cost Function for Two Frequency Responses

On simultaneous approximation of two frequency responses, the deviations (errors) along the two frequency responses are accumulated:

$$C = C_1 + gC_2$$

where

C_1, C_2 = Cost functions for the 1st and 2nd frequency responses as stated above

g = Weighting factor between the two frequency responses generally g = 1

The weighting factor g may be used to consider differing error levels of the two frequency responses.

A.1.4 Selection of the Approximation Method

A specific method will not be suggested, as any minimum finding method may be applied. When selecting such method, however, the following items should be given due consideration:

- structure of cost function (e.g. whether or not the gradients may be analytically expressed)
- possibility of fixing or varying specific parameters within given limits
- possibility of individual weighting (e.g. along the frequency or by means of amplitude and phase errors).

A.1.5 Selection of Frequencies

When selecting frequencies, the following items should be considered:

- low-frequency cutoff
- high-frequency cutoff
- distribution of values, e.g. linear or logarithmic equidistant along frequency
- number of frequencies

a) Frequency Cutoffs

Cutoff frequencies are generally dependent on the examined system. The eigenvalues, the influence of which is to be evaluated, must be inside this frequency range of approximation. Standard default values are given in paragraph A.2.3.

b) Distribution of Values

It appears reasonable to select values w_k as logarithmic equidistant in order to provide for a uniform distribution of the points of support in the Bode plot. This poses no problem when frequency responses of a mathematical model are to be computed.

In contrast to the above, frequencies are linear equidistant when evaluating flight test data by FFT directly. This fact introduces the following disadvantages for an approximation:

- the number of frequencies is very high and thus computation time is very extensive
- most of the values (as seen in Bode plot) are situated in the upper frequency range so that this part is weighted more intensely.

Therefore, conversion of these values with a linear distribution to values with an approximated logarithmic distribution is recommended. In order to accomplish this, the frequency band is subdivided into intervals of equal size in the logarithmic scale. Should more than one point lie inside such intervals, both frequency and frequency response values are averaged. In this way the original points are replaced by average values and a shorter data set is obtained.

c) Number of Frequencies

With a logarithmic-equidistant distribution, the number of values may be kept relatively low without degrading the quality of the approximation. When all poles and zeros of the system are sufficiently damped, a density of 20 points per decade may be recommended. Systems with weaker damping, however, will require a substantially higher number of points in order to provide coverage of all the resonance peaks and the steeper phase drop, and exact determination of the damping ratio.

A.1.6 Assessment of Curve Fits

No binding statement can be made as to when an approximation will be useful or not. Therefore, the use of error tolerance limits - as stated in US literature - is not to be recommended. The safest method is to make use of the experience of, and lessons learnt by, the person performing the assessment. In order to accomplish this, it is recommended that the assessment be carried out by interactive computer programs providing an immediate graphical display of the frequency response curves of both the higher-order system and the low-order model. It will be necessary to determine which parameters of the equivalent model are of special importance and in which frequency ranges these parameters will be of particular influence. Thus it is quite possible that one or several parameters of the equivalent model may not be usable because the curve fit in the respective frequency range is inadequate.

A.1.7 Assessment of Parameters

Even with perfect curve fit it cannot always be ensured that the approximation will produce meaningful and unambiguous values for the parameters of the equivalent model. Problems should be expected when output data (e.g. for frequency outputs obtained on flight test) are of poor quality or when model structure is inadequate for the problem. The following paragraphs are intended to provide some assistance to this end.

a) Frequency Response Values

When analyzing flight test data by FFT, the confidence limits and/or the coherence function should also be considered. For example, the frequency ranges for which the value of the coherence function is distinctly < 0.7 (maximum possible and most favorable value is 1.0) should be excluded from further computation.

b) Correlation of Parameters of Highly Damped Systems

With systems displaying a damping ratio of about 1.0 or with 2 adjacent real eigenvalues, the frequency response does not show distinct peaks. In such cases the natural frequency and the damping ratio may be increased or decreased simultaneously without achieving any substantial change in the frequency response curve.

This results in a margin of uncertainty which can become as high as ± 30 % and must be also considered when applying these parameters to Handling Qualities criteria of aircraft.

c) Correlation of Time Delays and Time Constants

When several time constants of the numerator polynomial and the denominator polynomial and an equivalent time delay are to be determined for a frequency response, the results may prove to be strongly correlated, so that several sets of values of the parameters may belong to very similar frequency responses. Because $e^{-T_s} \approx 1/(1 + T_s)$, such influences of time delays and lag time constants cannot be discriminated at low frequencies. If problems are encountered they might be solved by applying one of the following measures:

Reduction of the order of numerator and/or denominator.
Fixing of individual parameters.
Change of frequency range, e.g. extension to higher frequencies.
Selection of other or addition of further frequency responses which can contribute to obtaining better

information on specific parameters.

d) Correlation Between Poles and Zeroes

When poles and zeroes are close together, their location in the s plane has scarcely any influence on the frequency response, and the associated natural frequencies cannot be determined unambiguously. If problems are encountered they may be solved by applying one of the following measures:

Reduction of the order of numerator and denominator, when a single frequency response is to be assessed only.
Fixing of individual parameters, e.g. the eigenvalues of the denominator.
Addition of a further frequency response, in which poles and zeroes are more distinctly separated.

A.2 Determination of Equivalent Models

The general structure of equivalent models has been outlined in Section A.1.1 above. Special information for the individual models has been compiled in Table A.1. In doing so, potentially known elements of the model, e.g. factors 1/s or $1 + T_{\theta 2}s$ at a fixed $T_{\theta 2}$, were included in the transfer functions to be adjusted. This measure decreases the order of the models and thus computation effort without invalidating the cost functions of Sect. A.1) and thus the results.

A.2.1 Structure of the Models for Longitudinal Motion

The Full Longitudinal Model includes both, the phugoid mode and the short period mode, and will be employed in cases where both modes are not distinctly separated. The Short Period Model may always be employed in those cases where the modes are distinctly separated from each other in their frequency responses.

The Phugoid Model is listed for the sake of completeness, although it need not necessarily be employed, as stability may easily be verified without using an equivalent model.

The numerator time constant $T_{\theta 2}$ of the pitch frequency response determines one zero digit of the transfer functions (1) and (5). It may either be computed from the aircraft data and preassigned as a fixed value, or it may be determined in the course of the approximation. When fixed, a linear factor may be extracted from the model and correspondingly included in the values of the transfer function. To accomplish this, the values available are divided by $(1 + T_{\theta 2}s)$, and the degree of the numerator of the equivalent model reduced by 1. In this way we obtain the transfer functions (2) and (6) instead of (1) and (5).

To improve the validity of the equivalent model, approximation of a transfer function of normal acceleration (3, 4, 7 or 8) in conjunction with a transfer function of the pitch motion (1, 2, 5 or 6) is recommended. In doing so, we must differentiate whether the values available are for the so-called "center of rotation" (COR) or for another point, e.g. the c.g. or pilot station. When the reference point coincides with the COR, the numerator degree of the Short-Period Model = 0 (or = 2 with the phugoid mode model). For other points of reference, vertical acceleration contains a component of q dot and thus an additional quadratic component in the numerator (transfer functions (4) and (8)). This generally is the case during the assessment of test data, inasmuch as a computation of vertical acceleration for the COR is not possible without exact knowledge of the derivatives.

A.2.2 Structure of the Models for Lateral Motion

The object of approximation of equivalent models for lateral motion is to determine both the parameters of the dutch roll and the roll time constant. Therefore, two different transfer functions are to be approximated simultaneously, one for each dominating mode. Best suited for this purpose are β/F_r and Φ/F_a (transfer functions (10) and (11) or (13) and (14) from Table A.1). A third transfer function, β/F_a (12) has also been given. This function is only required, however, when the criteria 3.3.2.2.1, 3.3.2.3, 3.3.2.4 or 3.3.2.4.1, [2], which contain $\Delta\beta$ or β on the basis of roll control inputs, are to be applied.

Apart from the dutch roll mode and the roll mode, the "Full Lateral Model" also includes the spiral mode. For the purpose of assessments conducted outside the frequency range of the spiral mode, the model may be reduced by one pole by multiplying the transfer function to be assessed by s. This results in the model with transfer functions (13) through (15).

In cases where application of the above mentioned models does not render useful results, we can attempt to determine the parameters step by step, using the reduced model ((16) and (17)). To accomplish this we start with transfer function (16) above and only determine ζ_d, ω_{nd} and τ_R initially. Values of ζ_d and ω_{nd} then remain fixed during the second step, where τ_R and τ_e are determined by approximation of (17).

A.2.3 Frequency Ranges for the Approximation

The selected frequency range should contain the characteristic frequencies of all modes of interest. Additionally, an upward and downward margin of one octave should be provided. To determine the frequency range, the following procedure may be used:

Initially, a range from 0.1 to 10 rad/sec is defined, which may be modified on the basis of the following rules if required:

Lower limit for longitudinal motion:

For full longitudinal model with phugoid: < 0.5 ω_{np}

For Short-Period model, with the phugoid still showing strong influence at 0.1 rad/sec: The frequency between the peaks of resonance of the phugoid mode and the short-period mode, at which the minimum of the pitch rate amplitude appears.

Upper limit for longitudinal motion: The greater of $2\omega_{nSP}$ and $2\omega_{90}$(ω_{90} = frequency at which the phase angle of the pitch rate response is - 90°).
Lower limit for lateral motion:
For model with spiral Mode: < $0.5/\tau_s$
For model without spiral mode: < $0.5/\tau_R$ and < $0.5\omega_{nd}$.
Upper limit for lateral motion: $2\omega_{nd}$ minimum.

Model	Transfer function to be approximated	Order of numerator	Order of denominator	Time delay	Remarks	Handling qualities characteristics to be determined
Full longitudinal model	(1) $\frac{\theta}{F_S}$	2	4	τ_q	$T_{\theta 2}$ free or fixed	ζ_p or T_2, ζ_{SP}, ω_{nSP}, $T_{\theta 2}$, τ_q
	or (2) $\frac{\theta}{(1+T_{\theta 2}s)F_S}$	1	4	τ_q	$T_{\theta 2}$ fixed	
	(3) $\frac{n_{COR}}{F_S}$	2	4	τ_n	n at center of rotation	
	or (4) $\frac{n}{F_S}$	4	4	τ_n	n at any other point	
Short-period model	(5) $\frac{q}{F_S}$	1	2	τ_q	$T_{\theta 2}$ free or fixed	ζ_{SP}, ω_{nSP}, $T_{\theta 2}$, τ_q
	or (6) $\frac{q}{(1+T_{\theta 2}s)F_S}$	0	2	τ_q	$T_{\theta 2}$ fixed	
	(7) $\frac{n_{COR}}{F_S}$	0	2	τ_n	n at center of rotation	
	or (8) $\frac{n}{F_S}$	2	2	τ_n	n at any other point	
Phugoid model	(9) $\frac{\theta}{F_S}$	1	2	--	model without time delay	ζ_p, ω_{np}
Full lateral model	(10) $\frac{\beta}{F_r}$	3	4	τ_β		ζ_d, ω_{nd}, τ_R, τ_S, T_{2S}, ζ_{RS}, ω_{nRS}, τ_β, τ_ϕ
	(11) $\frac{\phi}{F_a}$	2	4	τ_ϕ		
	(12) $\frac{\beta}{F_a}$	2	4	$\tau_{\beta a}$	*)	*)
Model without spiral mode	(13) $\frac{s\beta}{F_r}$	3	3	τ_β		ζ_d, ω_{nd}, τ_R, τ_β, τ_ϕ
	(14) $\frac{s\phi}{F_a}$	2	3	τ_ϕ		
	(15) $\frac{s\beta}{F_a}$	2	3	$\tau_{\beta a}$	*)	*)
Reduced model for stepwise approximation	(16) $\frac{\beta}{F_r}$	0	2	τ_β	Separate appr. of (16) and (17)	ζ_d, ω_{nd}, τ_β
	(17) $\frac{\phi\,(s^2+2\,\zeta_d\,\omega_{nd}\,s+\omega_{nd}^2)}{F_a}$	2	1	τ_ϕ	ζ_d, ω_{nd} fixed	τ_R, τ_ϕ

Note:

*) Transfer function required for criteria of 3.3.2.2.1, 3.3.2.3, 3.3.2.4, 3.3.2.4.1 and for the determination of $|\phi/\beta|_d$, $\measuredangle(p/\beta)$, $\Delta\beta/k$, ϕ_β

Transfer function models of the form

$$F(s) = \frac{b_0 + b_1 s + b_2 s^2 + \ldots}{1 + a_1 s + a_2 s^2 + \ldots}\, e^{-s\tau}$$

TABLE A.1. Structure of Equivalent Models

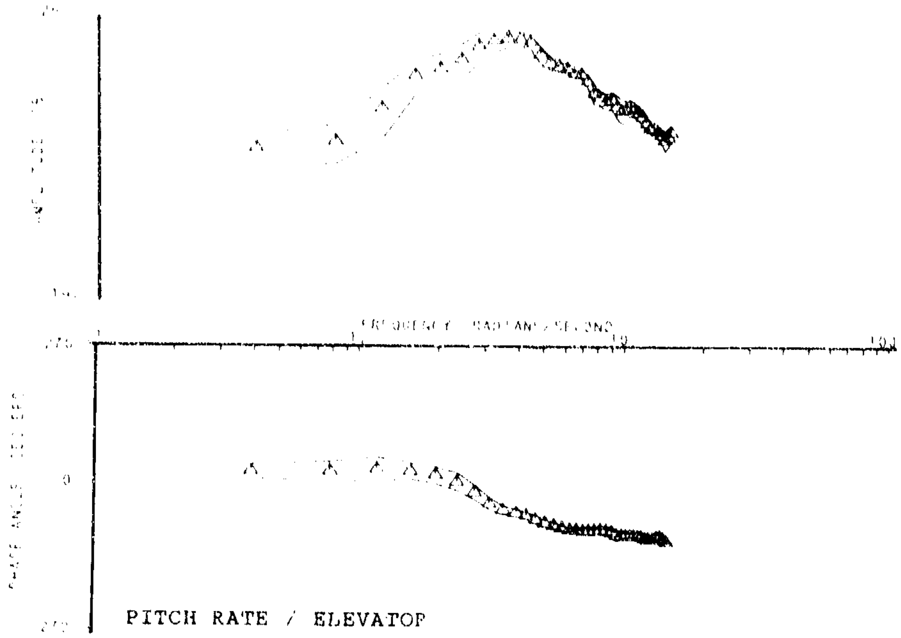


Fig. 1 Flight Test Derived Transfer Function
Pitch Rate / Elevator

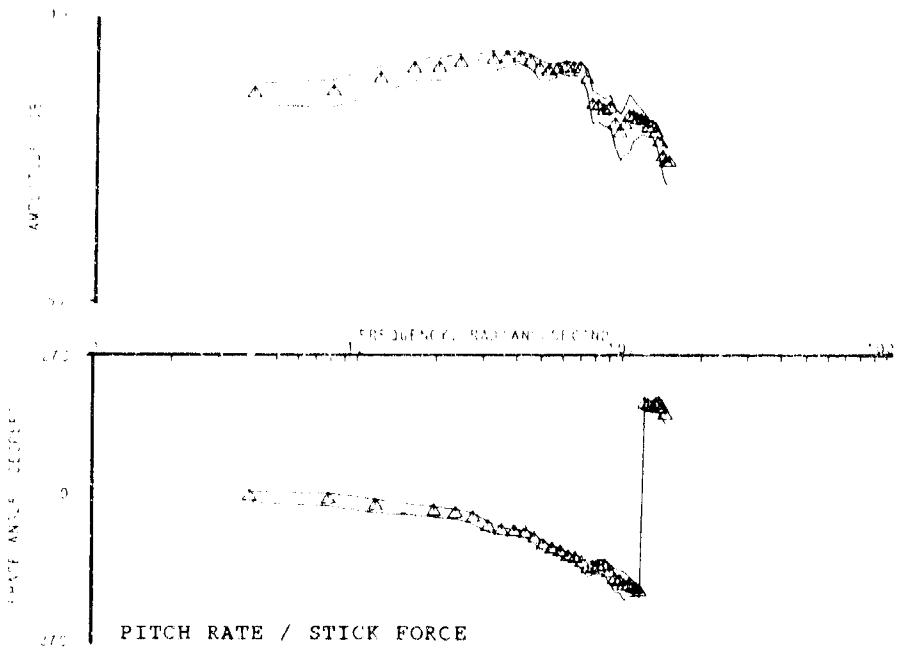


Fig. 2 Flight Test Derived Transfer Function
Pitch Rate / Stick Force

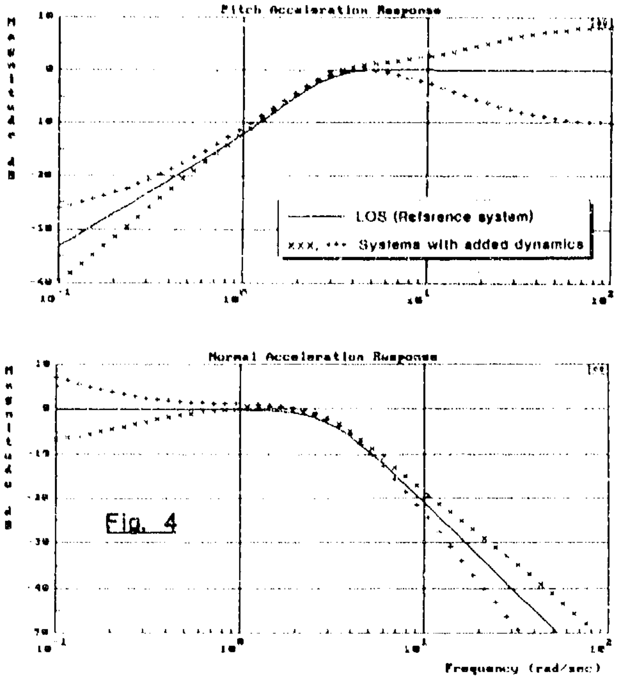
$$\frac{q}{F_s} = \frac{b_0(1 + T_{\theta 2e}s)}{\omega_{spe}^2 + 2\zeta_{spe}\omega_{spe}s + s^2} e^{-s\tau_{qe}}$$

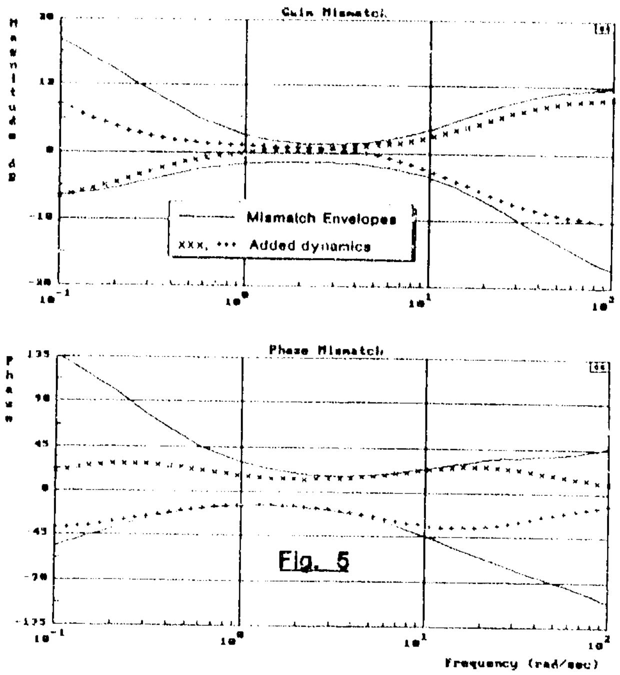
$$\frac{n_z}{F_s} = \frac{b_1}{\omega_{spe}^2 + 2\zeta_{spe}\omega_{spe}s + s^2} e^{-s\tau_{ne}}$$

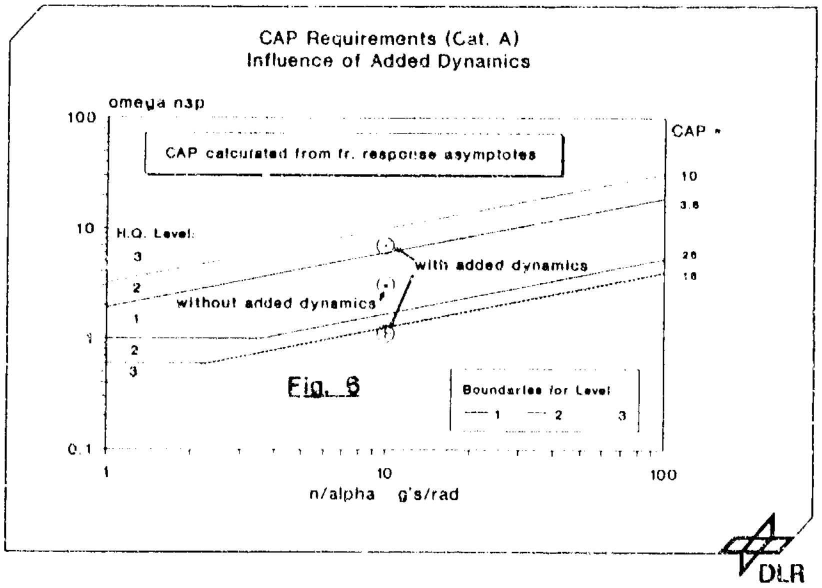
Fig. 3 LOES Short Period Approximation

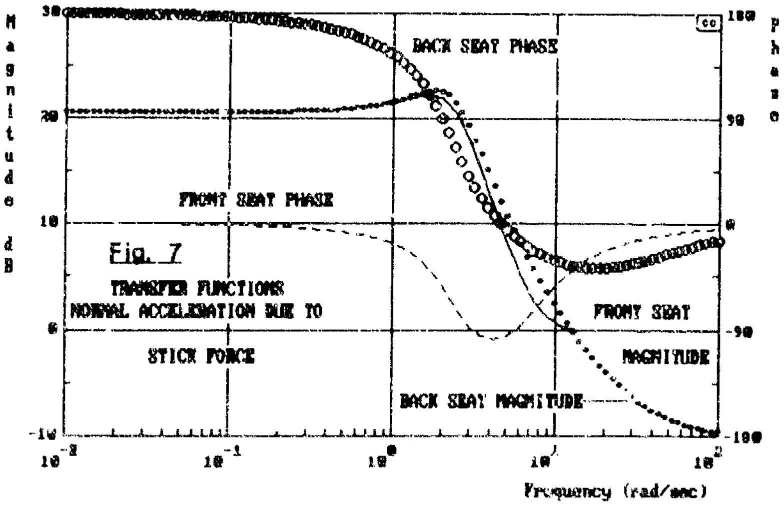
EXAMPLES OF ADDED DYNAMICS

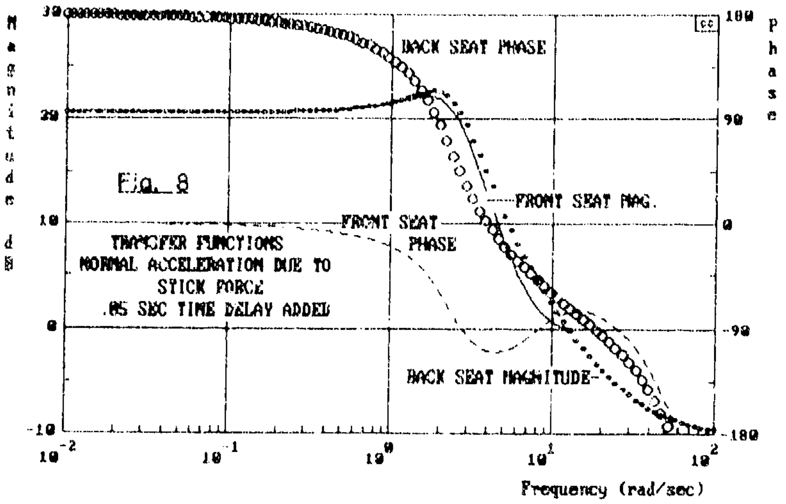
COMPARISON OF ADDED DYNAMICS
AND MISMATCH ENVELOPES

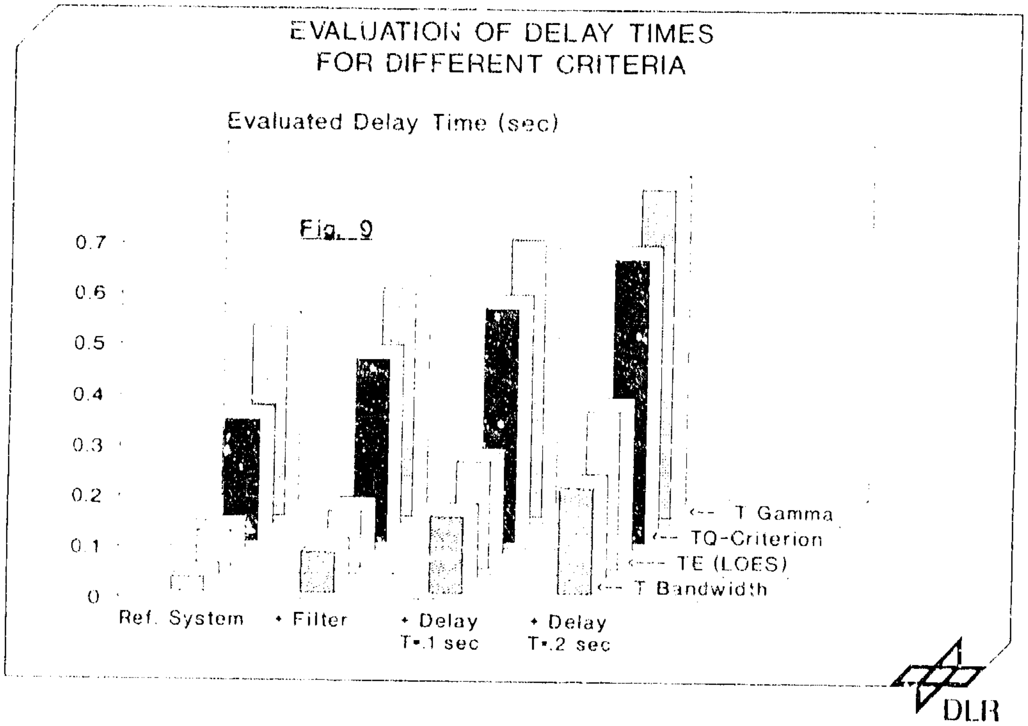






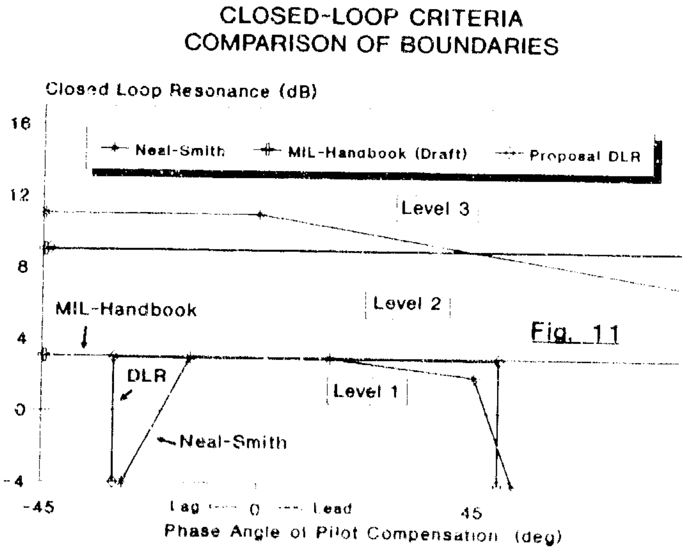






System	ζ_1	ω_1	ζ_2	ω_2	τ
HOS	.25	1.0	.5	30	0
LOES, RUN A	.046	2.63	-	-	.04
LOES, RUN B	.19	1.16	-	-	.04
LOES, RUN C	.23	1.05	-	-	.04

Fig. 10 Influence of Frequency Distribution on LOES Results



Bandwidth Requirements for DLR Proposal

Level 1: 3.5 rad/sec
Level 2: 2.5 rad/sec
Level 3: 1.5 rad/sec

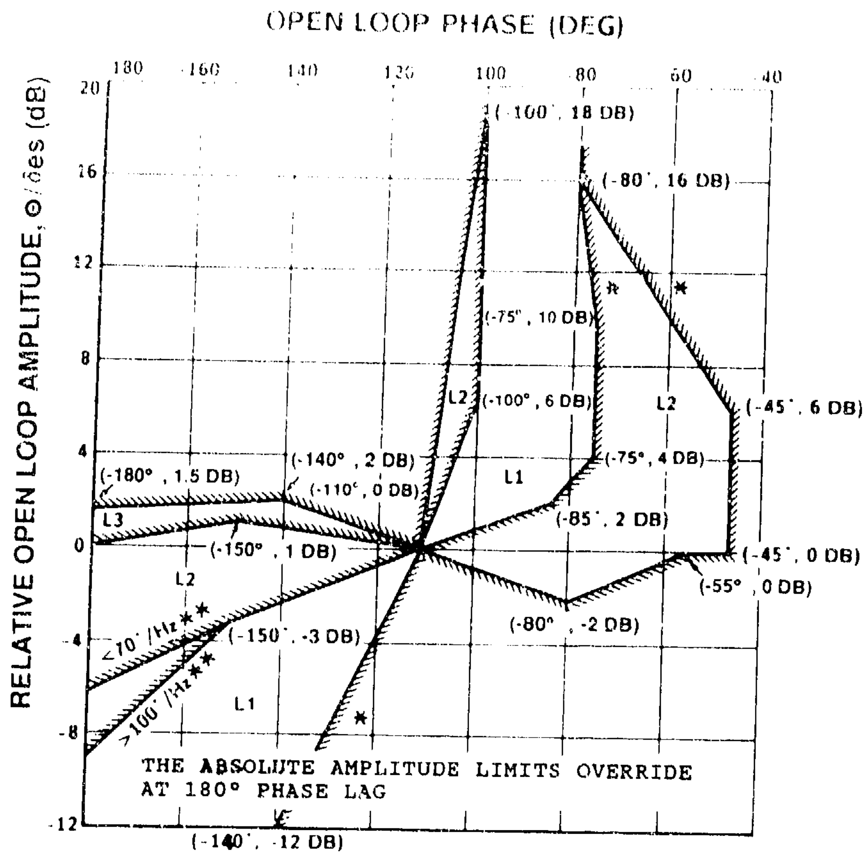


Fig. 12 High Order System Criterion
Pitch Attitude Frequency Response Limits

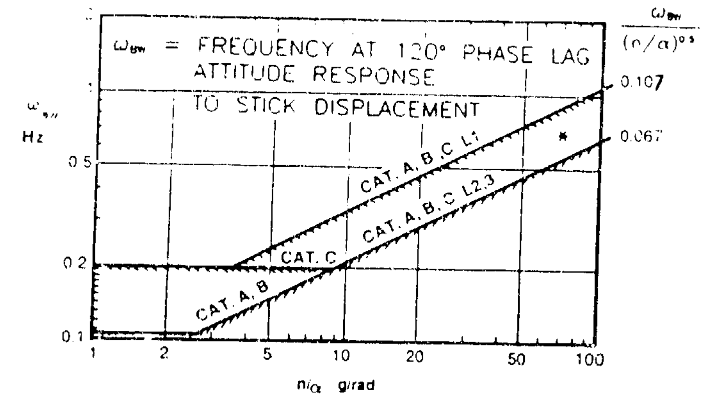
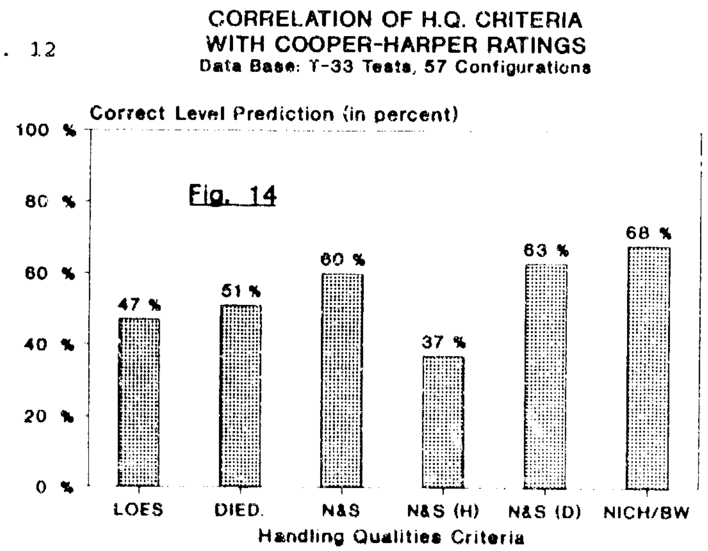


Fig. 13 Bandwidth Requirement for the
High Order System Criterion of Fig. 12



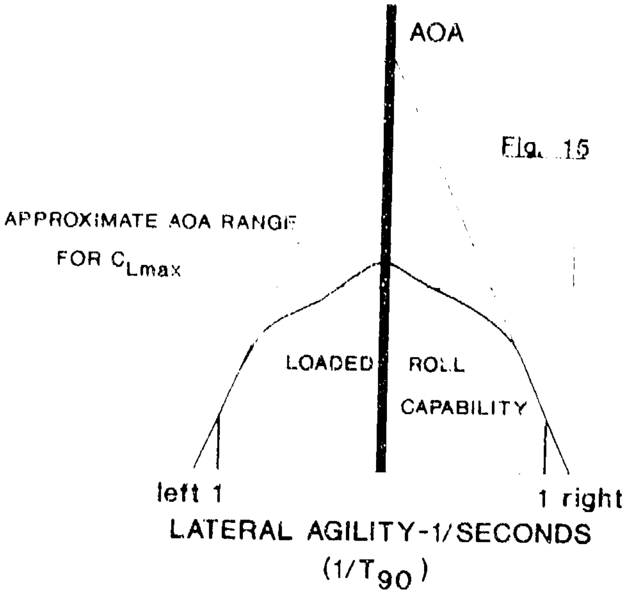
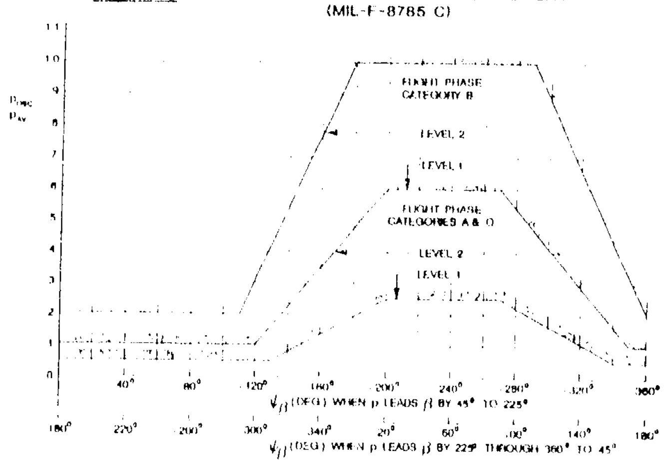


Fig. 15

Fig. 16 ROLL RATE OSCILLATION REQUIREMENT (MIL-F-8785 C)



DISTRIBUTION OF PHASE ANGLE p/B (TORNADO FLIGHT TEST EVALUATIONS)

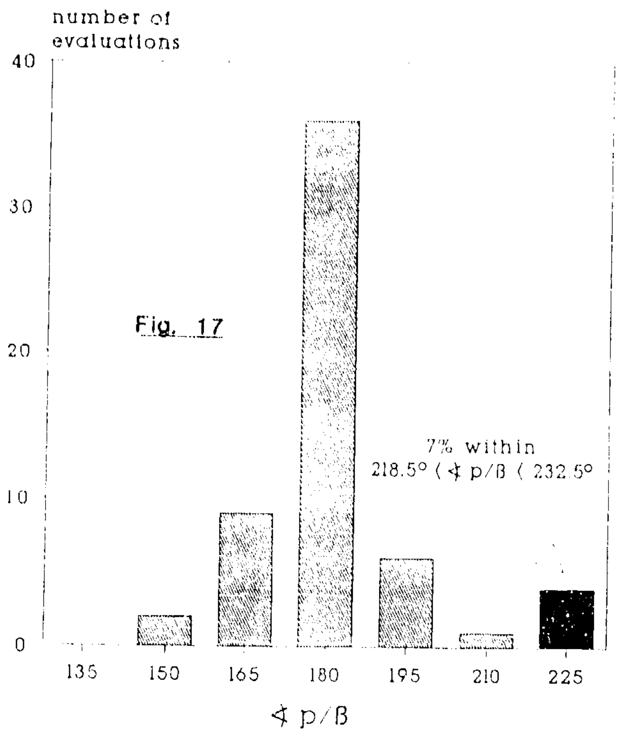
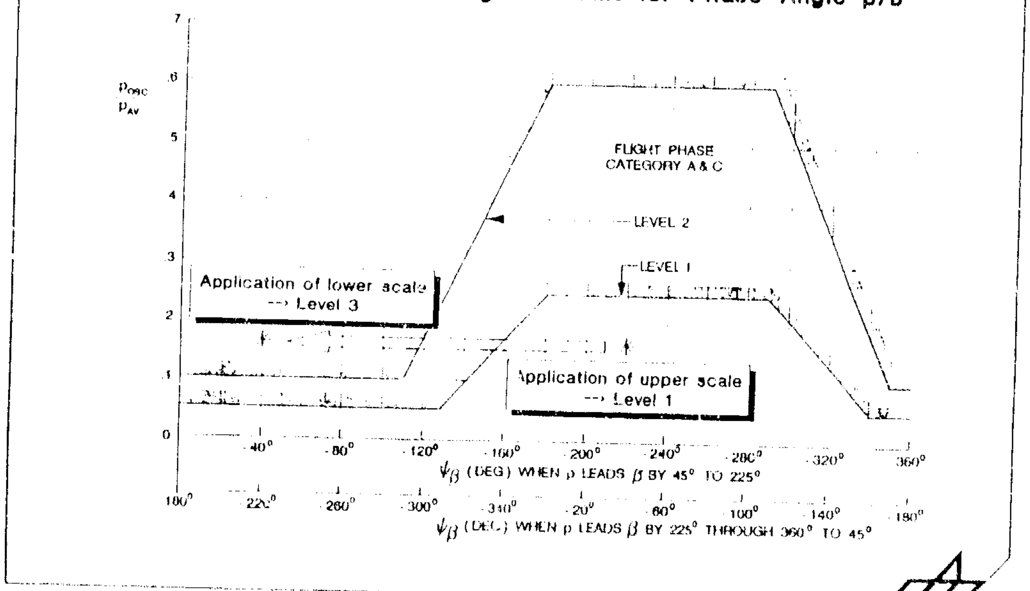


Fig. 17

Fig. 18 Transition of an Evaluation Point on Exceeding the Limit for Phase Angle p/B



DLR

Do Civil Flying Qualities Requirements Address Military Missions for "Off-the-Shelf" Procurement?

by

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Abstract:

"Off-the-shelf" procurement of civil aircraft for use by the military services is a tradition dating back to the earliest days of aviation. This relieves the military of the responsibility for development costs, takes advantage of civil designs already in existence, and has resulted in many capable - even famous - aircraft being added to the military inventory. However, while civil aircraft missions have remained relatively unchanged for over half a century, new military missions have continued to evolve: radar surveillance, battlefield management, aerial refueling, and routine low-level high-speed operations being examples. Yet, the military services still procure civil-certificated aircraft to accomplish these demanding new missions.

In the United States, Federal Aviation Regulations (FARs) 23 and 25 and their predecessors (e. g. CAR 4b) are the certification standards for civil aircraft. The primary objective of these regulations is to insure a minimum standard of airworthiness. Flying qualities requirements make up only a small portion of these regulations, and address primarily static stability characteristics. This has sometimes led to undesirable flying qualities when attempting to perform demanding military missions with civil-certificated aircraft.

The unique military missions are addressed in the U. S. military flying qualities specification, MIL-STD-1797A, and its predecessors (MIL-F-8785 series). These military specification requirements are compared to the civil (primarily FAR 25) requirements to substantiate their applicability to off-the-shelf procurement. Specifically, where military and civil missions differ, military flying qualities requirements should be invoked. To illustrate this, several examples will be examined. Finally, the future of off-the-shelf procurement will be contemplated, some implications discussed, and recommendations made.

Introduction:

From the earliest days of military aviation in the United States, many aircraft have been procured "off-the-shelf" by the armed forces. The first military aircraft - the Wright Flyer of 1909 - was an off-the-shelf purchase, albeit with a simple (by today's standards) military specification created to cover the aircraft. Other famous off-the-shelf aircraft include the DC-3, 4, 6, 9, and 10, Boeing 707, 727, 737, and 747 variants, and various small transports and trainers. In some cases, these aircraft were procured "as is", while in other cases military-requested modifications were made to the aircraft due to unique missions. In many of these procurements, civil certification was accepted rather than requiring compliance with military specifications.

When considering military and civil aviation operations, there are many applications which are military specific. Some of these, such as aerial combat, battlefield area interdiction, and close air support are obvious. These missions require aircraft produced exclusively for their accomplishment, and which are purchased exclusively by military services. Such aircraft are required to comply with military specifications as a matter of course. Other missions, such as basic training and air transport, are common to civil and military operations, and contain common elements: takeoff, climb, cruise, descent, and landing. Civil and military flying qualities specifications both address these basic missions and tasks. Seeing this, one would conclude that it may not be necessary to have different specifications, and one set should be able to adequately address aircraft performing these missions. More specifically, civil certification should be adequate to cover such aircraft.

The United States Air Force, recognizing certain advantages in procuring existing designs when possible, has provided guidance and direction in Air Force Regulation (AFR) 80-36 (Reference

1). This regulation addresses procurement of commercial off-the-shelf aircraft for Air Force use, and provides guidance concerning circumstances under which certification to civil standards is acceptable. According to AFR 80-36, "Transport aircraft must be designed to comply with civil airworthiness standards when their use is generally consistent with civil operations." (Emphasis added). This policy allows the USAF to lower development costs by taking advantage of existing civil certification testing, and lower production and operating costs by taking advantage of existing civil production and logistics programs and facilities. This policy also facilitates greater interchangeability of USAF and civil transport aircraft to gain maximum airlift and flexibility in emergencies, and improves the ability to dispose of surplus aircraft.

Because civil and military standards and practices differ - in some cases, significantly - the Air Force has experienced mixed results in operating commercial off-the-shelf aircraft. While generally successful, the experiences have not been painless, and many lessons have been learned concerning how "generally consistent" civil and military operations should be before allowing the use of commercial standards. This paper does not presume to state Air Force policy, except where noted, but instead will simply discuss civil and military flying qualities requirements, their applicability to military missions, and will relate the authors' experience in dealing with the various requirements on several recent Air Force programs. From this, the authors will try to derive lessons applicable to flying qualities specifications for future off-the-shelf procurements.

Flying Qualities Specifications:

Civil aircraft certificated by the United States' Federal Aviation Administration (FAA) are done so to the standards of Federal Aviation Regulations, generally part 23 (FAR 23) or 25 (FAR 25, Reference 2); for the purposes of this discussion the two documents are similar. The former addresses primarily light aircraft (up to 5,682 kg) while the latter is directed toward larger, transport-type aircraft. These regulations cover the entire range of topics associated with aircraft airworthiness, including structures and strengt , flight envelope, performance, and stability and control. The current United States military flying qualities specification is Military Standard 1797A (MIL-STD-1797A, Reference 3). Unlike FAR 25, MIL-STD-1797A confines itself exclusi ely to flying qualities.

The stabi ity and control/flying qualities portion of FAR 25 is contained in some 21 paragraphs. These paragraphs address:

- Controllability and Maneuverability
- Longitudinal Control
- Directional and Lateral Control
- Minimum Control Speed
- Trim
- Stability
- Static Longitudinal Stability
- Demonstration of Static Longitudinal Stability
- Static Directional and Lateral Stability
- Dynamic Stability
- Stall Demonstration
- Stall Characteristics
- Stalls: Critical Engine Inoperative
- Stall Warning
- Longitudinal Stability and Control (Ground Handling)
- Directional Stability and Control (Ground Handling)
- Wind Velocities (for Taxiing)
- Spray Characteristics, Control, and Stability on Water
- Vibration and Buffeting
- High-Speed Characteristics
- Out-of-Trim Characteristics

The user of the Federal Aviation Regulations must understand from the outset that the FARs primarily address flight safety. They are intended to insure only that the aircraft is safe to put on the market. The suitability of the design for its intended mission(s) is not assured; rather, it is assumed the success or failure of the design will be determined by marketplace forces. That is, if the users (pilots, passengers, and operators) find serious operational fault(s) with the design, it will through lack of sales be a failure.

The requirements of FAR 25 specifically address takeoff, climb, level flight (cruise), descent, and landing, including turns required in performing those items. Each of the items listed is considered by MIL-STD-1797A to be a non-precision (Flight Phase Category B) or terminal (Flight Phase Category C) task. There are no requirements which address high-precision tasks (MIL-STD-1797A Flight Phase Category A), as it is not contemplated by FAR 25 that FAA-certificated aircraft will be required to perform such tasks.

The vast majority of the requirements of FAR 25 are subjective in nature. At flight conditions specified in the regulation, the characteristics of the airplane must be acceptable to the pilot. In practice, guidance is usually provided by FAA regional offices concerning the definition of "acceptable", and the final judgment of any characteristic is by experienced company and FAA test pilots. This judgment is binary - the characteristic is either acceptable or it is not - and levels of acceptability are not addressed. As noted by generations of flying qualities engineers, exactly what is "acceptable" is open for interpretation by engineers and pilots; FAA regulatory processes add variations in interpretation between regional offices to further confuse the issue.

Where objective requirements are given, they are usually in areas critical to flight safety. Specifically, control force limits, longitudinal static (speed) stability, minimum control speed characteristics, stall warning, crosswind taxi, takeoff, and landing wind velocities, and characteristics following a trim system runaway or failure are specified.

Tasks requiring precise control of attitude or flight path (alluded to above) generally lead to requirements on the dynamic characteristics of an aircraft and its stick feel and dynamics. Again, as civil air transport operations rarely require such control precision, no quantitative requirements are levied against these characteristics by FAR 25.

It is also implicit in the requirements of FAR 25 that the flying qualities of the aircraft are those of the unaugmented airframe. If stability augmentation is used, it is assumed that the "bare airframe" dynamics are not unsafe; they must be demonstrated in flight test. Degraded-mode operation is not explicitly addressed except for failure of stability augmentation and/or trim systems (and engine failures). To date, except for the Airbus Industrie A320 (Reference 4), the FAA has not had to address aircraft whose flying qualities are highly augmented or which are open-loop unstable, nor have they had to address flying qualities degradations following system or subsystem failures, or failures of integrated control or guidance modes.

Finally, the FAA through its full complement of regulations exercises a "cradle-to-grave" philosophy. A certificated aircraft is manufactured in accordance with approved drawings and procedures using approved materials, parts, and processes, and is maintained in accordance with regulations by FAA-approved repair stations. Any change in any of those requires a new or amended certification to be granted for the aircraft to remain certificated.

Since 1942, dedicated military flying qualities specifications have been levied against aircraft being procured by the United States Armed Forces. These requirements have evolved through the MIL-F-8785 series to the Military Standard 1797 (MIL-STD-1797) series. MIL-STD-1797A is presently the standard and handbook for flying qualities requirements of US military aircraft; a capsule summary of the document is presented by Leggett in Reference 5. It is approved for use by all departments and agencies of the US Department of Defense (DOD). As a joint service document, it can easily be applied by any US service branch to the procurement of "off-the-shelf" as well as new military aircraft designs. The standard is suitable for specifying flying qualities of fixed wing aircraft on the ground and in the air as well as piloted transatmospheric flight when flight depends on aerodynamic lift and/or air breathing propulsion systems.

MIL-STD-1797A was especially set up to provide a framework and guidance for specifying the flying qualities of military aircraft. The logical sequencing and ordering of the document allows for ease of application. MIL-STD-1797A is in a format referred to as "MIL-PRIME" by the USAF. In this format, the document consists of a 50-page (approximately) specification framework, referred to as the "standard", followed by three appendices. The most important (and by far the largest) appendix is Appendix A, an approximately 630-page document called the "handbook". This handbook repeats each requirement from the standard, and gives guidance and lessons learned for adding actual numerical requirements to the standard to form a specification unique to the subject procurement. This process is called "tailoring", and is explicitly allowed for - even required - by the MIL-PRIME format. Also, a definitions section includes all of the pertinent aeronautical terms. Consequently, it is not necessary to cross reference terms with other publications for clarity.

In the interest of brevity it is not possible to review all the areas addressed by MIL-STD-1797A; the reader is referred to Leggett (5) and Woodcock(6) for more details. However, in general terms, the document addresses:

- Loadings and Inertias
- Flight Envelopes
- Flight Phase Categories
- Aircraft Configuration and States
- Failure Modes and Effects on Flying Qualities
- Interpretation of Requirements
- Static and Dynamic Stability Requirements for
    - Pitch Axis
    - Flight Path (Normal) Axis
    - Speed (Longitudinal) Axis
    - Roll Axis
    - Yaw Axis
    - Side (Lateral) Axis
    - Combined Axes
- Flight at High Angle of Attack
- Atmospheric Disturbances
- Stick Force-Feel-Deflection Characteristics
- Pilot Induced Oscillations
- Trim Systems

The above list is not meant to be exhaustive, but is rather meant to illustrate the breadth and scope of the subjects addressed by the document. As mentioned earlier, the various tasks to be performed by the aircraft are addressed by the Flight Phase Categories. Another important distinction is that MIL-STD-1797A contains many objective requirements, and allows for levels of flying qualities in demonstrating compliance with these objective requirements (for a discussion of objective and subjective requirements, see Leggett and Black (7)). The intent of this is to allow graceful degradation of the flying qualities as the edges of the envelope are approached, but still require the best flying qualities in the portions of the envelope where the aircraft accomplishes its primary tasks.

The framework and guidance provided by MIL-STD-1797A are geared directly to the flying qualities of aircraft performing specific military missions; the requirements themselves reflect lessons learned in the flying qualities found necessary for the performance of those missions. In contrast, Federal Air Regulations apply more generally to the airworthiness of aircraft transiting from point A to point B. As noted earlier, AFR 80-36 states "Transport aircraft must be designed to comply with civil airworthiness standards when their use is generally consistent with civil operations." Obviously, missions such as low-level operations, formation flying, and aerial refueling do not conform with "...generally consistent with civil operations." However, AFR 80-36 recognizes this by continuing, "This does not preclude using military specifications and standards in designing an aircraft when necessary to make sure that the aircraft performs its military role properly under the intended operating conditions." MIL-STD-1797A does apply to the unique military missions mentioned as well as the less demanding air transportation mission. The same cannot be said for civil airworthiness standards.

A Comparison of Select MIL-STD-1797A and FAR 25 Requirements:

It is useful at this point to contrast several requirements between MIL-STD-1797A and FAR 25. This comparison is not meant to be exhaustive; rather, it is meant to give the reader a taste of the type of requirements contained in the two documents. We have chosen to examine the requirements on Longitudinal Static Stability, Flight Path Stability, Phugoid, Short-Period Mode, Dutch Roll Mode, Roll Performance, and Control Forces.

Longitudinal Static Stability is seen in the tendency for an aircraft to return to a trim airspeed if disturbed, and can further be interpreted as a relationship between stick force and airspeed when the pilot intentionally untrims the aircraft for whatever reason. It is particularly important in terminal operations such as takeoff, climbout, and approach and landing, or in operations at or near stall speed. FAR 25 paragraph 173 (FAR 25.173) places objective requirements on the free-return characteristics of the aircraft when intentionally accelerated or decelerated via the pilot's control stick, and places requirements on the slope of the control force to speed variation ratio. This is quite appropriate for aircraft which spend the majority of their flying hours in steady state (or nearly so) conditions. However, a large stick force variation with airspeed can be tiring during maneuvering; pilots generally prefer light control force - airspeed gradients for aircraft which are required to maneuver even moderately as part of their missions. MIL-STD-1797A reflects this preference when it simply requires a stable response for Levels 1 and 2 in Paragraph 4.4.1. In practice, it is not uncommon for designers of fly-by-wire aircraft to provide neutral speed stability at moderate to high speeds for maneuvering, and high speed stability in low-speed flight for good "feel". Paragraph 4.4.1 can be tailored to reflect this as desired.

MIL-STD-1797A Paragraph 4.3.1.2 places requirements on the flight path stability of the aircraft. This is in essence a requirement on the degree of "back-sidedness" which is allowable in approach and landing. For conventional field operations, an approach speed may usually be found which is a good compromise of flying qualities and performance considerations; long runway length does not require low approach speeds or steep approach angles, thus the characteristic addressed in this paragraph is usually not a consideration under those circumstances. This is reflected in the absence of a corresponding FAR 25 requirement. This characteristic is critical, however, for STOL, assault, or naval shipboard operations, thus the requirement is present in MIL-STD-1797A.

FAR 25 also places no explicit requirement on the characteristics of the Phugoid mode other than requiring general dynamic stability. MIL-STD-1797A Paragraph 4.2.1.1 places objective requirements on damping or time-to-double amplitude for any longitudinal oscillation having a period of longer than 15 seconds.

Short-Period characteristics have been found to exert a major influence on pilot opinion for precision control tasks. FAR 25.181(a) requires that any short period oscillation simply be "heavily damped", and places no requirements on frequency. By contrast, MIL-STD-1797A has extensive objective frequency, damping, and time delay requirements in section 4.2.1.2 which apply to this mode; they are too extensive to consider here.

The requirements of FAR 25.181(b) specify that the Dutch roll mode be "positively damped with controls free" and that the mode be "controllable". MIL-STD-1797A Paragraphs 4.1.11.7 and 4.6.1.1 levy objective frequency and damping requirements on this mode.

Roll performance requirements are found in FAR 25.147(c) - (e), and are meant to insure the aircraft's ability to make 20 degree banked turns with one or more engines out, and that the "roll response must allow normal maneuvers with all engines operating." MIL-STD-1797A has at least seven sections which address roll performance requirements (see Bise and Black, Reference 8) for various maneuvers, portions of the envelope, etc, the majority of which are objective requirements.

Likewise, control forces are objectively addressed only by FAR 25.143(c), but values are specified for both temporary and prolonged force application. MIL-STD-1797A has multiple objective requirements addressing control friction, breakout, and maximum forces, maneuvering force gradients, and simultaneous application of forces in multiple axes. Again brevity prevents our citing the actual requirements here.

The seven sets of requirements we have just examined are contrasted in Table 1. Examined as a whole, a consistent trend emerges.

First, most (but not all) of the objective and subjective requirements levied by FAR 25 fall in the Level 2 region of the MIL-STD-1797A requirements. This is consistent with the guidance of MIL-STD-1797A indicating Level 2 is "acceptable", albeit with increased workload or degraded mission effectiveness. However, the workload increase or mission degradation is probably minimal for non precision control tasks, which is again consistent with the intent of FAR 25. Further, Level 2 is also the lowest level at which safety is guaranteed, which is the entire intent of the FARs! Hoh (Reference 9) elaborates on this further. He shows that "a rough equivalence between Cooper-Harper (ratings) and the decision to certify has been shown to exist at the 5 to 6 level", with many FAA pilots placing the break point at about a 5. This is the center to lower portion of the Level 2 region, again consistent with our observation.

By contrast, the requirements of MIL-STD-1797A are most explicit for flight phases where precise control during maneuvering is required. Most of these flight phases are the domain of the military, and are required in the accomplishment of various military missions. This raises the question of the adequacy of civil (flying qualities)

Topic	FAR 25 Requirement	MIL-STD-1797A Requirement
Longitudinal Static Stability	Objective, on Free Return and Slope	"Stable" for Level 1, 2
Flight Path Stability	None	Objective, on Slope
Phugoid	None	Objective, on Damping or Time to Double Amplitude
Short Period	"Heavily Damped"	Objective, on Frequency, Damping, and Time Delay
Dutch Roll	"Positively Damped", "Controllable"	Objective, on Frequency and Damping
Roll Performance	Subjective	Objective, on Time Delay, Time Constant, Time to a Bank Angle, Cross Coupling; Subjective, on Departure Resistance
Control Forces	Objective, on Temporary and Prolonged Application	Objective, on Friction, Breakout, Maximum Forces, Force Gradients, and Simultaneous Application of Forces in Multiple Axes

TABLE 1

COMPARISON OF SELECTED REQUIREMENTS, FAR 25 VS MIL-STD-1797A

certification for aircraft performing these type missions. In the procurement of off-the-shelf aircraft for military missions, this question usually lands squarely in the laps of the flying qualities engineer and test pilot.

The decision of whether to levy FAR 25 or MIL-STD-1797A flying qualities requirements (or some combination of the two) is not clearly prescribed by AFR 80-36. Therefore, the procuring agency's flying qualities engineer must examine the intended use of the aircraft and exercise professional judgment. This judgment should be based on an understanding of civil and military flying qualities specifications (already discussed) and previous military experience with civil-certificated aircraft. The most logical approach for the latter is to look at previous aircraft procured "off-the-shelf" and base one's conclusions on their degree of success in accomplishing military mission(s). We shall now do this by considering several examples.

Examples of Military Use of Civil-Certificated Aircraft:

We will consider the Beech King Air 200 (military C-12), Boeing 707 (military C-18) and derivatives (military E-6, E-8), McDonnell-Douglas DC-10 (military KC-10), Learjet 35/36 (military C-21), and Beechjet 400 (military T-1A) in this section. The authors have collectively flown several of these aircraft, and have performed flying qualities analyses and specification compliance comparisons on most. As such, some of the observations following are based on personal experience.

The first example is the Beech C-12 series. This sturdy, twin turbo-prop is used extensively by the United States armed forces for a myriad of missions. A derivative of the Beech King Air model 200T, it has been an overwhelming success. Most of the modifications made to the aircraft have been to extend its range or loiter time, and make it more able to operate out of austere locations. However, it still basically travels from point to point. The flying qualities are basically "honest" and generally Level 1, albeit with slightly heavy stick forces for precision tasks. However, it was originally built as a civil executive transport and observation/utility aircraft and that is how it is used by the military. Thus, in this case the civil and military mission and usage generally coincide. From a flying qualities perspective, the C-12 represents a successful off-the-shelf acquisition.

The next aircraft we will consider is the Boeing 707 series. This aircraft definitely has longevity and has fulfilled various military missions worldwide ranging from air transport to air refueling (tanker and receiver) to electronic warfare. Though this aircraft has been a workhorse for the U. S. Armed Forces, it is not without problems. These problems arise from the aircraft being originally designed as a point to point passenger and cargo transport, as well as being a pioneer jetliner. Like its near twin the KC-135, it can be a handful during heavy gross weight takeoffs. Also, flying qualities in high crosswind situations on very slick runway surfaces are very tenuous, yet it must operate in this environment also.

The U. S. Navy operates the E-6A, a CFM56-powered derivative of the Boeing 707. This aircraft provides a continuous survivable communications link between the national command authorities and submerged fleet ballistic missile submarines. To accomplish this, the E-6A has two very long retractable trailing wire antennas, one of which can be extended up to approximately 8 km. To operate properly, these antennas must be positioned vertically; to accomplish this the aircraft flies a tight orbit at constant altitude as slowly as possible. According to Feuerstein (Reference 10), many of the difficulties identified during E-6 testing relate to the orbit maneuver. The maneuver itself is flown at between 30 and 50 degrees of bank, with partial flaps, at just above stall speed. Bank angle control of plus or minus one degree and airspeed control of plus or minus one knot is required, and these conditions must be sustained for relatively long periods of time. This is a precision control maneuver not anticipated by the original certification baseline, and Feuerstein infers a high workload in its performance.

A final modification we will consider is the E-8A Joint STARS aircraft, an aircraft designed to perform real-time targeting of ground-based threats. Quoting from Reference 11, "The major external difference is the addition of a long, 'canoe'-shaped radome along the centerline of the aircraft between the nose landing gear and the wing root leading edge (Figure 1). Any projected side or planform area added ahead of the center of gravity on an aircraft is destabilizing, and must be balanced by the vertical or horizontal tail. In the case of the E-8A, the radome causes an unstable break in directional stability at sideslip angles beyond which the vertical tail stalls (i.e., the airflow separates; see Figure 2).

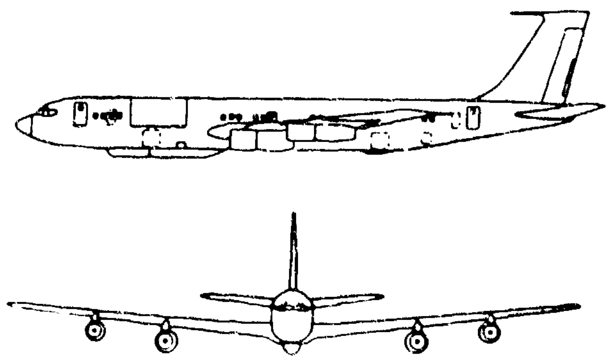

FIGURE 1

E-8A JOINT STARS

"While this could normally be countered by using opposing rudder to return the aircraft to lower sideslip angles, some other interesting characteristics exist. The limited wind-tunnel and flight-test data available indicate that for low angle of attack, high flap deflection conditions, rudder deflections exist for which the aircraft may diverge in sideslip angle, and opposing rudder may not halt the divergence (again, see Figure 2). The result of such a condition would be a departure from controlled flight, which could have serious consequences. The maneuvers during which this condition

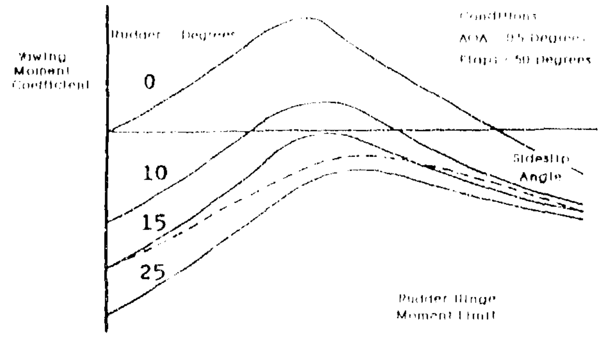

FIGURE 2

E-8A JOINT STARS DIRECTIONAL STABILITY

could arise are a crosswind takeoff or landing, or an engine-out condition at low airspeed. Simulation and flight test data indicate the latter does not exceed the critical sideslip angle, while the former may be dealt with by restricting the E-8A to lower crosswind limits than the basic B707. Should the user, however, find the lower limits overly restrictive, aerodynamic changes to the aircraft will be necessary."

Thus, though the Boeing 707 is a solid airframe and excellent for passenger and cargo transport, it has had a few significant problems when modified for the military mission. The E-8A represents perhaps the extreme case of a new mission requiring modifications to the aircraft which are extensive enough to seriously effect the flying qualities. (It is interesting to note that the aerodynamic effects of the modification are so extensive that the modified aircraft meets neither FAR 25 nor MIL-STD-1797A directional stability requirements at high sideslip angles.)

In contrast, the McDonnell Douglas KC-10 has performed well as an air refueling and transport platform. Flying qualities are exceptional and it is a versatile aircraft. The KC-10 was able to adapt to the military mission so readily because military specifications were taken into consideration during the original design of the aircraft. This fact has yielded benefits for both the commercial world and military sector. Again, from a flying qualities perspective this was a successful off-the-shelf acquisition.

The Learjet family has been used extensively for corporate transportation. These aircraft have been able to effectively and efficiently cruise from point to point at high altitude. It was a natural candidate for military VIP transport. It does the mission quite effectively; however, it does have some flying qualities problems in the region of slow speed, single-engine operations. There was not enough investigation in this area prior to acquisition which may have contributed to a fatal accident in a C-21 aircraft. Also, control harmony is notably poor with very heavy longitudinal maneuvering stick forces, and the cockpit is very cramped. This was a marginally successful off-the-shelf acquisition when viewed from a flying qualities perspective.

Another off-the-shelf corporate jet acquisition is the T-1A Jayhawk which is a derivative of the Beechjet 400A. This robust little aircraft is an excellent point to point transport and has good, predominately Level 1 flying qualities. It is a very stable platform with good control harmony but slightly heavy roll axis forces. Modifications are being made to the cockpit to adapt the aircraft to the training role. As a trainer, the aircraft will be required to perform numerous touch and go landings as well as operate routinely in the high-speed, low-level environment. Even though the Beechjet was not explicitly designed for this, its inherently good flying qualities make it adaptable to these missions.

Conclusions and Recommendations:

In summary, it is evident that in off-the-shelf acquisitions the military has had varying levels of flying qualities success. In those instances where there was success, the aircraft designers used the military flying qualities specifications as a guide, or the aircraft generally complies with those specifications. Alternately, civil flying qualities specifications have been adequate where the civil and military missions <u>and the conduct of those missions</u> have been nearly identical. In the not-so-successful cases, the military mission and the civilian usage were simply too diverse.

It is important to note the emphasized phrase above. In the authors' opinion, it is not enough that the civil and military missions coincide; the way the missions are flown must also coincide. For example, tactical airlift could be argued to parallel freight operations, yet civil freight operators are not required to operate in low-level terrain following or perform STOL operations from austere locations as a matter of course. This is almost exclusively a military-unique type of air transport, and must be treated as such by the invoking of military flying qualities specifications.

In cases where only certain elements of an aircraft's mission are military-unique, or require precision control, it is possible to use a blend of civil and military flying qualities requirements. The MIL-PRIME format of MIL-STD-1797A allows the document to be tailored to reflect this blending. This may be done, for example, by substituting civil requirements in appropriate paragraphs of a tailored MIL-STD-1797A where more demanding military requirements are not warranted. The resulting document then becomes a military specification unique to the subject application. The authors advocate this approach for off-the-shelf acquisition of aircraft where the military mission or mission conduct does not differ extensively from the civil, and the aircraft is not being greatly modified. In the case of substantial modifications and/or a uniquely different military mission, the authors suggest using the handbook-recommended values from MIL-STD-1797A, invoking full military flying qualities requirements.

Postscript:

Finally, though not central to the topic of this paper, an observation must be made regarding civil flying qualities requirements and the FAA certification process. As noted earlier, the Airbus Industrie A320 represents the first fly-by-wire civil air transport with highly integrated guidance and control modes. McElroy (4) described this aircraft as "a complete assault of technology on much of the U. S. regulatory flight criteria." (Reference 4). The authors agree that the criteria of FAR 25 do not envision such technology and are ill-prepared to address it, yet we have no doubt that the A320 represents only the beginning. The FAA must reassess its own criteria, regulatory, and certification processes to meet future fly-by-wire transports. The experience basis to do this is alive and well in the military services, and is preserved in the criteria, guidance, and lessons learned of the military specification system. This does not mean that the FAA should abandon its own criteria in favor of military specifications; rather, the FAA must be familiar with military specifications, and adopt their criteria or methods where necessary or advantageous. In like manner, the military services should be prepared to extend their knowledge and experience to the FAA when asked.

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Handling Qualities of Highly Augmented Unstable Aircraft

Summary of an AGARD-FMP Working Group Effort

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1. Preface

The flying characteristics and handling qualities of all types of aircraft are major items of interest in the activities of the AGARD Flight Mechanics Panel. A subcommittee of the Panel has specifically addressed this subject over a long period and initiated a questionnaire several years ago to determine the ongoing research, future plans and the need for additional activities in the area of aircraft handling qualities. Responses from interested organizations and institutions in the AGARD community indicated that the item "Handling Qualities of Unstable Highly Augmented Aircraft" showed the first priority. In response to this interest, the Panel formed a Working Group, WG-17, in 1987, consisting of specialists from all interested AGARD countries, to study this specific handling qualities subject. Six working sessions were held within the years of 1987 - 1990 the outcome of which is an AGARD-Advisory Report Nr. AR-279 which is to be published early 91. This report was a team effort and consists of contributions from all of the members of the working group. AGARD has been most fortunate in finding these competent people willing to contribute their knowledge and time in the preparation of this document. This paper only gives a short overview of the contents and tries to highlight the most important aspects and results.

2. Special Features of Highly Augmented Unstable Aircraft

Statically unstable aircraft are not new; for example the Wright Flyer was statically unstable and the pilot provided the control "augmentation". As knowledge of the balance between stability and control improved, aircraft were balanced stable to allow safe piloted control for demanding or protracted tasks. Today we again relax stability to produce configurations with substantially increased performance. With today's technology we now have the advantage of actuation, sensor and computing devices to augment, with full authority, the pilot's effort. Benefits of task-tailored handling, carefree handling and automatic functions and control modes outweigh penalties like larger actuators with high power consumption, high sensor performance, redundant controls and demanding computer speed and capacity requirements.

Handling Qualities of these highly augmented vehicles are largely the designer's choice; however, the effects of any increased flight control system complexity on handling qualities should be transparent to be pilot. It should not be necessary to distinguish between stable and unstable aircraft or even whether the aircraft is highly augmented, when specifying flying qualities. The stability of the basic design is immaterial to the pilot, who rightly expects low workload in an aircraft with full authority hardware and software.

Unlike the classic highly augmented aircraft, the handling qualities of the unstable highly augmented aircraft cannot degrade after failures to those of the basic aircraft. Instead, when failures occur the handling qualities do not change appreciably but the level of "protection" in the form of failure tolerance is reduced. For example, the X-29 technology demonstrator is highly unstable. With times to double amplitude in pitch of about 0.15 sec., it cannot be controlled by a pilot without augmentation. Following failures in its digital system, either the system logic or the pilot can select alternate redundant sensors or the analog reversion system, with virtually no flying qualities degradation.

3. Outline of the Working Groups AGARD-Report AR-279

The purpose of the report is to present methods and criteria which have been found to be useful by members of the working group as design guides and for the evaluation of handling qualities of highly augmented aircraft. It is the unanimous opinion of the members that no one method or criterion is adequate by itself, and that several, or even all of the recommended criteria should be checked. Experience has shown that one metric may not show a deficiency that will be exposed by other criteria. Alternatively, a configuration that passes several of the proposed criteria has a high probability of being accepted as desirable by most pilots.

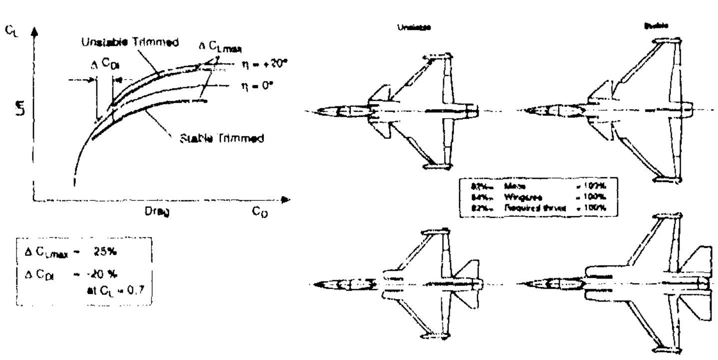
The report is organized in a series of major sections in which the major themes of this working group are presented followed by appendices in which important supporting information and other areas of interest to this working group are presented.

- A review of existing highly augmented aircraft (stable and unstable is given in Section 2.
- A unified method to match the shape of the response properly (i.e. type of augmentation) with the required mission tasks is presented in Section 3. This section also contains some guidance on the proper choice of criteria for different Response Types, discusses the influence of divided attention and presents a methodology for the minimum required stabilization according to visual environemental condition (outside world plus cockpit displays).
- Handling qualities criteria recommended by the working group members are contained in Sections 4 (longitudinal small amplitude) and 5 (longitudinal large amplitude).
- Considerations for the basic design of highly unstable airframes are presented in Section 6.
- There is growing evidence that feel systems must be treated as a separate entity, i.e., not as an integral part of the augmented airplane. This is covered in Section 7 along with the important issue of control sensitivity. It is important to note that none of the criteria in this report include the effect of control sensitivity, and that it must be separately optimized.
- Evaluation techniques utilized in simulation, both ground-based and in-flight, and flight test are discussed in Section 8. Guidance for the conduct of handling qualities evaluations, based on the collective experience of the working group, is presented in this section.

- o Important lessons learned from this handling qualities review and general guidelines for the conduct of handling qualities evaluations are presented in Section 9. In addition, a series of recommendations which represent the collective advice from this working group are presented to assist in the proper design and evaluation of future advanced highly augmented aircraft.
- o The conclusions and recommendation of the working group members are presented in Section 10. This summary includes recommendations on areas which are in need of additional research and testing.
- o An overview of the important subject of envelope limiting and carefree handling is presented in Appendix A.
- o Although the instabilities of interest are generally in the pitch axis, for completeness lateral-directional handling qualities are reviewed in Appendix B.
- o Since agility and handling qualities are closely related subjects with considerable overlap, this subject was of particular interest within the working group. In fact, it may be argued that the non-performance related aspects of agility are essentially handling qualities. This interesting subject is riefly discussed in Appendix C.
  One of outcome of this Working Group is the installation of a follow-on Working Group, dealing with this special subject.

The Report gets its final review during this symposium and will be published in the early 1991.

A few examples of the presented results:



AGARD Effect of Optimum Unstable Design on Aircraft Performance and Size

Advantages

- Task-tailoring of the Flying Qualities for each Mission Segment
- Higher Degree of Automation to augment the Pilot as Flight Envelope Protection and Carefree Handling
- No Degradation in F.Q., when Failures occure

Penalties

- Higher Effort for Actuator Power, Sensor Performance, Redundancy and Computer Speed and Capacity
- Complex Designs with High-order Responses (X-29 Pitch Control is 48th Order)
- New Problems created by Time Delays, Phase Lags and Actuator Saturation

AGARD Aspects of full-authority electronic flight control systems for unstable aircraft

Aircraft Type	Time to Double worst case	FCS Characteristics	Lessons learned Comments
X-29	0.15 sec	3 dig. channels + 3 analogue as back up	Feel System influenced the basic Pilot Ratings
FBW-Jaguar	0.25 sec	4 dig. channels, no back-up	Maneuver Limiting by combined q, α demand mode
EAP	0.16 sec	4 dig. channels, no back-up	Command path filtering to optimize piloted H.Q.
Rafale A	0.40 sec	3 dig. channels + 1 analogue back-up	Actuator Priority Management according to pilots inputs or atmospheric disturbances
Tornado	stable	3 analogue channels + 1 electrical + 1 mechanical link	PIO-Tendencies discovered during early flight test due to rate limits at large inputs
F-16C	1.5 sec	4 dig. channels	PIO problems at 1.flight due to misinterpretation of ground simulator results

AGARD **Examples of some reviewed existing highly augment aircraft**

- Early Designs of advanced electronic FCS exhibited significant Flying Qualities Problems.
- Even to-day not all Problems are totally understood (Grippen-accident)
- Different FCS-Design Philosophies are used for
  - Stable highly augmented aircraft:
    Improve "acceptable" Boundary (Level 2/3)
  - Unstable highly augmented aircraft: Achieve Level-1 Flying Qualities
- Level 2/3 - Characteristics still not be avoided for violent use of Fighters with "carefree" - Modes
- Complete math. Model of the total FCS including all Nonlinearities is a vital Requirement to discover Problem Areas
- Though the extensive use of Ground Simulators is essential for a proper FCS-Design, it is not suitable to tune the responsiveness of the aircraft.

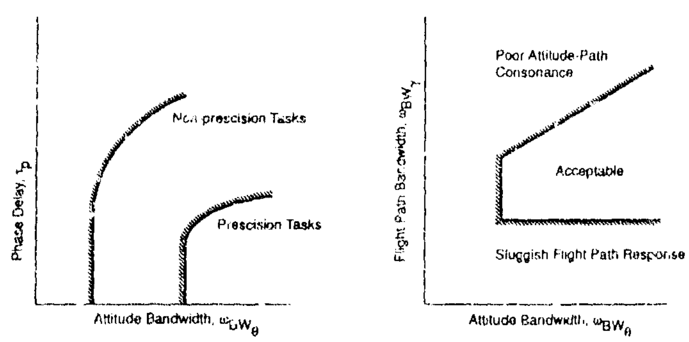
AGARD **Examples of "Lessons learned" by the review of existing FBW-aircraft**

- Short Description
- Rationale Behind Criterion
- Guidance for Application
- References

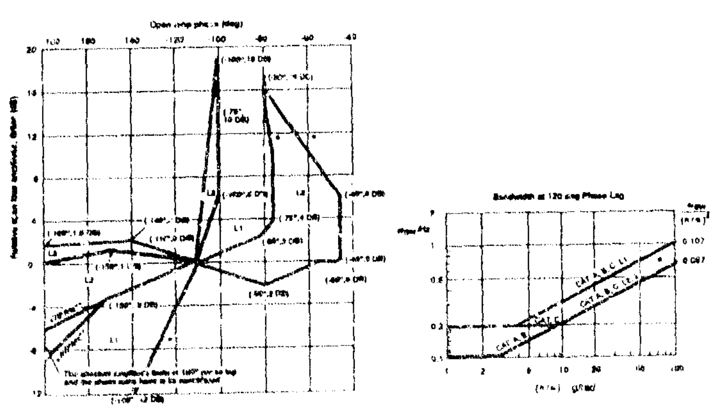
For the following Criteria:

- Low Order Equivalent System (LOES)
- Bandwidth Criterion
- Phase Rate Criterion
- Neal-Smith Criterion
- Frequency Domain Criterion
- Dropback Criterion

Longitudinal criteria for small amplitude precesion attitude and flight path control



AGARD **Generic Shape of Attitude and Flight Path Bandwidth Criteria**

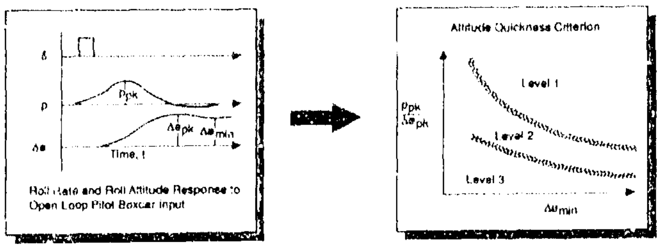


AGARD **Pitch Attitude Frequency Response Limits**

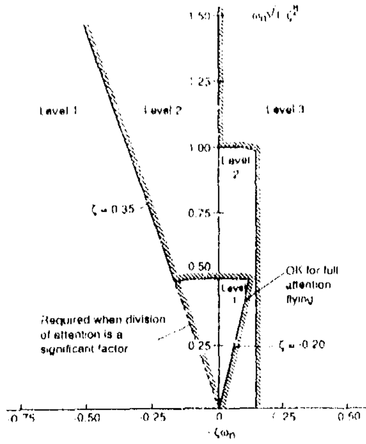
Based on Open Loop Boxcar Inputs of Varying Duration and Amplitude

Is Analogous to Bandwidth, Except it applies to Larger Amplitude Manoeuvres

Definition of Criterion Parameters, and expected Shape of Boundaries is shown below



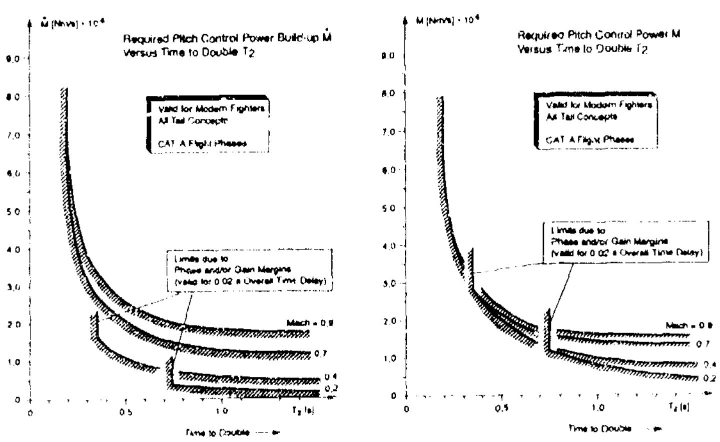
AGARD **Attitude Quickness Criterion as a Moderate Amplitude Requirement**



Limits on Pitch and Roll Oscillations as a Funktion of Required Pilot Division of Attention

- Too small Time Constant leads to excessive lateral Sensitivity and Roll Ratcheting
- In Tracking Tasks increased Roll Rates plus lateral Sensitivity cause PIO - Tendencies
- Frequency Response Method is a suitable Tool also for analysing the lateral/directional Behaviour
- Some Simulator Studies based on the AMX-Aircraft was done. Future Research is needed.

AGARD **Aspects of Lateral/Directional characteristics of highly augmented aircraft**



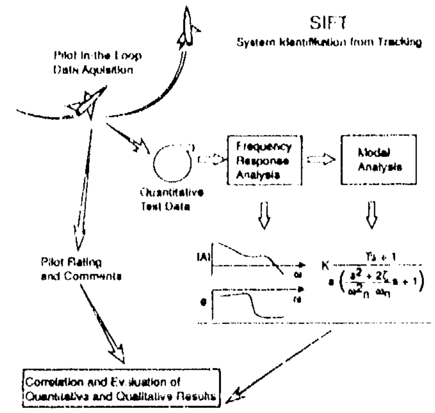
AGARD **Criteria for Pitch Control Power**

Objectives
- Reduction of pilot attention needed to control the aircraft especially in air combat situation
- Simplification of piloting for some mission phases by possible "bang bang"-control

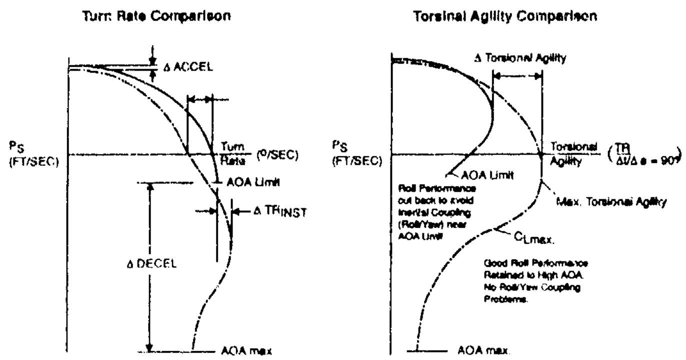
Features
- Angles of attack, Sideslip and Airspeed Limiting
- Load Factor, Load Factor Rate, max. Roll Rate Limiting
- Engine Limitation
- Weapon Delivery Conditions Limitations

Problems
- Proper Design according to the Vehicles Characteristics, Mission Aspects and Human Perception is difficult to achieve
- Loss of Consciousness, Reduction of Agility, Overconfidence of the Pilot are problems been found during Design and Flight Test Phases
- The Integration of these Systems into the basic FCS is not yet always fully understood

AGARD **Envelope limiting and carefree handling**



Schematic Outline of the SHIFT Pilot-in-the-Loop Handling Qualities Test Techniques



Proposed Metric for Torsional Agility

Definitions of the operational payoff of functional agility.

Definitions of agility measures of merit to be used for requirements design and evaluation.

Definition of specialized flight test techniques for agility.

Exp... at... n and documentation of the theoretical foundations.

Documentation of practical lessons learned in design or test programs.

Specialized aspects of the agility of rotary-wing aircraft.

Definition of the needs for future activities, including identification of possible cooperative projects.

Terms of Reference for WG 19 on "Functional Agility"

- WG 17 fulfilled a challenging task with outstanding engagement, know how and interesting results.
- The produced AGARD-Report AR 279 will represent the combined knowledge of the relevant specialists in the AGARD Community.
- Sufficient knowledge is available for the design of the longitudinal FCS and the use of adequate design criteria.
- Uncertainties are still in the area of combined axis maneuvering, divided attention, the integral approach including display technologies and the integration of the Feel System dynamics.
- Further research is needed on the above cited uncertainty areas, for the lateral/directional motion, carefree handling and the integration of agility.

Conclusion and Recommendations

THE HANDLING QUALITIES OF THE STOL & MANEUVER TECHNOLOGY DEMONSTRATOR FROM SPECIFICATION TO FLIGHT TEST

by

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INTRODUCTION

The USAF contracted with McDonnell Aircraft Company in 1984 for the development of the STOL and Maneuver Technology Demonstrator (S/MTD). The S/MTD program was structured to investigate, develop and validate through analysis, experiment and flight test, four specific technologies:

- o Two-dimensional thrust vectoring and reversing exhaust nozzles
- o Integrated Flight/Propulsion Control (IFPC) System
- o Advanced Pilot/Vehicle Interface
- o Rough/soft field landing gear.

These technologies, together with all-moving canard surfaces, have been incorporated into an F-15B (see Figure 1) with the overall objective of providing current and future high performance fighters both STOL capability and enhanced combat mission performance. Level 1 handling qualities were required across the full envelope from subsonic and supersonic maneuvering through precision approach and touchdown to ground handling on a wet runway with crosswind. Various control modes were also required in order to demonstrate and evaluate the technologies. Many hours of piloted simulation were performed to verify the analytical control laws. The aircraft has been flying a phased test program since September 1988, with handling qualities results that agree very well with the piloted simulations. The general development of the control system configuration is presented in some detail in Reference 1. The present paper documents the development of the handling qualities of the S/MTD aircraft from the original specification through the analytical development and simulator verification to the flight test results.

HANDLING QUALITIES SPECIFICATION

A new version of the Military Flying Qualities Specification, MIL-F-8785C, had been published in 1980. Major revisions incorporated in this version were expected to be applicable to the S/MTD development, e.g. equivalent system representation of highly-augmented system dynamics, low altitude disturbance, etc. At the same time, however, the contract was written to allow for deviations from individual requirements if an improvement could be substantiated. As far as possible for the overall design, required performance was specified rather than design approaches or solutions. Some performance requirements also translate into implicit handling qualities requirements. The required 1500 ft landing distance implies a requirement to minimize touchdown dispersion, i.e., "very good" handling qualities are needed to control the flight path and sink rate through the approach to touch down. Similarly, the 50 ft width of the landing strip plus a 30 kt crosswind formed the flaps down lateral/directional handling qualities requirements. The handling qualities task developed from these requirements was to touch down in a box 60 ft long by 20 ft wide at the threshold of the operating strip. A stringent ground handling task was produced by specifying a runway surface friction coefficient corresponding to wet bordering on icy conditions, while using reverse thrust to stop with crosswinds.

The preceding discussion reflects implied handling qualities requirements, whereas more direct ones were as follows. The overriding requirement of the IFPC system was stated to be "capable of functionally integrating all aspects of flight, engine, nozzle control including aerodynamic control surfaces, engine thrust, thrust vectoring, thrust reversing and differential efflux modulation, control and stability augmentation, high lift system, steering and braking". The intent was to convey the understanding that integration was an objective of the demonstration program, not just a means to achieve mission requirements. The IFPC system was required to provide "good inner-loop stability and positive manual control in all axes of the air vehicle throughout its intended operating envelope both in flight and on the ground (satisfying the intent of MIL-F-8785C)". This subjective requirement was intended to convey that we were seeking good flying qualities over the whole envelope guided more by the intent than the letter of the specification. This recognizes that, while the intent is to provide flying qualities clearly adequate for the mission, the letter of the specification is no guarantee. In addition, the requirement for 'positive manual control' was intended to preclude consideration of automatic landing systems, for instance. One flying qualities requirement that was explicitly called out in the Statement of Work was to minimize time delay, i.e., lag in aircraft response to pilot control input. Although the importance of time delay is more widely accepted now, it still should be an explicit, hard requirement in any control system to be designed for any precise task.

Specific flight control modes were required with the rationale: "In order to provide the ability to assess task performance and minimize pilot workload in the flight vehicle, the integrated system shall also provide the flexibility to permit inflight selection of mission task oriented control modes as determined by analysis and simulation. Mode switching transients shall not produce unsafe aircraft responses. As a minimum, the following modes are required:

A CONVENTIONAL mode shall be designed for satisfactory performance over the flight test envelope, including conventional landing, without the use of the added technologies. This mode will serve as a baseline for performance evaluation and as a backup in the event of multiple failure of the new technology components.

A STOL mode shall be designed to provide precise manual control of flight path trajectory, airspeed and aircraft attitudes. The integrated control system and other technologies shall be combined to provide short field performance in weather and poor visibility. The purpose of this mode is to minimize pilot workload during precise manual landings, high reverse thrust ground operations and maximum performance takeoffs.

A CRUISE mode shall be designed to enhance normal up-and-away and cruise task performance, with and without external stores. The purpose of this mode is to use the integrated control system and other technologies to optimize appropriate measures of merit representing an improvement over the cruise capability of the baseline aircraft.

A COMBAT mode shall be designed to enhance up-and-away maneuverability, with and without external stores. The purpose of this mode is to use the integrated control system and other technologies to optimize appropriate measures of merit representing an improvement over the combat maneuvering or weapon delivery performance of the baseline aircraft".

During the development process, these required modes evolved into the list shown in Figure 2 and described further throughout the paper. The S/MTD control law development was done analytically but, as with any other, was supported by extensive piloted simulation. A fixed-base facility at McDonnell Aircraft and a motion-base one at Wright Research & Development Center were both used - with identical modelling. The simulation results have proven to agree with flight test results to be discussed.

HANDLING QUALITIES DEVELOPMENT

All the bidders on the S/MTD contract were strongly encouraged to use multivariable control theory, although it was not expressed as an absolute requirement. With integration as a program objective, there was some uncertainty that a classical approach would optimize use of all the available effectors. At the same time, there was no desire to commit to a totally multivariable design approach. The S/MTD control laws were designed using a combination of classical techniques by McAir and multivariable theory by Honeywell (see e.g. Reference 2). One of the results of this development is that problems with the flying qualities specification are independent of the design methodology. Both classical and multivariable approaches first require the definition of the optimum transfer function to use as the design requirement. Figure 3 shows the final implementations that are being flown in the aircraft.

Longitudinal Axes

The obvious emphasis in developing the up-and-away modes was for pitch tracking. The initial design represented a Level 1 configuration by MIL-F-8785C requirements, but received Level 2 comments and pilot ratings for a tracking task in piloted simulation. The development to the final Level 1 configuration (References 3 and 4) also led to a natural separation of requirements between the CONVENTIONAL/COMBAT mode tracking characteristics and the CRUISE mode flight path control. The method chosen to improve the original design (i.e. increase the pitch attitude bandwidth) was to increase the effective numerator time constant to a higher frequency, and augment short period frequency to maintain the chosen value of ω_n^2 / n_α (Figure 4a). The simulation results were favorable pilot comments and ratings for the tracking task. At the same time, the short term response of normal acceleration to pitch rate differed from the classical long term response. This gives the characteristics shown in Figure 4b. The decision was made to implement the improved tracking characteristics in the CONVENTIONAL and COMBAT modes, but to leave the CRUISE mode with Level 2 tracking characteristics and a precise load factor (flight path) response. This yielded the opportunity of flying the different characteristics in back-to-back tests.

A significant development effort went into achieving precision flaps-down flying qualities to support the landing distance requirement. In the Statement of Work, landing was defined as a Category A tracking task in the notation of MIL-F-8785C. This had the practical effect of increasing the minimum allowable short period frequency, ie., increasing minimum pitch attitude bandwidth. Even this increased requirement was less stringent than bandwidth requirements proposed as alternate criteria and now included in MIL-STD-1797. It did establish that the intent was to develop a maneuverable and controllable configuration, including the effects of crosswinds, gusts, turbulence etc. This applied to the CONVENTIONAL and STOLTOA modes, but most emphasis was placed on the SLAND mode.

The short landing distances are facilitated by providing maximum reverse thrust immediately after touching down. To achieve this, the final approach is made with the engine spooled up to 100% RPM and the exhaust passing through vanes which are controllable from 45 degrees aft to 45 degrees forward of normal. These fast-acting vanes also provide for high-bandwidth control of airspeed because there is no influence of engine spool-up time. Design of the SLAND longitudinal control laws (pitch and thrust axes) was accomplished using multivariable control techniques. The complete design requirements included a specified form of the pitch and airspeed responses. There are also requirements to decouple the two axes, so that there is neither airspeed response to a stick input nor pitch rate response to throttle input. In the final form of the control laws, two features are to be noted -- decoupling is achieved by sending both pitch and thrust commands to the upper and lower reverser vanes, and the form of the response in each axis is defined by a command prefilter. Now, in terms of pilot action, the feedback of airspeed to vane angle holds airspeed constant until the throttle is moved. The pilot retains control because throttle movement commands a new airspeed and consistent flightpath change. The stick commands pitch rate directly but, with speed held constant, it is effectively a flightpath rate command. On approach, the pilot sets airspeed with throttle and, once the correct airspeed is acquired, flightpath is controlled solely with stick input.

This development highlighted the lack of criteria for a precision touchdown (see also References 4 - 6). Figure 5 (and References 4 & 5) presents results from a moving-base simulation which show that the alternate bandwidth required is too stringent with speed hold and not stringent enough without it, using the S/MTD touchdown dispersion goal. Reference 6 presents results from a fixed-base simulation in which generally lower values of bandwidth were satisfactory.

Motion cues may explain some or all of the differences. We conclude, however, that more research is needed to define landing requirements for pitch as a function of both the speed stability and the required touchdown dispersion.

Lateral & Directional Axes

Initially, the lateral directional requirements were believed to be a conventional application of MIL-F-8785C. No multivariable design or analysis was attempted for these axes, the lateral-directional control laws were developed using classical design methods. The lateral control laws utilize a conventional roll rate feedback path, in addition to a limited roll rate feedback for gust rejection. The directional control laws incorporate N_y and estimated feedback paths. In addition, interconnects from the lateral control surface commands to the directional controls are used for roll coordination. The gains were defined using design guidelines established early in the development phase. The goal for the Dutch roll damping was 0.7, the Dutch roll frequency was designed to emulate the F-15, and the roll mode time constant was 0.3 seconds above 180 knots.

Equivalent system analyses indicated that the Dutch roll and roll mode characteristics were close to those desired. In addition, the equivalent time delays in the lateral axis were between 0.055 and 0.060 seconds. Sideslip excursions to lateral stick commands were very small, and the roll rate oscillations (P_{OSC}/P_{AV}) were zero. All of the handling qualities parameters were within the MIL-F-8785C Level 1 requirements.

Results from the McAir manned flight simulator indicated that the large amplitude, open loop response was Level 1. The pilots considered the roll response crisp, with excellent roll capture characteristics. However, tracking a 3G or 5G reversing target was Level 2. When moving the pipper from one wing tip to another, it was noted that there was no initial response to a lateral stick input. Then, about the time the pilot would typically begin to add more stick, the pipper would rapidly move across the target, resulting in an overshoot. The difficulty this caused resulted in Cooper Harper ratings consistently between four and six. The tracking appeared to degrade with increasing airspeed, with the worst flight condition evaluated being at Mach 0.9, 10,000 feet. Load factor had little effect on the tracking.

The tracking evaluation was repeated on the LAMARS facility at WPAFB with the same results. The presence or absence of motion was not a factor. A matrix of control law point designs was then tested at selected flight conditions. Based on the results, a shaping prefilter was added to the crossfeed path of differential stabilator to rudder and implemented in subsequent simulation testing. The result was Level 1 flying qualities in tracking, as well as open loop response, at all flight conditions evaluated.

The control law change had minor effects on the MIL-Spec flying qualities parameters. Since there was no obvious reason for the improvement in tracking, further analyses were undertaken in an attempt to identify the parameters important in this task and to understand the dynamics involved. To accomplish this, time histories of the lateral control response were generated in a 3g banked turn. The roll rate, yaw rate and sideslip angle responses with the original and the revised control laws are compared in Figure 6. The sideslip is very small in either case, the pipper excursions during tracking far exceeded those caused by the sideslip. The roll rate response was essentially identical in each case. The biggest difference was in yaw rate. While the magnitude was the same in each case, the lag between the yaw rate and the roll rate was noticeably smaller with the revised system.

The next step taken was to obtain a better indication of the dynamic characteristics of the pipper aim point on the target aircraft. The target was assumed to hold the initial load factor throughout the run. A range of 1,500 feet was used. Time histories of the lateral variation of the aim point on the target aircraft as a result of lateral control inputs were obtained. The results using the original control laws confirmed the pilot observations. As shown in Figure 7, little motion in the target aim point is evident for at least a half a second following a lateral stick input. Then the lateral aim point begins to move very rapidly. The control law revision was effective in both decreasing the initial time delay observed in the aim point response, and in moderating the rate at which it traversed the target aircraft.

It was further observed that the difference in the dynamics of the target aim point is related to the lag in the yaw rate described earlier. Because the relative phasing of the roll and yaw rates appeared to be important, a frequency response of the yaw rate/roll rate phase angle was generated. The results presented in Figure 8 indicate large differences in the frequency responses of the original and revised control laws. Above a frequency of about 2 radians/second the phase lag with the original control laws begins to increase, while the revised control law phase lag actually decreases somewhat. Above 10 radians/seconds the differences in the phase lag become even more pronounced. Hence significant differences in dynamic characteristics of the two systems exist even though the conventional flying qualities parameters are similar.

Flight test results have confirmed the simulator findings. Pilot ratings obtained in air to air tracking tasks were in the Level 1 category, with the exception of one point which received a rating of 4. It was found that an offset in the corrected AoA used by the flight controller resulted in a larger phase angle between yaw rate and roll rate at this condition than was predicted. A software change corrected the AoA offset and improved the phase angle. A repeat of this test point received a pilot rating of 2. This data provided a variation of pilot rating with phase angle that is presented in Figure 9. A frequency of 6.0 rad/sec was used, since it was found to be typical of pilot inputs during the tracking task. Simulator results for the CONVENTIONAL mode are also shown. The data available are inadequate to establish a firm specification but do indicate that this phase angle is a candidate for a second-tier criterion to provide guidance in designing the lateral/directional axes to have Level 1 tracking characteristics.

The development of the flaps-down lateral and directional axes would have been conventional except for the heavy emphasis on crosswind landing and also on an unconventional control capability of the configuration. Differential canard deflection can be combined with rudder deflection to yield direct sideforce control. During the approach, the direct sideforce is commanded by the rudder pedals based on the assumption that the pedals are only used for a crosswind landing. The implementation still allows the pilot the choice of a crabbed or slipped technique. The appearance, however, is of a crosswind approximately 50% of the true value. After touchdown, command of direct sideforce is transferred to lateral stick. This again gives the pilot a natural technique of putting "stick into the wind" during the rollout.

Ground Handling Mode

Development of the ground handling mode was also an intense effort. There were very strong interactions between the jet interference effects and ground effects (Reference 1), the most adverse being a nose-up pitching moment at forward deflections of the lower reverser vanes. Special logic was needed to ensure that the aircraft could achieve a 3-point attitude following touchdown. The forward deflection of the lower vanes is scheduled with pitch attitude to minimize pitching moment, maximum reverse thrust being achieved as 3-point attitude is reached. Weight-on-wheels indication introduces a nose-down control input, raises the flaps and drooped ailerons, cancels the lateral/directional interconnects and commands maximum reverse thrust and braking if selected by the pilot. The requirements to optimize both wet- and dry-runway stopping produce conflicting effects. Figure 10 shows calculated ground tracks for both conditions in gusting 30 kt crosswinds. Ground track stability was enhanced by the addition of yaw rate feedback to nosewheel steering. Finally, controllability is assured by commanding direct sideforce with lateral stick and yaw rate with rudder pedals. Figure 11 shows the predicted control activity corresponding to the two runs in Figure 10. Wet runway and crosswind testing have not been done, so it is appropriate to present simulation results here (summarized from Reference 7).

With no crosswind, the rollout was no problem, with the runway icy or dry. The pilots achieved desired performance most of the time and no deficiencies were noted. The pilots rated it solid Level 1, all Cooper-Harper 2's.

With crosswinds the pilots encountered some handling qualities problems depending on the runway condition. On a dry runway there was a strong roll transient right at touchdown, apparently caused by the reaction of whichever gear struck the runway first. Since a crosswind approach is usually made with the upwind wing low, it was typically the upwind gear that hit first and thus the roll was almost always away from the wind. This made the roll tendency even worse since the crosswind then increased the rolling tendency. The higher the crosswind the greater the rolling tendency. The second problem was a tendency for the aircraft to run in the direction the nose was pointing at touchdown instead of skidding in the direction of the flight path and the nose coming around. With a 15 knot crosswind, the touchdown transient and the initial direction change were not difficult to control. The pilots usually were able to counter the direction change at touchdown and keep the airplane within desired criteria (within 12.5 feet of centerline). The task was rated a Cooper-Harper 3. With the 20 and 25 knot crosswinds, the pilots had more difficulty controlling the initial transients. The initial direction change sometimes carried them outside of the desired criteria before they could counter it. The higher the wind, the more frequently they failed to contain the direction change within desired criteria. Once desired criteria was achieved it was easily maintained, pilots rated these cases borderline Level 1/Level 2. Ratings for the 20 knot crosswind cases ranged from 3 to 5, mostly Level 2. Ratings for the 25 knot case ranged from 3 to 6, mostly Level 2. With a 30 knot crosswind the pilots were rarely able to contain the initial transient to desired criteria. Again, once desired criteria was achieved it was easily maintained, Cooper-Harper ratings were solid Level 2, ranging from 4 to 6. In all cases, the Level 2 ratings were due entirely to difficulties keeping the initial transient within desired criteria. Once recovered from the initial transient, control of heading was no problem.

On icy runways, the reduced runway friction allowed the aircraft to skid in the direction of the flight path and also seemed to reduce the magnitude of the gear reaction so the initial touchdown transients were not a problem. However, a bigger problem was a dendency to get into a slide downwind that could not be stopped before the aircraft slid off the MOS. Thrust reversing on the runway gave the aircraft a tendency to yaw away from the relative wind during the rollout. The pilot would put in rudder to correct this and the nose would come around, but the aircraft would continue to slide downwind. The slide could not be stopped until the velocity got slow enough to get effective nose wheel steering.

With a 15 knot crosswind, the yawing and sliding tendency could be controlled well enough with direct sideforce control that the pilots didn't need to come out of reverser to keep the airplane within desired criteria, the ratings for these cases were Level 1. In the 20 knot crosswind cases the pilots noticed the sliding tendency more but were still able to control it. The pilots rated this borderline Level 1/Level 2 with Cooper-Harper ratings from 2 to 4, mostly Level 1. In the 25 knot crosswinds the pilots could not stop the slide before getting out of desired criteria. If they came out of reverser, the aircraft was controlled before sliding off of the MOS. If they did not come out of reverser, the aircraft could not consistently stay on the MOS. When the pilots used the technique of coming out of reverser, rollout task ratings were borderline Level 1/Level 2. When they did not come out of reverser pilot ratings were Level 2, 5's and 6's. With a 30 knot crosswind, they had to come out of reverser to stay on the MOS, pilot ratings were Level 2, 6's and 7's.

All the pilot ratings for this task are presented in Figure 12. Also shown is the "allowable" variation of rating with increasing disturbance intensity, given by Section 3.8 of MIL-F-8785C. Thus, the pilot ratings of 4-6 are satisfactory.

FLIGHT TEST RESULTS

The flight test program contains three distinct phases. An initial phase used axisymmetrical (production) exhaust nozzles to perform systems validation, flutter and aeroservoelasticity clearances, and an evaluation of the CONVENTIONAL mode. This phase began in September 1988. Two-dimensional nozzle testing with vector capability began in May 1989. Initial enhanced mode and vectoring evaluations were performed. Finally, testing of the 2-D nozzle with full capability commenced in March 1990 and is scheduled to continue through April 1991. The test program is a full-envelope evaluation of all the technologies, e.g. a list of test milestones is presented in Figure 13. Reference 1 presented some early results of the axisymmetric configuration including the minor software fixes that were required. References 8-10 present the pilots' opinions of the aircraft. The simplest and most direct form of presentation of handling qualities flight-test results would be tables or graphs of Cooper-Harper ratings. In fact, the table of ratings shown in Figure 14 has been published. There is a lot more information behind the simple numbers. Here, we present the flying qualities results mostly in the form of pilot comments. These comments (with ratings) show excellent agreement with the analytical and simulation development results. They also provide an interesting diversion in comparing different likes of the different pilots, which would be masked by a simple presentation of Cooper-Harper ratings.

Handling Qualities During Tracking

Mach 0.6/10K:	Gross	Fine
CONV HQR	2	2
COMBAT HQR	3	3

- "Combat received HQR 3 or 3.5 because of increased pitch bobble." (Pilot A)
- "Lateral accelerations at the cockpit in COMBAT mode are much higher. (Pilot B)
- "CONV stopped when put it there." (Pilot A)
- "COMBAT not as predictable, tracks better CONV." (Pilot A)
- "COMBAT HQDT Improves with G increases." (Pilot A)

Mach 0.7/20K:	Gross	Fine
CONV HQR	3	3
CRUISE HQR	4	4
COMBAT HQR	3	3

- "Slight pitch sensitivity in CONV." (Pilot C)
- "Tendency toward small directional overshoots in CRUISE with pitch sensitivity equivalent to CONV." (Pilot C)
- "Very nice pitch control in COMBAT: less pitch sensitive than CONV. This was the best mode of all for tracking." (Pilot C)
- "COMBAT was much less PIC prone." (Pilot C)

Mach 0.9/20K	Gross	Fine
CONV HQR	3	2
CRUISE HQR	4	3
COMBAT HQR	3	3

- "HQDT in all three modes was very nice. (Pilot B)
- "The biggest delineator for all tasks in the three modes was the relatively poor gross acquisition in CRUISE. The initial overshoot was large, 25+ mil, but I usually only had one overshoot in CRUISE. In COMBAT and CONV I generally had a much smaller overshoot, better predictability, but often more than one overshoot. One small difference was noted in that the COMBAT mode seemed to have less pitch acceleration in the gross acquisition task than CONV mode - to get the same performance I had to pull harder and this was evident as apparent higher stick forces." (Pilot B)
- "Fine tracking was really nice in all three modes. In CRUISE I did have more directional errors than in the other two modes." (Pilot B)
- "COMBAT gross acquisition is better than CONV. A little more wandering in the fine tracking though." (Pilot B)
- "Big overshoot in CRUISE mode." (Pilot B)

Mach 0.9/30K:	Gross	Fine
CONV HQR	3	2
CRUISE HQR	4	3
COMBAT HQR	3	2

- "CONV-solid, easy, well damped, delightful." (Pilot A)
- "CRUISE - poorly damped, difficult acquisition, fair tracking but bobble prone." (Pilot A)
- "COMBAT - bobble prone, not as predictable as CONV but better than CRUISE." (Pilot A)

Note that all of the ratings are Level 1 for the CONVENTIONAL and COMBAT modes, and Level 1/borderline Level 2 for the CRUISE mode. Very little discrimination would be implied by the ratings alone. The comments reflect perceived differences and preferences even when the ratings are identical. Thus Pilot B says that COMBAT gross acquisition is better than CONV, but assigns a rating of 3 to both. Pilot C says that COMBAT was the best mode of all for tracking, but assigns the same rating as the CONV. Also, it can be seen that Pilot A consistently prefers the CONV mode while Pilot C chooses the COMBAT mode. No explanation will be attempted to explain these results. The "engineering evaluation" of the different pilots is that both pilots A and B are high gain relative to pilot C. We can certainly rationalize their preferences on this basis, and it supports the technique of requiring multiple pilot opinions. In a development program, however, do we have to satisfy all pilots? If not, whose opinion is given precedence? In the S/MTD program we are lucky - the differences are within the Level 1 range and the distinction is academic. This aircraft is unique in having the capability to switch modes at will, and fly the modes back-to-back. Fine distinction within Level 1 characteristics is not normally a problem in a development program. What is important is that these comments do indeed repeat the comments that were noted during the piloted simulation program.

Landing Configuration

10K/10.5° AOA:

- Pitch Captures - "Nice and stable" (Pilot B)
- Flight Path Captures - "Sluggish just like normal" (Pilot B)
- "Everything was fine. I didn't see anything I didn't like." (Pilot B)

10K/12° AOA:

- Pitch Captures - "Slight bobble, not bad." (Pilot C)
- Flight Path Captures - "Not bad" (Pilot C)
- Heading Captures - "Slight under shoot." (Pilot C)
- Bank Angle Captures - "Slight over shoot, not bad" (Pilot C)
- "I think it flies pretty nicely, wallowing on heading capture a little." (Pilot C)

- "Pitch axis is fine, harmony between axes is good" (Pilot C)
- "Easy to control flight path" (Pilot C)
- "45° bank to bank rolls using full lateral stick produce a sloppy feel, comparable to other aircraft at this condition and not unique to the S/MTD. Overall SLAND felt good in all axes. (Pilot C)
- Formation Handling Qualities in SLAND mode showed no PIO tendencies. Response was somewhat sluggish but was satisfactory for this flight condition." (Pilot A)
- "Pitch capture in SLAND mode had some bobble" (Pilot A)
- "Velocity vector capture was easy and felt good for a landing mode" (Pilot A)

The above comments can be augmented with the following quotes taken from Reference 10: "The initial SLAND test was at 10,000 ft at 165 kt. Engine operation was flawless and the pitching moment transient was negligible. Flight path angle had to be placed into a mild descent to avoid limiting the vanes at their maximum push position of 45 deg, but once stabilized, the speed hold feature worked well. Pitch control power was fine with good controllability. Lateral-directional feel was slightly heavier, but bank angle capture remained good as did directional control power. The Dutch roll was also well damped. No PIO tendencies were apparent.

After an early morning takeoff, the first SLANDing was planned for idle power at touchdown to assess ground effects, pitchdown discrete, and lift dumping. Results showed that ground effects caused a very slight nose rise prior to touchdown, but nothing which would change pilot technique.

The second SLAND approach, planned for a slow pull to full reverse, was flown slightly slower, on-speed, with a 3 deg glideslope. Autobrake was armed to provide full wheel braking. At touchdown, the pilot pulled to full reverse. The SMTD derotated, dumped lift, reversed, and braked 'It was almost like catching an arresting cable.' Handling qualities were good - no lateral directional inputs were needed, nor pitch inputs. Deceleration was quite good. Actual landing distance was 1,707 ft, in spite of the heavier weight and Edwards' altitude and temperature. If corrected to sea level standard day and demonstration gross weight, the distance would have been in the fifteen hundreds".

- "Spot landings in both CONVENTIONAL and STOL mode were difficult. I could not see any difference in precision between the two modes".

These comments are consistent with the piloted simulation results, i.e. with no speed hold in either mode and "normal" flaps down flying qualities. The results will form a reference for touchdown dispersion evaluation of the SLAND mode still to be done.

CONCLUSIONS

The STOL and Maneuver Technology Demonstrator has integrated thrust vectoring and reversing into an Integrated Flight/Propulsion Control system. Level 1 handling qualities have been developed across the full envelope from supersonic and subsonic maneuvering to precision landing. The design was done to the "intent" of MIL-F-8785C, which had to be supplemented in some areas with second-tier criteria that are now included in MIL-STD-1797. This paper has reviewed the S/MTD development from the initial specification through to current flight test results. Lessons learned have been presented with respect to both individual specification criteria and design methodology. Two areas are identified in which it is considered that the current criteria are inadequate: 1) pitch axis requirements as a function of touchdown dispersion, and 2) directional axis requirement for target tracking.

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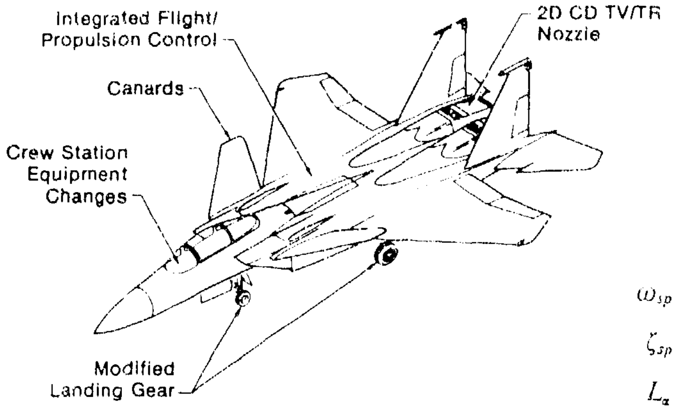


Figure 1. S/MTD Configuration

	ORIGINAL	CRUISE	CONV/COMBAT
ω_{sp} (rad/sec)	3.31	3.31	4.68
ζ_{sp}	0.7	0.8	0.8
L_α (rad/sec)	1.2	1.2	1.2
$1/T_{\Theta 2}$ (rad/sec)	1.2	1.2	2.0
τ_e (msec)	< 70	65	65

a. Pitch Transfer Function Characteristics

CONVENTIONAL
- All effectors except vectoring/reversing

CRUISE
- Optimum flight path control

COMBAT
- Optimum pitch tracking

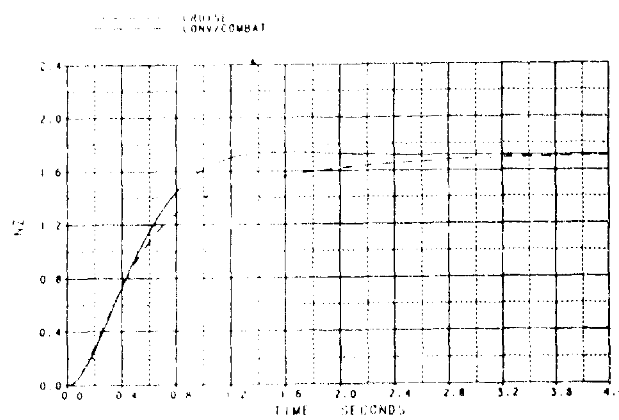
STOL-TOA
- Max performance takeoff/normal approach

STOL-LAND
- Max performance approach/touchdown

STOL-GH
- Rollout handling qualities

SPIN RECOVERY
- Full control authority/no feedbacks

Figure 2. Modes of Control



b. Load Factor Responses

Figure 4. Comparison of CRUISE & CONV/COMBAT Modes

	Honeywell LQG/LTR Analysis	MCAIR Classical Analysis
Longitudinal		
CONV		X
CRUISE/COMBAT	X	
STOL-LAND	X	
STOL-TOA	X	
STOL-GH		X
Lateral/Directional		
CONV		X
CRUISE/COMBAT		X
STOL-LAND		X
STOL-TOA		X
STOL-GH		X
Thrust		
CONV		X
CRUISE/COMBAT		X
STOL-LAND	X	
STOL-TOA		X
STOL-GH		X

Figure 3. Implementation of Modes

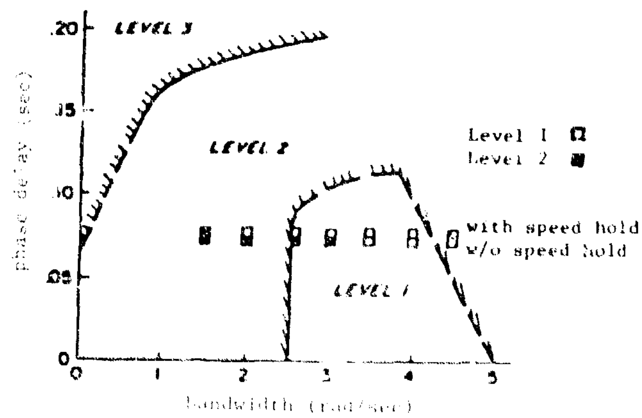


Figure 5. Required Bandwidth for Landing

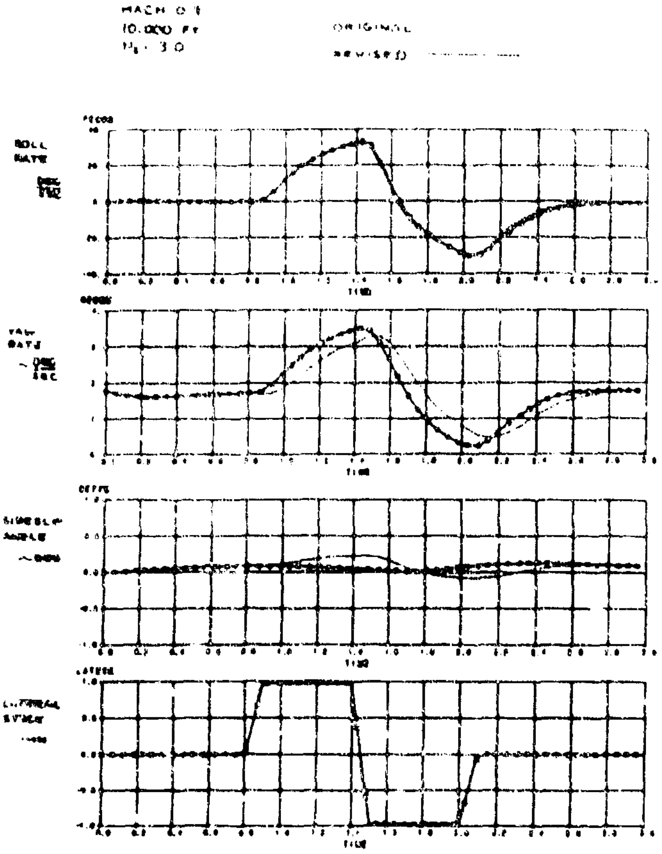


Figure 6. Lateral Tracking

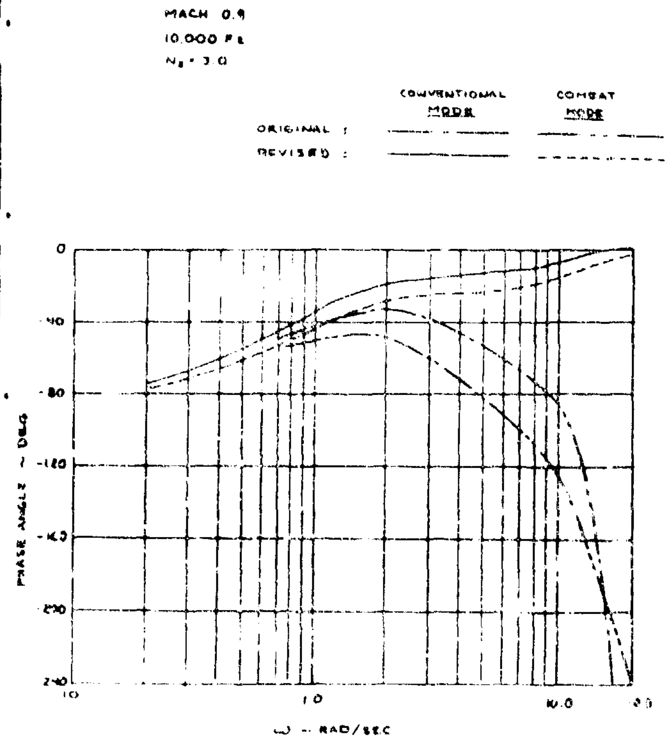


Figure 8. Yaw/Roll Phase Angle

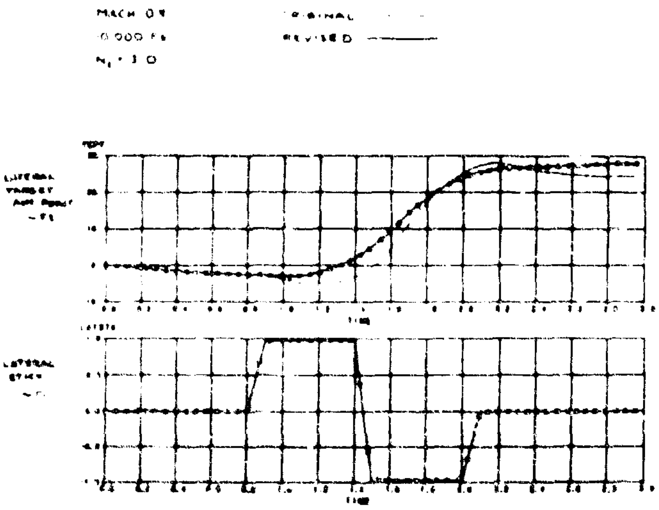


Figure 7. Pipper Aim Point Response

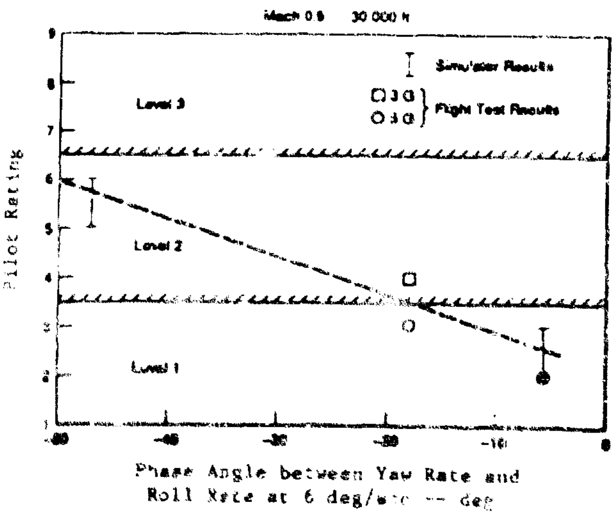


Figure 9. Potential Tracking Criterion

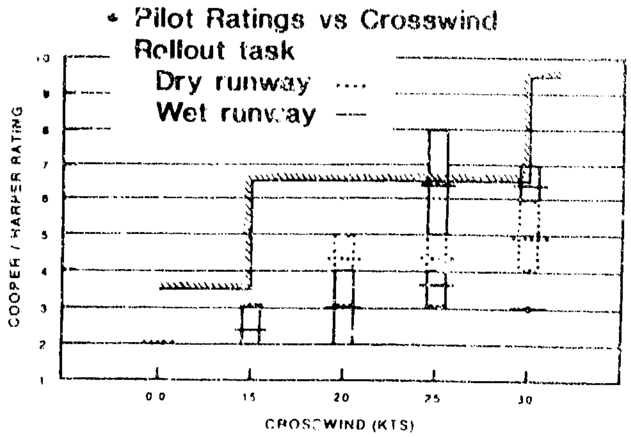


Figure 12. Effect of Crosswind on Pilot Rating

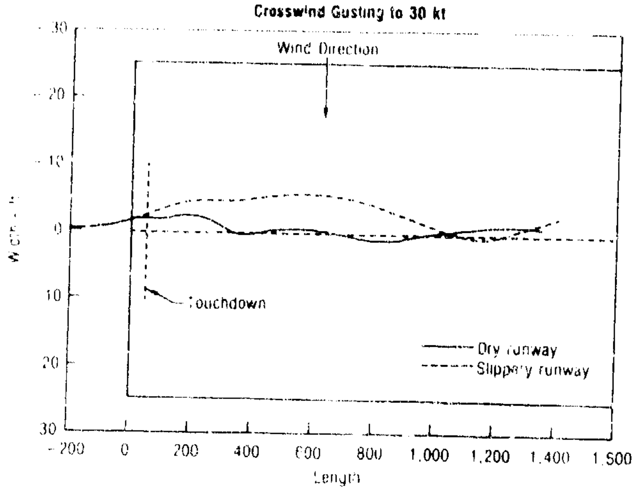


Figure 10. Ground Handling Time Histories

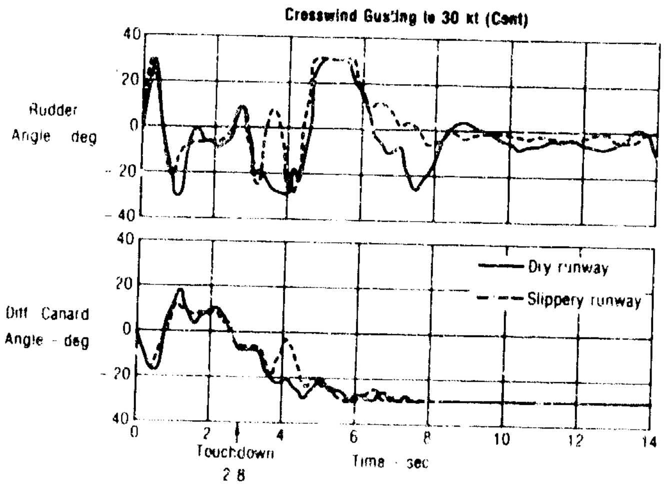


Figure 11. Direct Sideforce and Yaw Control

07 DEC 89 - AUTONOMOUS LANDING GUIDANCE

22 FEB 90 - REVERSING DURING TAXI

23 MAR 90 - FIRST IN-FLIGHT REVERSING

29 MAR 90 - TAKEOFF WITH VECTORING

22 MAY 90 - FIRST SHORT LANDING

01 JUN 90 - SUPERSONIC REVERSING

11 JUL 90 - VECTORING EFFECTIVENESS 30 deg

Figure 13. Flight Test Milestones

FLIGHT CONDITIONS	MODE	GROSS AQUISITION	FINE TRACKING
0.6/10K	CONV	2	2
	COMBAT	3	3
0.7/20K	CONV	3	2
	CRUISE	4	4
	COMBAT	3	3
0.9/20K	CONV	3	2
	CRUISE	4	3
	COMBAT	3	3
0.9/30K	CONV	3	2
	CRUISE	4	
	COMBAT	3	

Figure 14. HQ-I Pilot Ratings

METRICS FOR ROLL RESPONSE FLYING QUALITIES

by

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Summary

Roll characteristics of highly augmented aircraft during compensatory tasks such as tracking and landing have shown to present degraded flying qualities and unstable oscillations similar to those observed in the pitch axis. The present work extends the ideas behind Gibson's method to develop handling qualities criteria for the roll axis control system. The analysis is performed using an existing data-base for highly augmented class IV aircraft and parameters such as roll time constant, system's delay and loop sensitivity are considered for designing for good handling qualities and to evaluate control system performance. Levels of flying qualities are determined in the time domain as well as in the frequency domain for both tracking and landing tasks. Furthermore, the presence of pilot induced oscillations and roll ratcheting is identified.

List of Symbols

ϕ_5	roll angle at t = 5 sec.
ϕ_{nos}	normalized bank angle overshoot
ω_{180}	frequency at phase lag = 180 deg.
ω_{200}	frequency at phase lag = 200 deg.
τ_d	pure time delay
τ_r	roll time constant
τ_{eff}	effective time delay
LATHOS	Lateral High Order Systems
L'_{Fas}	roll control effectiveness
P_r	phase rate
p_{ss}	steady state roll rate
q_{max}	maximum pitch rate
q_{ss}	steady state pitch rate
R_a	acceleration ratio

1. Introduction

Modern high performance aircraft rely heavily on stability augmentation for increased performance. Current experience with aircraft having high order dynamics due to artificial stabilization controllers, digital implementation, etc. has shown a dramatic improvement in system response, controllability and maneouverability. Handling qualities deficiencies, however, have surfaced due to generic problems such as over sensitivity to small control inputs and sluggish response to large inputs. Improper control gains and time delays coupled with higher order effects have resulted in low frequency PIO and high frequency ratcheting interaction [1].

As reflected in the latest military specifications [2], [3], the majority of the work in handling qualities research has been limited to longitudinal motion. Among the reasons were the recognition of the longitudinal plane as the primary plane in closed loop control tasks, the availability of a large experimental data-base and the introduction of aircraft with high level of longitudinal augmentation to compensate for relaxed static stability.

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Traditionally, lateral dynamic characteristics have not been considered critical in the assessment of the overall handling qualities. Specifications for the roll response, for example, are based essentially on open loop type parameters such as roll time constant, maximum roll rate and time to go through a bank angle for roll control effectiveness [2], [3].

Lateral handling qualities, however, are becoming more important in relation to increased control system augmentation level at high angle of attack, to pitch-roll coupling during loaded maneuvers and to agility requirements. Recent experimental results [4] have shown the presence of low frequency pilot induced oscillations in roll (due to mismatch between roll zeros and poorly damped dutch-roll) and high frequency ratcheting [5]. Highly augmented aircraft have also shown poor handling qualities in the roll axis due to the compensation of roll time constant lag to yield an integral-type response, resulting in large lateral acceleration at the pilot's station. The attenuation of such acceleration and consequent lagging has shown [4] to introduce pilot induced oscillations similar to those experienced in the pitch axis.

A comprehensive technique for longitudinal handling qualities analysis and assessment has been developed over the years by Gibson at British Aerospace [6], [7], [8] using experience acquired in the EAP (experimental aircraft programme) and in the development of aircraft like the Jaguar FBW and the Tornado. The main objectives of Gibson's derivations (also referred as the dropback method) are the explanation and prediction of PIO experienced during critical flight phases of highly augmented aircraft. Good correlation has been obtained with the dropback method in matching pilot opinions to aircraft characteristics in tracking and landing for military [8], [9] as well as transport aircraft [10]. Agreement with the new longitudinal handling qualities specifications has also been reported [11], [12].

The basic idea behind Gibson's method is that of achieving "good classical airplane characteristics" in the presence of high order effects due to augmentation. The flight control system should be designed to have not only classical response but also to satisfy the specific task requirements. It is clear that easily and directly identifiable metrics are required instead of specifications given in terms of modes. These would then encompass both classical and modern airplanes in a more straightforward manner.

Gibson's method combines time response and frequency response techniques in the pitch axis to cover the frequency spectrum of interest to the pilot. The time domain analysis looks at attitude, pitch rate and acceleration responses to a block type stick input, as shown in Fig. 1 [7]. The behavior of the initial response at the time of stick release is related to pilot comments. The frequency domain analysis, based on Nichols charts of the open loop response with pure gain manual compensation, highlights PIO tendencies by relating such instability to the phase rate at -180 degrees and its corresponding phase lag crossover frequency. Parameters such as dropback, overshoot and phase rate can be easily obtained. These are related to typical short period dynamics comments like bobbling, sluggishness and induced instability. The Gibson's criterion is not reviewed here and the interested reader can refer to [6], [7], [8], [9], [13] for more details.

The proposed work extends Gibson's technique to the roll axis using the LATHOS experiment as data-base [5]. Metrics for handling qualities evaluation are defined and level boundaries as well as dynamic instabilities are identified for the tracking and landing tasks.

2. Roll Response Requirements and Data Base

The limitations of lateral handling qualities with respect to high performance aircraft have been acutely felt since the MIL-SPEC-8785C, which, while being used for both design and testing of developing aircraft, failed to provide the desired handling qualities. These limitations could be attributed to a limited data base which mainly represented aircraft of the World War II era, having little or no lateral augmentation. The need for identifying problems related to a lateral highly augmented fighter aircraft led to a series of in-flight simulations to generate experimental data and to investigate the effects of lateral high order systems (LATHOS) [5]. The additional insight gained through the LATHOS experiment provided a large enough data-base for the revision of MIL-SPEC-8785C. Currently, MIL-STD-1797 [3] presents the requirements in a format similar to the older specification and, at the same time, provides some guidance for highly augmented systems.

The present paper suggests new roll response metrics based on the application of Gibson's method to the LATHOS data base and presents qualitative as well as quantitative relationships between pilot ratings and typical parameters such as roll time constant, time delay and control system gain, which constitute the scope of the present analysis. Fig. 2 shows the modified block diagram of the lateral-directional flight control system for the stick-roll loop in the LATHOS flight tests. The three primary variables are shown by their corresponding symbols, linear command gain δ_a/F_{as}, roll time constant τ_r, and pure time delay τ_d. The time delay τ_d is the pure time delay generated by the NT-33A's time delay circuit which merely holds the signal by τ_d seconds. This could be thought of as representing delays due to various possible sources like structural lag, transport lag and digital delays present in actual system. The filters surrounding the time delay circuitry are third order Butterworth low pass filters which help in smoothing the signal. The dutch roll characteristics and the lead-lag time constants were held to their nominal value in this work.

The typical first order response for the aircraft is assumed to be given by the transfer function

$$p/F_{as} = \tau_r L'_{Fas}/(\tau_r s + 1) \qquad (1)$$

in which the dutch-roll mode is absent due to dipole effect and the spiral mode is absent due to a large time constant.

The roll requirements and justifications for choosing the above parameters are as follows :

(a) The roll damping requirement given by MIL-STD-1797 limits the lower damping to 0.3 sec. ($0.3 \le \tau_r \le 1$ seconds for level 1). These results are supported by the fact that some modern aircraft equipped with high gain augmentation have time constants that are too small and experience an excessive lateral sensitivity described as roll ratcheting.

(b) The time delay in a control system can drastically degrade lateral flying qualities. The MIL-STD-1797 retains the pure time delay limits given in the 8785C, mainly because of a lack in supporting data to revise them, which limits maximum delay to 0.1 seconds for level 1 and no more than 0.2 seconds for level 2.

(c) A very important parameter that has not been dealt with in sufficient depth is the effect of command gain δ_a/F_{as}. Earlier metrics normally excluded the effect of this open loop gain variation, as it was considered as an independent variable to be optimized. Usually the gain is very critical to the pilot in that it changes his/her equalization and it is known to contribute greatly to the pilot opinion and ratings. Hence the roll metrics developed in this paper include the effect of command gain variation along with other roll parameters. Roll control sensitivity and roll control effectiveness are directly related to command gain, δ_a/F_{as}. We define roll control sensitivity as the gain required by the steady state roll rate, p_{ss} to a unit step input. Roll control effectiveness, L'_{Fas} is defined as roll control sensitivity per unit roll time constant, or

$$L'_{Fas} = p_{ss} / \tau_r \qquad (2)$$

Tasks performed during the LATHOS experiment were representative of flight phases category A and C and consisted of actual target tracking, air-refuelling, precision approach and landing as well as special head up display tracking tasks.

The complete set of configurations, including all the cases with linear gain only, for the tracking task are analyzed (29 cases in all). The analysis of the landing is similar to the tracking (although fewer data points are available) and involve ILS (Instrument Landing System) and visual landing tasks. The analysis makes no distinction between these tasks. In all, 10 configurations were considered with only linear gain variations. Throughout the analysis the pilot behavior is represented by a pure gain and time delay, with transfer function

$$Y_P = Ke^{-\tau s} \qquad (3)$$

where τ is the pilot's time delay constant, usually of the order of 0.2 seconds (these are typical experimental values for the pilot in compensatory tracking tasks). The gain K for the tracking task is adjusted so as to obtain a gain crossover frequency of 0.3 Hz. In the case of the approach and landing task, the gain K is adjusted so as to have a phase lag of 120 degrees at the gain crossover point. Table 1 and 2 list the tracking and landing configurations respectively alongwith the values for the variables, pilot comments and the pilot ratings based on the Cooper-Harper scale.

3. Time Response Analysis

Several new metrics are needed to characterize the roll handling qualities. Gibson's metrics like dropback θ_{db}, the effective time delay θ_{db}/q_{ss} and q_{max}/q_{ss} [6], [7], [8] are no longer directly applicable due to the basic difference between pitch and roll responses and the inclusion of control system command gain.

The basic roll response to step input is of a first order type and hence consists of overshoot only (in the context of Fig. 1). The traditional definition of dropback could now be replaced by 'normalized bank angle overshoot' ϕ_{nos}, and is given by

$$\phi_{nos} = (\phi_{ss} - \phi_5)/\phi_{ss}\,\tau_r \qquad (4)$$

where ϕ_{ss} is the steady state bank angle, and ϕ_5 is the bank angle at time t = 5 sec (time of block input release). However it is seen that direct use of Eq. (4) for the normalized overshoot, though similar in nature to dropback, is not helpful since ϕ_{nos} is independent of roll control sensitivity and by its direct relation to pilot comments no correlation is observed.

The metrics that are introduced in this work are the effective time delay τ_{eff}, the roll control effectiveness p_{ss}/τ_r and the acceleration ratio R_a. The maximum bank angle acceleration is also used in the analysis since it is known that the pilots respond to acceleration cues sensed by their vestibular sensory system.

The pure time delay τ_ϕ does not account for the effect of higher order dynamics. To overcome this difficulty, an effective time delay, τ_{eff} is defined as the time taken by the system to respond to a change in forcing function and it is the sum of pure time delay τ_d and delays due to presence of other higher order terms. This parameter represents the actual response delay as seen by a pilot and it is similar to τ_{eff} defined in [3], the difference being only in the way it is measured.

Roll control effectiveness p_{ss}/τ_r, also represented by $L'_{\delta as}$ in Eq. (1), is used to compute initial roll acceleration. This could be derived by applying initial value theorem to the roll acceleration transfer function, hence the terms roll control effectiveness and initial acceleration are used interchangeably depending on the context.

As it will be shown, a good correlation exists between τ_{eff}, p_{ss}/τ_r and pilot ratings, which is similar to the longitudinal case (θ_{dt}/q_{ss}, q_{max}/q_{ss}) as pointed out in previous work by Rynaski [15]. Some anomalies exist, but these were found to be related to the acceleration ratio, R_a, defined as the ratio of maximum acceleration, (measured from the block step response simulation of the entire system) to initial acceleration p_{ss}/τ_r, (computed from the roll acceleration transfer function), and given by

$$R_a = (\tau_r / p_{ss}) d^2\phi/dt^2 \qquad (5)$$

Tracking Task

The configurations listed in Table 1 were simulated in terms of bank angle, roll rate and roll acceleration time responses. Based on the metrics discussed above, the results of the analysis are shown in Fig. 3.

Fig. 3 relates effective time delay τ_{eff} to initial acceleration p_{ss}/τ_r. Clearly initial acceleration is a function of steady state roll rate p_{ss} and roll time constant τ_r. On the other hand, τ_{eff} is mainly a function of pure time delay τ_d. Qualitative boundaries are drawn in Fig. 3 showing lower and upper bounds on p_{ss}/τ_r. From a physical standpoint, a sufficient threshold value is required for the initial acceleration in order for the response to be sensed by the pilot. However, too large a value is not acceptable. Comments like quick and jerky responses are typified. For any given level of flying qualities, the threshold value for the initial acceleration remains constant (for level 1 L'_{Fas} is > 15 sec ^{2}/lb), though the upper limit is seen to decrease with increase in effective time delay τ_{eff}.

Let us consider a few typical cases and analyze the initial acceleration parameter p_{ss}/τ_r from a different viewpoint. Configurations A, E, N, R all have constant roll control sensitivity but decreasing roll time constant τ_r. Thus the roll control sensitivity p_{ss}/τ_r increases as τ_r decreases. The new MIL STD 1797 conditions specify a lower limit for τ_r. For τ_r < 0.3 sec., ratcheting is predicted. Thus cases A, E do not have ratcheting as τ_r > 0.3 sec., whereas cases N, and R do (τ_r < 0.3 sec.).

Now consider cases F, J, A2 which have τ_r = 0.45 sec. (τ_r > 0.3 sec.), but different roll control sensitivities. In this case A2 exhibits ratcheting deficiency due to high roll control sensitivity. Frome the previous discussion we can conclude that configurations with τ_r < 0.3 sec. are likely to show ratcheting while when τ_r > 0.3 sec. ratcheting may or may not occur depending on roll control sensitivity. The frequency analysis also corroborates the statement. From Fig. 3, for zero time delay, we obtain boundary level limits on L'_{Fas} as

$$\begin{array}{lll} \text{Level 1} & 15 < L'_{Fas} < 60 & \tau_{eff} < 0.07 \text{ sec} \\ \text{Level 2} & 7.5 < L'_{Fas} < 110 & \tau_{eff} < 0.1 \text{ sec} \end{array} \qquad (6)$$

The effective time delay τ_{eff}, affects roll handling qualities in a fashion similar to θ_{dt}/q_{ss} for the pitch axis. Large time delays slow down the response forcing an induced oscillation by the pilot as in configurations C and M. Both are commented as slow responding aircraft and exhibit definite PIO in precision manoeuvers. For level 1 criteria the effective time delay τ_{eff} is limited to less than 0.07 sec. In the case of configurations with higher roll control sensitivity (coupled by large time delays), pilot comments about oscillatory motion border between the PIO and ratcheting regions.

Some anomalies do exist and, as mentioned earlier, can be evaluated using the accelerstion ratio parameter R_a. Agreement with the boundaries is observed if a given configuration satisfies the inequality

$$0.91 > R_a > 0.71 \qquad (7)$$

Failure to satisfy Eq. (7) yields relatively poor pilot ratings compared to neighboring points.

Configurations B and U are typical in this respect. They both have level 3 handling qualities but lie very close to configurations N and L which are level 2. B and U, however, do not satisfy the specified bounds on R_a, whereas, configurations N and L do. The acceleration ratio thus can be used as a test for normal/abnormal response as described by pilot comments.

Landing Task

The landing task usually deserves particular attention because it can never be avoided and may present pilot induced oscillations non detectable by non terminal flight tests and ground simulations. The results of the analysis are summarized in Fig. 4 and should be compared to Fig. 3. Again large time delays are associated with sluggish response and, possibly, PIO as shown by configurations M and E.

Too high values of initial acceleration are unacceptable and the results in Fig. 4 are consistent with the limits given in the present military specification [3]. On the other hand, lower values of roll acceleration are acceptable unless the time delay is too large. For the landing task, the following boundaries can be identified:

$$\begin{array}{lll} \text{Level 1} & 10 < L'_{Fas} < 42 \text{ sec}^{-2}/\text{lb} & \tau_{eff} < 0.11 \text{ sec} \\ \text{Level 2} & L'_{Fas} < 10,\ 42 < L'_{Fa} \text{ sec}^{-2}/\text{lb} & \tau_{eff} < 0.15 \text{ sec} \end{array} \qquad (8)$$

The presence of ratcheting appears to be function of roll control effectiveness. Unlike the tracking analysis, however, not enough data was available to fully support the conclusion.

Separate comments must be made for configuration G. There seems to be a discrepancy between its location well within level 1 boundaries and the pilot comments indicating the presence of ratcheting and a level 2 rating. In this case, the acceleration ratio R_a can be used effectively to identify the anomaly. It can be computed that normal response (absence of oscillations and instabilities) requires the acceleration ratio to be $0.76 < R_a < 0.94$. Lower values are indicative of ratcheting, while values of R_a near unity imply the presence of PIO. Since configuration G has

$R_a = 0.75$, the ratcheting instability can be accounted for in Fig 4. In conclusion, level 1 and 2 boundaries can again be identified, keeping in mind the constraints on R_a.

In summary, the metrics used in the context of roll performance using Gibson's method are given in Table 3.

4. Frequency Response Analysis

The frequency analysis does not deviate much from that of the pitch axis, except that it includes the roll acceleration frequency response to identify ratcheting phenomena which are unique to the roll axis. Compensatory tasks, like precision tracking and approach and landing tasks, consist of an open loop closure via the pilot in the feedback loop. During the task execution it is essential that the aircraft have good predictability and a smooth response. Such good behavior is observed to be related to an open loop frequency response around the crossover region. The crossover point, defined as the point where the frequency ratio has unit gain (0 db), plays an important role in the analysis of such tasks. Gibson's frequency domain analysis is applied to the open loop frequency response of roll attitude around the crossover region and relates it to the pilot comments. Deviation from good behavior leads to additional closed loop control activity and possibly to dynamic instabilities.

The two types of dynamic instabilities exhibited by the roll axis are PIO and ratcheting. PIO is a low frequency instability and is related to pilot's control activity around the phase crossover region, where the deficiency is caused by the pilot's stick input being out of phase with the aircraft response. The attitude frequency response shows that this is possible with a rapid increase in phase lag accompanied by a very little gain attenuation around the phase crossover point. Quantitatively, the phase rate parameter P_r [8], [9] gives excellent correlation with PIO in the pitch axis as well as in roll. The phase rate is evaluated using Eq. (9).

$$P_r = 20/(\omega_{200} - \omega_{180}) \text{ deg/Hz} \qquad (9)$$

The other instability, unique to the roll axis, is ratcheting. Historically, roll ratcheting was identified by pilots as a high frequency PIO. Recent findings indicate that roll ratcheting is excited by the pilot's neuromuscular activity [4]. It has been shown [4] that the natural frequency of the neuromuscular system lies in the frequency band from 12 to 18 rad/sec. Ratcheting affects the ride qualities but does not usually endanger the pilot / aircraft system and it can be improved by altering the aircraft stick feel system [8].

Tracking Task

Frequency response for roll attitude and roll acceleration are analyzed for all the tracking configurations. Deficiencies like PIO, quick and jerky responses are identified by using Nichols charts of roll attitude. The roll attitude frequency responses are compared for all the configurations, with gain equalization so as to have a gain crossover frequency of 0.3 Hz (0.3 Hz is identified with the pilot's control activity). The same gain adjustment is also used to obtain the roll acceleration frequency responses, necessary to identify ratcheting. From the analysis of roll acceleration frequency response and stability margins, it is found that the presence of ratcheting occurs if the roll acceleration crossover frequency lies within the limits

$$12 \text{ rad/sec} < \omega_{180} < 18 \text{ rad/sec}. \qquad (10)$$

corresponding to typical pilot's neuromuscular bandwidth. These boundaries can extend to 10 rad/sec on the lower side and to 20 rad/sec on higher side if other deficiencies like PIO, jerky, oscillatory response are present.

The present roll criteria [3] associate ratcheting to a low roll time constant ($\tau_r < 0.3$ sec.), however they fail to single out configurations such as Q and A5. Ratcheting is not limited to low τ_r values, for example configuration A3 ($\omega_{180} = 20$ rad/sec) has ratcheting although it has $\tau_r = 0.45$ sec. Use of Eq. (10) appears therefore more accurate in describing occurrence of ratcheting.

Fig. 5 shows the roll attitude frequency response boundaries. The various regions are labeled showing the nature of the aircraft response. There are three primary regions : (1) sluggish, PIO prone region; (2) optimal tracking region; and (3) quick, oscillatory ratcheting prone region. The central band, which marks a region of good and optimal response, shows a good phase-gain relationship.

If the attitude plot is to the left or right of the central band the response of the aircraft becomes less or more sensitive, respectively. To the left of the good response band lies the sluggish, PIO prone zone. Time delay and higher order terms cause the frequency lines to shift to the left of the central band. Time delay causes a quicker phase lag increase, with smaller gain attenuation causing sluggish response. The PIO, sluggish response boundary shows good correlation with most of the PIO cases, provided the attitude response be greater than -10 dB. Frequency response less than -10 dB does not effect the pilot comments.

In Fig. 6, three cases are selected as an example. Configuration J shows a well-behaved response. Configuration M clearly indicates the presence of PIO and configuration A5 is consistent with pilot comments reporting a quick, jerky response and ratcheting.

The above examples are typical of what can be expected in general, with frequency response boundaries showing the general trends in phase lag and magnitude attenuation. Frequency boundaries plots, however, do not include the effects of control sensitivity nor τ_r and τ_d directly. The effects of variation in roll time constant and time delay on dynamic instabilities is shown in Fig. 7 by plotting phase rate versus phase lag crossover frequency. The phase rate plot shows τ_r and τ_d isoclines, constant τ_r are represented by vertical curves moving to the left as τ_r increases, whereas τ_d are the horizontal curves which shift upwards as τ_d increases.

From the Figure it can be seen how τ_r affects ratcheting (note the MIL-STD level 1 boundary at 0.3 seconds). The time delay τ_d, on the other hand, affects phase rate which in turn is indicative of PIO. Since phase rate increases with τ_d, large delays cause PIO and the horizontal boundary drawn for $P_r = 75$ deg/Hz separates PIO from non-PIO regions.

In the above discussion roll control sensitivity was not considered because pilot-in-the-loop equalization acts as a normalizing variable gain. To account for sensitivity effects, a thumbprint plot is drawn in Fig. 8. The ordinate of the thumbprint is the gain equalization required to attain a 0.3 Hz crossover, while the abscissa represents the corresponding phase lag value for the adjusted attitude gain crossover point. The figure shows configurations to be separated in three bands. These bands are the gain sensitivities of the configurations used in the analysis. The central zone identifies the level 1.

Landing Task

The landing frequency analysis is carried out in a fashion similar to the Gibson's frequency analysis done for the longitudinal landing case [6], [7]. The frequency response for the longitudinal landing task looks at the open loop response of the pitch attitude and the attitude gain is adjusted so as to have a unity gain with a phase lag value of 120 degree. Similarly, the bank angle frequency response with the loop gain adjusted so as to have a unity gain at a 120 degree phase lag value is considered in the present analysis. Using pilot comments in [5] we obtain the frequency response boundaries for the task as shown in Fig. 9.

The various regions labeled on Fig. 9 correspond to different aircraft responses. Four main zones are identified : (1) optimal approach and landing region, (2) conditionally good region (3) sluggish, PIO prone region and (4) quick, oscillatory and ratcheting prone region.

The region marked (1) shows a zone of good and optimal behavior, configurations B, I, J for example all satisfy these boundaries. Pilot induced oscillations are excited when the crossover frequency ω_{180} is low (typically $\omega_{180} < 0.5$ Hz) and the phase lag increases rapidly with little or no gain attenuation. Regions (2) and (3) are representative of PIO and sluggish response, mainly due to the effect of large time delays. This region is divided into a sluggish PIO prone section (3) and a conditionally good response section (2). The condition for the response to be good is that ω_{180} be greater than 0.5 Hz. For example configuration C lies in this region and has $\omega_{180} > 0.5$ Hz and it is commented to have good flying qualities, whereas configuration L has $\omega_{180} < 0.5$ Hz. and exhibits PIO Configurations lying in region (3) have sluggish or delayed response. Pilot induced oscillations are also seen to accompany these responses provided the condition of $\omega_{180} < 0.5$ Hz is satisfied.

Region (4) includes configurations exhibiting sharp, quick, oscillatory responses as well as ratcheting. To be noted that the presence of the ratcheting instability requires a crossover frequency ω_{180} between 12 an 18 rad/sec and a magnitude greater than zero dB in the roll acceleration frequency response, a condition similar to what was found for the roll tracking task [2].

Similar to Fig. 8, the effects of roll time constant and time delay variations on landing dynamic instabilities are summarized in Fig. 10. Here, the phase rate limit between PIO and non-PIO regions is found to be $P_r = 90$ deg/Hz. A thumbprint plot for landing, accounting for roll control sensitivity effects is shown in Fig. 11. The ordinate of the thumbprint is the gain equalization required to attain a phase lag crossover of 120 degree, while the abscissa represents the corresponding frequency at the crossover point. The configurations are separated into two bands depending on the gain sensitivities used in the analysis and the central zone indicates a possible level 1 region.

The frequency response analysis results are summarized in Table 4 which shows ranges of the metrics leading to dynamic instabilities such as pilot induced oscillations and ratcheting.

Conclusions

An analysis of roll performance handling qualities was carried out using the Gibson's method and applied to the LATHOS (Lateral Higher Order Systems) data-base. The method, consisting of a combination of time domain and frequency domain techniques has proved to give results consistent with the experimental data. New time response metrics were introduced to account for control sensitivity, roll time constant and time delay effects. Configurations relative to both tracking and landing tasks were considered and level boundaries were obtained as well as indication of regions subject to PIO and other dynamic instabilities. The Gibson's method appears to have a general applicability in both the pitch and roll axes and it is an attractive alternative to the modal requirements of present handling qualities specifications and to designing for good aircraft performance.

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Table 1 Pilot Comments, Tracking Task

CASE	MEAN PILOT RATING /LEVEL	SYMBOL	τ_r (sec)	τ_d (sec)	P_{ss}/F_{ss}	PILOT COMMENTS
(TR)1-3	3/L1	A	0.8	0.00	18	NPio,NRa,GIR,GFR,GAg,GP,GFt,GSen
(TR)1-3T1	7/L3	B	0.8	0.03	18	Pio,NRa,SIR,GAg,BP,BFt,LSenL
(TR)1-3T2	8/L3	C	0.8	0.06	18	Pio,NRa,SIR,GFR,BAg,BP,BFt,LSen
(TR)1-2	5/L2	D	0.8	0.00	10	NPio,NRa,SIR,BFR,GAg,BPL,GFt,SSen
(TR)2-3	3/L1	E	0.45	0.00	18	NPio,NRa,GIR,GFR,GAg,GP,GFt,GSen,HSenS
(TR)2-3T1	4.5/L2	F	0.45	0.03	18	Osc,NPio,NRa,OIR,OFR,BAg,OP,GFt,HSen
(TR)2-3T2	5.5/L2	G	0.45	0.06	18	Osc,NPio,NRa,OIR,BAg,BP,BFt,HSen
(TR)2-3T3	8.5/L3	H	0.45	0.08	18	Osc, Pio,OIR,OFR,BAg,BP,BFt,LISen,HFSen
(TR)2-2	3/L1	I	0.45	0.00	10	NPio,NRa,SIR,OFR,GAg,GP,GFt,OSen
(TR)2-2T1	2/L1	J	0.45	0.03	10	NPio,NRa,GIR,GFR,GAg,GP,GFt,GSen
(TR)2-2T2	5.5/L2	K	0.45	0.06	10	NPio,NRa,SIR,OFR,BAg,BP,BFt,LSen
(TR)2-2T3	7/L3	L	0.45	0.08	10	Pio,NRa,NIRS,OIRL,GFRL,BAg,BPS,GPL,OSen
(TR)2-2T4	8/L3	M	0.45	0.18	10	Osc,Pio,NRa,BIR,BFR,BAg,BP,BFt,LSen
(TR)3-3	5/L2	N	0.25	0.00	18	Osc,Pio,Ra,OIR,OFR,BAg,OP,BFt,HSenS
(TR)3-3T2	7/L3	O	0.25	0.06	18	Ra,OIR,BFR,BAg,BP,HSen
(TR)3-3T3	7/L3	P	0.25	0.08	18	Ra,OIR,BFR,BAg,OP,GFt,HSen
(TR)3-2	3.5/L1	Q	0.25	0.00	10	NPio,NRa,GIR,GFR,GAg,GP,OFt,NSen(about neutral)
(TR)5-3	7/L3	R	0.15	0.00	18	Osc,Ra,OIR,GFR,BAg,BP,BFt,LISen,HFSen
(TR)5-3T1	7/L3	S	0.15	0.03	18	Osc,Ra,OIR,OFR,BAg,BP,BFt,HSen
(TR)5-3T2	8/L3	T	0.15	0.06	18	Osc,Pio,OIR,BFR,BAg,BP,HSen
(TR)5-2	7/L3	U	0.15	0.00	10	Osc,Ra,OIR,BFR,BAg,BP,BFt,OSen
(TR)5-2T1	7/L3	V	0.15	0.03	10	Osc,Ra,BAg,HSenS
(TR)5-2T3	8/L3	W	0.15	0.08	10	Osc,Ra,OIR,OFR,BAg,BP,HSen
(TR)2-4	3.5/L1	A1	0.45	0.00	25	NPio,NRa,GIR,GFR,GAg,GP,GFt,LISen,OFSen
(TR)2-4T1	5/L2	A2	0.45	0.03	25	Osc,Ra,OIR,OFR,BAg,BP,BFt,HSenS
(TR)2-4T2	6/L2	A3	0.45	0.06	25	Pio,Ra,OIR,OFR,BAg,BP,HSen
(TR)2-4T3	9/L3	A4	0.45	0.08	25	Pio,BIR,BFR,BAg,BP,HSen
(TR)3-4	6/L2	A5	0.25	0.00	25	Osc,NPio,NRa,OIR,OFR,BAg,BP,BFt,HSen
(TR)3-4T2	7/L3	A6	0.25	0.06	25	Osc,NPio,Ra,OIR,BAg,BP,HSen

PREFIX
B = Bad
G = Good
H = High
L = Low
N = No
O = Okay
Q = Quick/Jerky
S = Sluggish

SUFFIX
L = Large input
R = Response
S = Small input

Table 2 Pilot Comments, Landing Task

CASE	MEAN PILOT RATING /LEVEL	SYMBOL	τ_r (sec)	τ_d (sec)	F_{ss}/F_{as}	PILOT COMMENTS
(LA)1-1	4.5/L2	A	0.8	0.00	5	NPio,NRa,SIR,BFR,BAg,GP,GSen
(LA)2-1	2/L1	B	0.45	0.00	5	NPio,NRa,GIR,GFR,GP,OSen
(LA)2-1T1	4/L2	C	0.45	0.03	5	NPio,NRa,GIR,GFR,GAg,OP,LSen
(LA)2-1T2	5/L2	D	0.45	0.06	5	Osc,NPio,NRa,SIR,BFR,BAg,BP,LSen
(LA)2-1T4	9/L3	E	0.45	0.18	5	Osc,Pio,NRa,SIR,OFR,BAg,BP,LSen
(LA)3-1	3.5/L1	F	0.25	0.00	5	NPio,NRa,GIR,GFR,GAg,GP,LSen
(LA)4-1	5/L2	G	0.2	0.00	5	NPio,Ra,OIR,OFR,BAg,BP,OSen
(LA)4-1T2	3/L1	H	0.2	0.06	5	Osc,NPio,NRa,BFR,BAg,BP,LSen
(LA)1-2	3/L1	I	0.8	0.00	10	NPio,NRa,OIR,OFR,OAg,OP,OSen
(LA)1-2T1	3/L1	J	0.8	0.03	10	Osc,NPio,NRa,GIR,OFR,BAg,GP,GSen
(LA)1-2T2	6/L2	K	0.8	0.06	10	Pio,NRa,SIR,BFR,BAg,BP,OSen
(LA)1-2T3	8/L3	L	0.8	0.08	10	Osc,Pio,NRa,SIR,BAg,BP,LSen
(LA)1-2T4	8/L3	M	0.8	0.18	10	Pio,NRa,BIR,BFR,BAg,BP,BSen
(LA)2-2	2/L1	N	0.45	0.00	10	NPio,NRa,GIR,GFR,GAg,GP,GSen
(LA)3-2	2/L1	O	0.25	0.00	10	Osc,NPio,NRa,GIR,GFR,GAg,GP,OSen
(LA)3-2T2	5/L2	P	0.25	0.06	10	BIR,BP (Partial evaluation)

ROOT
Ag = Aggressiveness
F = Final
Ft = Fine Tracking
I = Initial
Osc= Oscillation
P = Predictability
Pio = Pilot Induced Oscillation
Ra = Ratcheting
Sen= Sensitivity

Table 3 Time Response Metrics

Metric	Description	Tracking	Landing
τ_{eff}	Effective time delay	Figure 3	Figure 4
L'_{Fas}	Initial roll acceleration p_{ss}/τ_r	Figure 3	Figure 4
R_a	Roll acceleration ratio	$0.91 > R_a > 0.71$	$0.94 > R_a > 0.76$

Table 4 Values of Metrics for Dynamic Instabilities

Deficiency	Tracking	Landing
PIO	$\omega_{180} < 0.75$ Hz and $P_r \geq 60$ deg/Hz or $\tau_r < 0.3$ sec	$\omega_{180} < 0.75$ Hz and $P_r \geq 60$ deg/Hz
Ratcheting	$\omega_{180} = (10\text{-}20)$ rad/sec roll accel. > -10 dB	$\omega_{180} = (12\text{-}18)$ rad/sec roll accel. > 0 dB

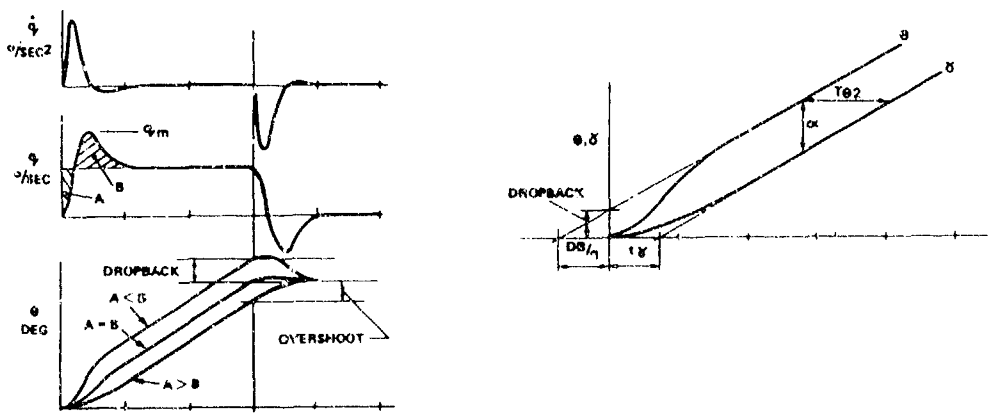

Figure 1. Longitudinal Dropback Time Responses (taken from [7])

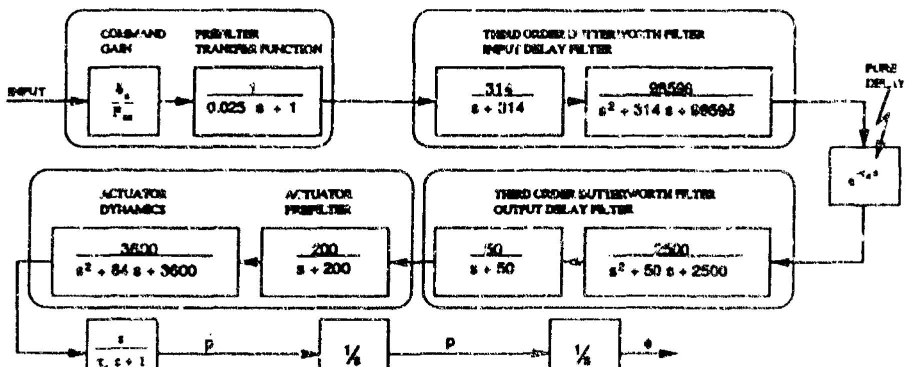

Figure 2. Lateral-Directional Block Diagram (taken from [3])

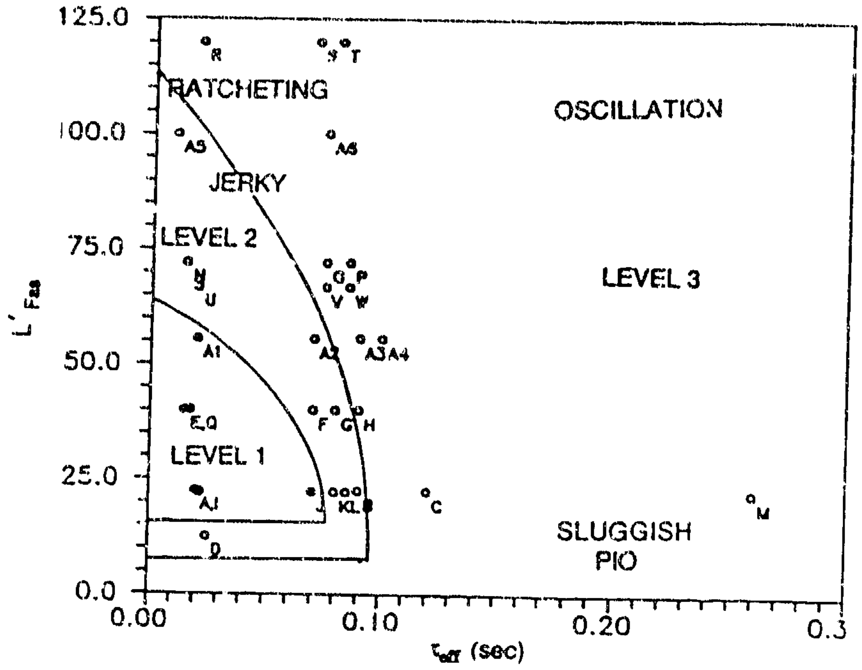

Figure 3. Time Response Analysis, Tracking Task

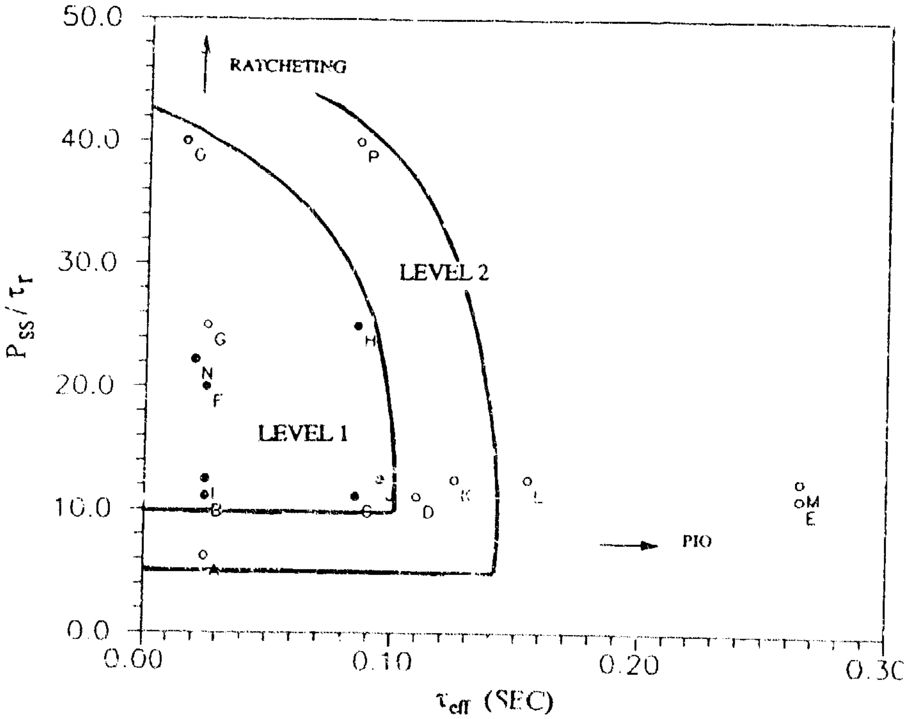

Figure 4. Time Response Analysis, Landing Task

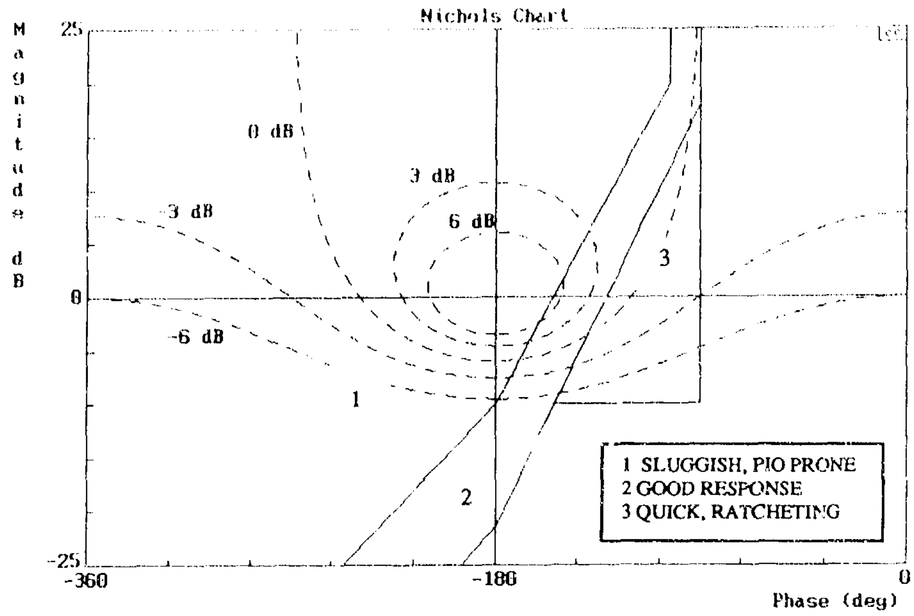

Figure 5. Frequency Response Boundaries, Tracking Task

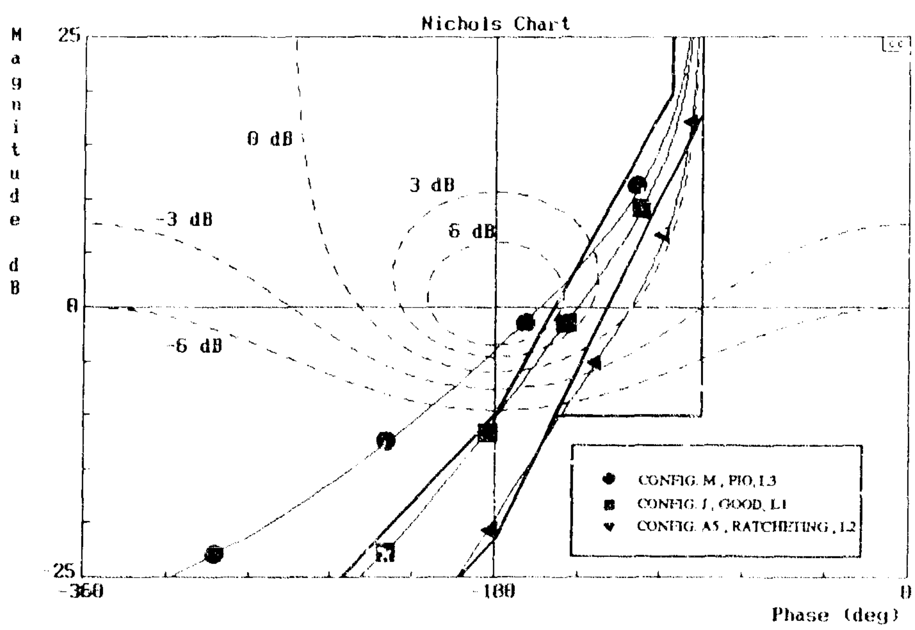

Figure 6. Frequency Response, Sample Configurations, Tracking

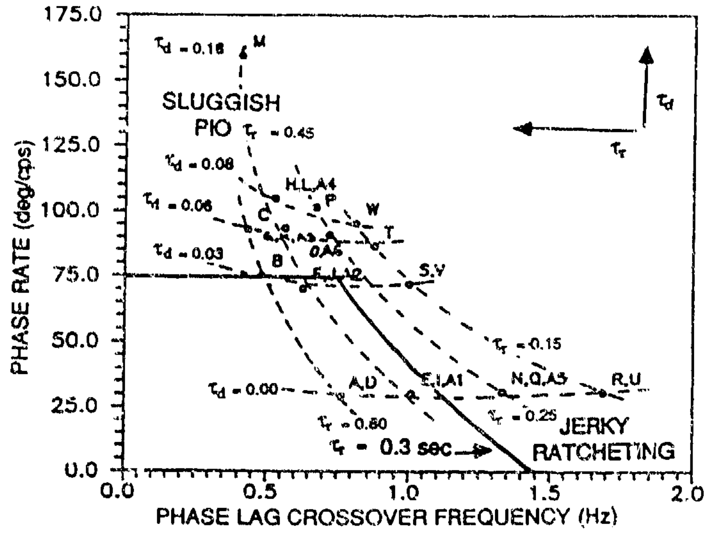

Figure 7. Phase Rate Diagram, Tracking Task

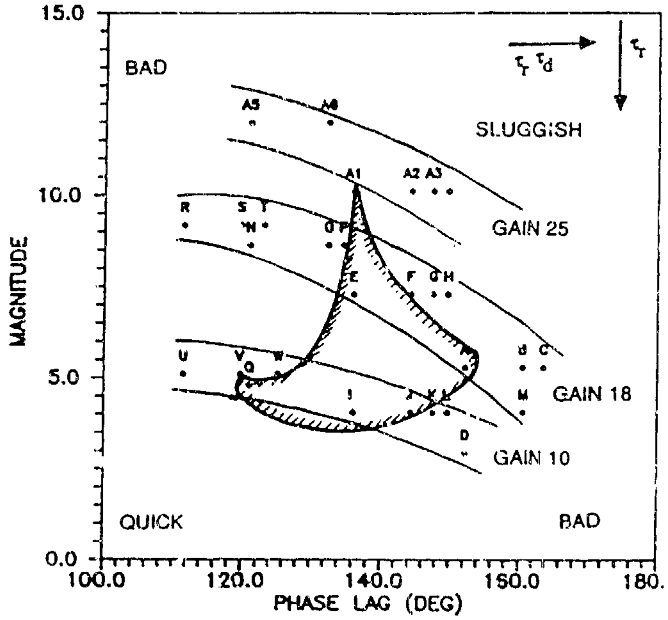

Figure 8. Level 1 Boundary with Control Sensitivity, Tracking

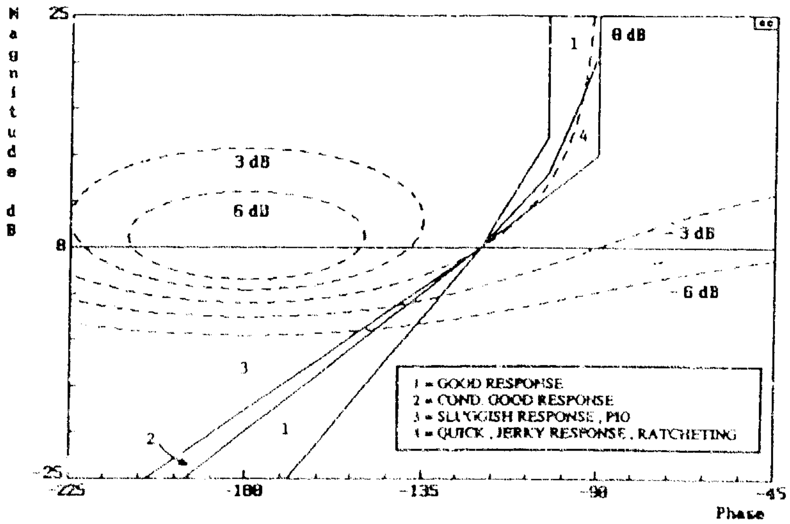

Figure 9. Frequency Response Boundaries, Landing Task

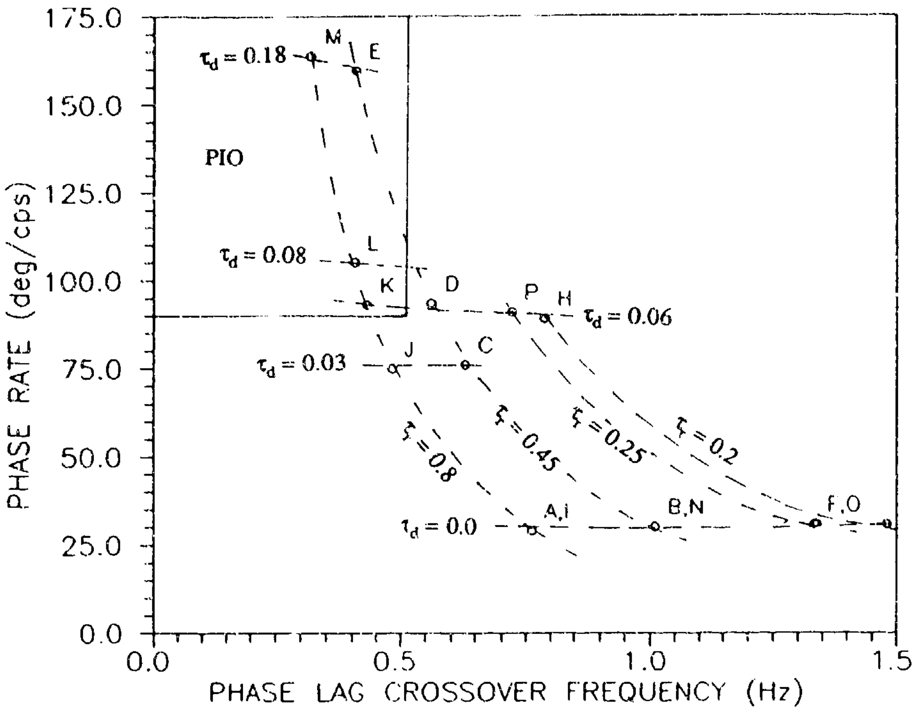

Figure 10. Phase Rate Diagram, Landing Task

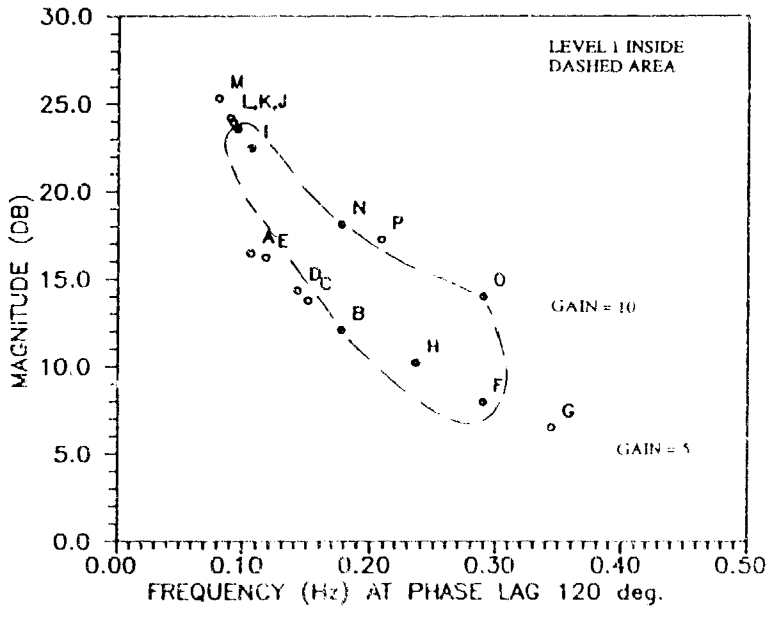

Figure 11. Level 1 Boundary with Control Sensitivity, Tracking

HANDLING QUALITIES GUIDELINES FOR THE DESIGN OF FLY-BY-WIRE FLIGHT CONTROL SYSTEMS FOR TRANSPORT AIRCRAFT

by

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SUMMARY

This paper describes the work of GARTEUR Flight Mechanics Action Group 01 on Handling Qualities and summarises the contents of its final reports. First, the objectives (which concentrated on longitudinal control) are outlined. Secondly, the flight control systems designed and used, and the simulator trial, are described. Thirdly, the results are reviewed. Fourthly, existing handling qualities criteria, and the Action Group's tentative proposals for handling qualities measures which can be applied to flightpath control and system changeover, are assessed. Finally, the Group's guidelines and recommendations for further work are reviewed.

1 INTRODUCTION

The Group for Aeronautical Research and Technology in EURope (GARTEUR) was set up in 1981 by France, the Federal Republic of Germany, the Netherlands and the United Kingdom. It aims at stimulating and coordinating cooperation between research institutes and industry in the areas of aerodynamics, flight mechanics, helicopters, structures and materials and propulsion technology. The GARTEUR management has approved the publication of this paper which describes the work of GARTEUR Flight Mechanics Action Group 01 on Handling Qualities, (which was set up in 1982 and has recently completed its work), and summarises the Action Group's final reports. This work involved collaboration between the ONERA, DLR, NLR and RAE research institutes of the four GARTEUR member countries, with advice from industry.

2 OBJECTIVES

The Action Group first investigated, through a questionnaire to the aerospace industry of the four European countries involved, the anticipated manual control tasks and control concepts for future transport aircraft with advanced flight control and display systems. Following industry's response and a review of existing longitudinal handling qualities criteria, a comprehensive study on the NLR moving-base piloted flight simulator was approved, with the aim of establishing longitudinal handling qualities guidelines for future transport aircraft with advanced systems.

Some concern had been expressed by Industry that the results aimed at by the Action Group could possibly develop into additional requirements to be applied by the certifying authorities. However, the Action Group was convinced that the application of ACT brings with it such wide possibilities that useful, generally applicable, handling qualities guidelines will be necessary for the design of ACT transport aircraft, and could be generated as a result of the proposed flight simulator experiment.

Three distinct aspects were investigated, in the terminal flight phases:

a. The validity of various existing and proposed handling quality criteria, based on pitch rate control, for the design of a sophisticated (longitudinal) control system which instead applies flightpath as the primary controlled parameter.

b. The value of these handling quality criteria for the design of a satisfactory backup system based on pitch rate control, to which control reverts in case of improper functioning of the sophisticated primary system, eg due to the occurrence of a massive sensor failure.

c. In particular the implications of the change in handling qualities when reverting from a sophisticated primary manual Flight Control System (FCS) to a simpler backup longitudinal system having different, but still good, characteristics. (Industry had commented that the handling qualities of a future backup Fly-By-Wire (FBW) system would be as good as those of current aircraft).

3 FLIGHT CONTROL SYSTEMS AND ASSOCIATED DISPLAYS

Preparatory investigations were first performed in the simple fixed-base simulation setup of the ONERA Flight Mechanics Laboratory. The purpose of these investigations was to design and validate a harmonized primary flight control and display system for the later comprehensive investigation at NLR, by undertaking a limited pilot assessment. At the end of this preparatory work two primary and one backup longitudinal flight control systems were flown by evaluation pilots at ONERA in early 1987. Apart from pilot comments on the obvious limitations of the ONERA experiment, it was concluded that the objectives had been met, and that a sound basis had been established for the study at NLR. The conclusions and recommendations from this study formed the background for the comprehensive simulator experiment at NLR.

Following the preparatory investigations at ONERA, a Typical Heavy Electrical Transport Aircraft (THETA-4) based on an existing NLR aircraft model was represented on the NLR simulator. This model was derived from a Boeing 747 model with some modifications so that the aircraft behaved almost like the aircraft model (THETA-3) used in the investigation at ONERA, that means having the same approach speed and a slightly unstable natural longitudinal behaviour. It was also equipped with an all flying tail and a fly-by-wire FCS.

Four longitudinal flight control systems were designed for this study.

Two were sophisticated primary systems with flightpath rate command, and two were satisfactory backup systems.

The first primary system, A, had a flightpath angle hold term which held the actual angle existing at the instant of stick release. Stick inputs could therefore produce undesired step redatuming of the commanded flightpath angle, because of discontinuities in the flightpath angle feedback loop. A gamma-trim switch was therefore included to permit the pilot to produce small flightpath angle changes without discontinuous effects.

The second primary system, B, had the same flightpath angle term as system A, but it held the angle being commanded at the instant of stick release. In this system, stick inputs went through a forward flightpath integrator in combination with a continuous gamma-feedback loop. The system had a somewhat more complex layout than system A, but it did not produce discontinuities so did not require a gamma-trim facility.

Fig 1 shows typical responses of commanded and actual flightpath angle for systems A and B. Fig 2 illustrates step redatuming of system A.

As noted, the two alternative primary longitudinal flight control systems provided flightpath rate command with flightpath angle hold. Both smoothly changed to a pitch rate mode for flare and landing. All these primary systems provided bank compensation and an autothrottle. On failure a step change to one of two possible pitch rate backup systems (C or D) occurred. The backup systems had no bank compensation or autothrottle. A single lateral system, providing roll rate command with bank angle hold, was used throughout and did not change on failure.

The fundamental differences between the response of the flightpath mode of primary systems A and B, and the backup systems C and D and the flare mode of A and B, to a block pilot input are shown in Figs 3 and 4.

a. The normal acceleration (and therefore flightpath) responses of the flightpath mode of A and B, are crisper than those of C, D and the flare mode of A and B

b. This is achieved by using substantially more pitch rate overshoot for the flightpath mode of A and B, than for C, D and the flare mode of A and B.

c. The responses of the flightpath mode of A to the onset and removal of the input are different, reflecting its switched flightpath hold term, while B, C, D and the flare mode of A and B are symmetrical because all their terms are continuous.

Head-down instrumentation consisted of an experimental Electronic Flight Instrument System (EFIS) which included two display units positioned in a vertical array in front of the pilot. The top, Primary Flight Display, provided primary flight information including flightpath vector. The bottom, Navigation Display, provided what was basically an electronic equivalent of a Horizontal Situation Indicator.

No attempt was made to design a full head-up display (HUD). The restricted HUD format (Fig 5) used in the NLR simulator included commanded and actual flightpath, -3 deg flightpath target and synthetic runway, but did not provide airspeed or altitude.

Changes to the HUD format (accompanied by an audio signal) provided a flare warning as the aircraft descended through 30.5 m (100 ft).

It was assumed that the massive sensor failure which caused reversion to the backup control system would also affect the displays, so the HUD and head-down flightpath symbols were removed when a control system failure occurred.

4 SIMULATOR TRIAL

The NLR moving-base flight simulator was fitted with a transport aircraft cockpit flown using a sidestick controller.

The piloting task consisted of flying selected parts of 'circuits' including take-off, climbing left turn to inverse runway heading, levelling off at 609.6 m (2000 ft), manoeuvring the aircraft onto the localizer, performing an ILS approach, flare and landing.

Failure of the primary FCS might occur somewhere in the circuit. Failure was announced by an audio signal which the pilot had to cancel. He was required to continue and land with the backup system.

Daylight visual conditions were provided, with a cloud base at 152.4 m (500 ft).

Disturbances included both wind profile and turbulence. The wind profiles defined the mean wind strength and direction as a function of altitude, and the turbulence defined the random deviation around the mean values.

Wind profile could be different for each circuit.

In the experiments, two levels of environmental disturbance were used:

1 'low' level of disturbance.

2 'high' level of disturbance.

Some familiarization runs were made with no disturbance.

Four different wind profiles were used for each level of disturbance.

The turbulence was structured according to a model described in Ref 1 which is able to generate time histories having the property of 'intermittency', representative of a non-gaussian distribution of velocity differences in real atmospheric turbulence.

Each of the four participating test pilots evaluated the four possible control system changeovers and the corresponding failure-induced change in characteristics, based on flying about 50 (partial) circuits in the simulator. The results obtained consisted of:

a. Cooper-Harper ratings given by the pilots for the behaviour of the primary and backup flight control systems separately.

b. Extensive commentary given by the pilots during the execution of the runs, and during the debriefing in response to a questionnaire.

c. Recorded performance measures, consisting of values of selected parameters at the moment of failure, threshold and touchdown and of mean values, standard deviations and level crossings of selected parameters for certain segments of the circuit.

5 SIMULATOR RESULTS

Preliminary results were presented at the AIAA Atmospheric Flight Mechanics Conference at Minneapolis, USA, in August 1988 (Ref 2). The full results have been published in the Action Group's final report, GARTEUR/TP-055 (Ref 3).

Primary systems:

System A flightpath mode received an equal number of Level 1 and Level 2 handling qualities ratings, whereas system B was said by all pilots to provide Level 1 handling qualities and to be a near-optimal FCS for the aircraft under consideration. System A flightpath mode was rated less satisfactory than system B mainly because of the disliked jumping of the flightpath command and its symbol on the HUD.

Performance level crossings indicate that either the automatic airspeed control was not tight enough or the performance limits (on the approach ±2 kn desired, ±5 kn adequate) were too stringent.

The pitch rate command mode of A and B used below 100 ft wheel height permitted the pilots to make landings which were good for a simulator.

Changeover in handling qualities:

The loss of system sophistication at FCS failure sometimes made it necessary for the pilot to react immediately to sustain the flight condition at which the failure occurred, this was considered objectionable.

The loss of information from the HUD at FCS failure made the remaining HUD incomplete for full control of the aircraft, which made its use doubtful.

The handling qualities changeover from flightpath to pitch rate command, for the primary systems at 100 ft wheel height, did not provoke any pilot comment.

Backup systems:

Although the performance measures sometimes indicate slight differences between the two backup systems, the pilots mostly could not distinguish between them.

In contrast to the primary systems, the pilot ratings for the backup systems were dependent on the level of atmospheric disturbances; most of the ratings indicated Level 1 handling qualities in 'low', and Level 2 handling qualities in 'high' level disturbances, ie their overall rating was Level 2.

The reason for the Level 2 handling qualities rating was not the dynamics of the backup systems themselves, but merely the change in augmentation features at FCS failure.

Although the pilots mentioned difficulties in performing precise flare and landing with the backup systems, the performance measures did not indicate a degradation in touchdown performance in comparison with the primary systems.

Displays:

The way of displaying the necessary information to the pilots certainly influenced their ratings for the handling qualities of the aircraft.

Although no attempt was made to design a complete HUD, the value of a HUD in a transport aircraft was confirmed by all pilots.

Sidestick:

When the FCS provided hold terms, pilots tended to steer in separate axes in a pulse-like manner, which provoked the possibility of accidental cross-coupling and making unintended inputs.

Cross-coupling, due to a poor harmonization between the pitch and the roll channels, also influenced the handling qualities ratings in the present study.

6 ASSESSMENT OF EXISTING CRITERIA

General

The flight control systems were assessed against the predictions of selected existing handling qualities criteria (Table 1), although the available criteria do not give consideration to the flightpath rate command laws used in this investigation, or to the changeover effects. The criteria also do not consider autothrottles, or terms such as bank compensation. It has to be remembered also that the control strategy of introducing pulse type stick inputs, applied mostly by the pilots with the primary FCS because of its 'hold' terms, is different from the continuous tight control behaviour underlying most of the criteria.

As noted, the pilots' ratings of the handling qualities of the control systems were influenced by the HUD format. Up to now the kind of display has not been taken into account by any criterion, leading to some uncertainties when comparing the ratings obtained with the predictions provided by applying these criteria.

Primary flight control systems:

System A flightpath mode

The pilots rated system A in the Level 1 to 2 handling qualities region. However, when the pilot ratings indicated Level 2, the reason given for the degredation was the redatuming of the flightpath command and its symbol on the HUD, a factor which is not considered by any criterion. This implies that those handling qualities criteria, based on pitch attitude and pitch rate responses, which predicted Level 2 are clearly not applicable to aircraft with flightpath rate command/flightpath angle hold control laws. While most criteria under consideration predict Level 1, the Bandwidth Criterion and the Equivalent Time Delay requirement of the Transfer Function Criterion both predict Level 2. The pilot ratings, which indicate that the Equivalent Time Delay of this control system is in fact acceptable for a transport aircraft, are consistent with results from other investigations in the US and Europe that for transport aircraft this boundary should be moved (see the following section).

System B flightpath mode

As noted system B was a near-optimal FCS for the kind of aircraft under consideration. The fact that all pilots rated Level 1 handling qualities implies that the criteria which predicted Level 2 (Bandwidth Criterion and Equivalent Time Delay) are clearly not applicable to aircraft with flightpath control laws. However, these Level 2 predictions were based on three particular parameters: effective time delay obtained from open loop responses; equivalent time delay determined through equivalent system transfer functions; and bandwidth. The present boundaries for these three parameters are mainly based on results of fighter flight tests in category A flight phases (ie rapid manoeuvring, precision tracking, etc). A relaxation of these requirements for their application to large transport aircraft, is therefore already under discussion. Even though the particular criteria may not be fully valid for aircraft with flightpath control laws, the results from the present experiment may contribute to the establishment of new boundaries appropriate for large transport aircraft in terminal flight phases.

Control system changeover

The case of a failure in the FCS and the changeover to a backup system which has different handling qualities is not covered by any existing handling qualities criterion.

Backup flight control systems

The handling qualities criteria considered sometimes predict significant differences between the backup systems C and D. This does not correspond to the pilot ratings obtained, which indicated that the pilots mostly could not distinguish between the two different backup systems.

Most of the criteria predict Level 1 for these systems, while the overall pilot rating was Level 2. Since the pitch rate command laws of the backup control systems are suited for an assessment against existing handling qualities criteria, the results of the assessments performed do not give any consistent explanation for their Level 2 handling qualities.

However, the loss of control augmentation features was mentioned as the main reason for handling qualities degradation relative to the primary systems, a fact which is not covered by any criterion. Apart from these deficiencies, the handling qualities of the backup control systems were rated Level 1. This can almost be taken as confirmation of the predictions from the application of handling qualities criteria, taking account of the fact that for transport aircraft the lowering of the boundaries of those criteria predicting Level 2 handling qualities is under discussion.

7 PROPOSALS FOR NEW HANDLING QUALITIES MEASURES

Flightpath control

Most existing criteria assume that the system must provide good pitch control in order to permit the pilot to achieve good flightpath control. Ref 3 studies the basis of each of the criteria of Table 1, to consider whether as it stands it is appropriate for systems which give direct control of flightpath, and if not, where possible proposes a logical adaptation which should make it appropriate.

System changeover

Pilot comments on the control system changeover at failure, and the associated transients, were highly dependent on the actual flight condition at the instant of the failure. Consequently it will be extremely difficult to define a single parameter or a combination of parameters on which a measure of handling qualities during changeover transients can be based, but it may be possible to measure the change in handling qualities once the transients are over.

If the existing criteria are applied to the flightpath control primary systems considered here then, although many of the criteria are logically inappropriate for such systems, most of them give results which are consistent with the pilots' ratings. This suggests a possible technique for quantifying the change in handling qualities when an FCS changeover from flightpath to pitch rate control (as well as from pitch rate to pitch rate) occurs. The change could be measured by assessing the magnitude of the changes within individual existing criteria. Further work will be necessary to quantify such measures of handling qualities change, in order to establish sound changeover measures.

As noted earlier, the changeover of either primary system to its flare and landing mode was fully acceptable. This was effectively a smooth change from system A to C or B to D, but with no change of other features, at 'low' speed. Thus a change of this magnitude within any relevant criterion is acceptable.

8 CONCLUSIONS

A prime objective of the investigation was to assess the value of longitudinal handling qualities criteria when applied to a relaxed longitudinal stability transport aircraft controlled through an ACT system. The aim was to establish guidelines which could be applied to: sophisticated primary systems having flightpath as the primary control parameter; associated modes or simpler backup systems with pitch rate control; and the changeover between them.

It must be recognised that existing longitudinal criteria have been developed from systems which in general provide only pitch rate control. Also, most existing criteria have not been developed for a control strategy with pulse-like inputs such as are used when the FCS incorporates hold terms. Furthermore, they do not give consideration to changeover effects, augmentation features like autothrottle and bank compensation, or advanced displays like the HUD used here.

In order to assess the value of existing criteria and proposed measures it has been necessary to estimate what the ratings in the present study would have been if the pilots had been able to disregard the augmentation features and advanced displays. It is considered that the criteria and measures should indicate the following handling qualities.

Level 1-2 for primary system A.

Level 1 for primary system B.

Level 1 for the flare and landing modes of A and B.

Level 1 for the changeover from A or B to their flare modes.

Level 1-2 for backup systems C and D.

Level 2 for the changeover from primary systems A or B to backup systems C or D.

Based on these ratings the following conclusions can be drawn:

> Fourteen of the sixteen existing criteria assessed in this investigation are consistent with the pilots' ratings of the backup systems, and the primary flare and landing systems, provided that the time delay requirements are relaxed. The two which seem less appropriate are the Bandwidth Criterion with its current fighter-based boundaries and the Dropback Criterion.
>
> The philosophy on which all but three (Longitudinal Static Stability, Steady Manipulator Forces in Manoeuvring Flight, and Dynamic Manipulator Forces in Manoeuvring Flight) of the existing criteria are based is inappropriate for assessing flightpath control systems, so they should not be used for this purpose.
>
> However, for application specifically to flightpath control systems, logical adaptations are tentatively proposed in Ref 3 for ten of the remaining thirteen existing criteria which are not appropriate as they stand. Further work will be necessary to establish boundaries between levels.
>
> A technique is proposed for using existing criteria to measure the change in handling qualities when an FCS mode change or reversion to a backup control system occurs. Although existing criteria are in general not appropriate to flightpath control systems, it is suggested that these proposed changeover measures can still be applied when the primary system controls flightpath. Further work on this topic is necessary. However, even with the same handling qualities for primary and backup systems according to existing criteria, handling qualities may be rated quite different by the pilot because of changes in augmentation and displays which the criteria do not recognise.
>
> Criteria are required for autothrottle systems.

9 GUIDELINES

9.1 General

A summary of the main results deduced from the reported experiment follows. It has been extracted from Ref 4 and is sub-divided into some general experimental results in Section 9.2, and the handling qualities design guidelines in Section 9.3. Each generic result or guideline is introduced by a short description of the background that led to it.

9.2 Generic experimental results

With respect to the setup of the experiment, it was concluded from the expressed pilot opinions that it is a viable concept for future FBW transport aircraft to be equipped with a HUD and controlled by means of a sidestick. Also the philosophy of a primary longitudinal FCS as mechanized in the reported experiment was well accepted by the pilots. Therefore:

a. A future fly-by-wire FCS for transport aircraft can satisfactorily be mechanized as flightpath rate command/flightpath angle hold with associated autothrottle and bank compensation features, be controlled by means of a sidestick, and be equipped with a HUD.

Pilots readily accepted the changeover characteristics from the primary system to the flare system below 100 ft and, apart from the loss of sophistication, the changeover to the backup system at FCS failure.

b. The dynamic changeover from an FCS based on flightpath control to a pitch control mode at flare, or to a backup pitch control system at FCS failure, can satisfactorily be mechanized as in the reported experiment: fading from flightpath to pitch control mode at flare, and instantly freezing parameter values and switching to the backup mode at FCS failure, followed by fading from primary to backup signals.

Although force versus displacement characteristics for pitch and roll control with the sidestick were harmonized, the pilots complained about overall harmonization problems due to different response and display characteristics, and different levels of system sophistication.

c. Harmonization between pitch and roll should include all aspects: sidestick response and display characteristics, and system sophistication.

9.3 Guidelines

Future transport aircraft are likely to be equipped with FBW flight control systems with which pilots can directly control flight variables that are only controllable indirectly in most current aircraft. This means that the function of the controls is changed, and that the controlled parameter must be displayed to the pilot. Therefore, in such a situation the assessment of handling qualities of an aircraft/flight control system combination becomes inseparable from control and display characteristics. This leads to the following guideline:

Guideline 1: When assessing handling qualities, the complete system must be considered: including not only control laws, but also the characteristics of cockpit controls and displays.

The existing handling qualities criteria, which have been used here in the assessment of the various flight control systems, only consider the aircraft and control laws, without the characteristics of cockpit controls and displays. Hence the following guideline:

Guideline 2: When applying existing handling qualities criteria, be careful because they do not include the characteristics of cockpit controls and displays.

The existing longitudinal handling qualities criteria have been established for aircraft where the stick controls elevator or pitch rate, and in most cases are based on the pilot exercising inner-loop pitch control in order to obtain outer-loop flightpath control. Although pitch attitude (or rate) remains a very important parameter for pilots judging the handling qualities of transport aircraft (eg passenger comfort considerations), such criteria are not directly applicable to flightpath control systems, even though they may appear to give valid measures of handling qualities levels.

Guideline 3: When applying any handling qualities criterion, care must be taken to ensure that its basis is compatible with the control system being studied. Specific criteria for flightpath control systems must be developed.

One of the investigated primary flight control systems featured discontinuities in the commanded flightpath angle when that was different from the actual flightpath angle at initiation of a longitudinal stick input. This was highlighted by the HUD and was disturbing with intentional inputs, but even more so when accidental cross-coupling from roll inputs occurred, especially in high turbulence.

Guideline 4: Flight control systems should not introduce step changes in demand if the pilot does not put in inputs of this type.

Ref 3 proposes adaptations to the basis of ten existing longitudinal criteria, to make them applicable specifically to flightpath control systems, however, boundaries between handling qualities levels are only presented for one of these proposals.

Guideline 5: The adapted criteria presented in Ref 3 should be applied to flightpath control systems. To ensure good handling qualities where handling qualities level boundaries are currently not available, care should be taken that the parameter values for such systems do not differ too much from the data points for these criteria in Ref 3.

Although the handling qualities of the backup systems were judged good by the pilots, the changeover at failure from the primary to the backup system was the main reason for the degraded handling qualities ratings for the backup systems. The pilots argued that the change in system sophistication was too big in the current experiment (especially mentioned were the loss of turn coordination and autothrottle functions, and the loss of the flightpath vector information in the HUD).

Guideline 6: If a primary FCS failure results in a changeover to a less sophisticated backup system, then the change in sophistication must be limited, as must changes in flight information displayed, to be acceptable to the pilots.

The study also addressed the simpler question of pilots' reaction to the change in basic handling characteristics on changeover to a different control mode, or backup system. Ref 3 proposes that existing longitudinal criteria can provide the basis for measuring such changes in handling qualities. However, limits of acceptability have yet to be established.

Guideline 7: The measurement technique proposed in Ref 3 should be applied to any sudden change in longitudinal FCS characteristics. To ensure good handling qualities, care should be taken that the magnitude of such a change is not substantially larger than indicated by the data points in Ref 3.

During the experiment, the pilots were frequently confronted with FCS failures, and hence were well trained to cope with the change in system sophistication. They were however worried about future day-to-day operation, when failures only occur very rarely and the pilots are likely to be less well trained. This leads to the following guideline:

Guideline 8: Crew training should highlight the consequences of the change in system sophistication at a failure in the primary FCS.

For the primary flight control systems under consideration, which were based on flightpath rate, an autothrottle was imperative. The pilot comments indicated that the autothrottle system used during the experiment was too slow. There are, however, no existing criteria for the design of autothrottles. Therefore:

Guideline 9: When use is made of an autothrottle, this system should be designed according to certain criteria (for which criteria should be developed).

10 RECOMMENDATIONS FOR FURTHER WORK

The recommendations for further work evolved from the less than satisfactory characteristics of the lateral/directional FCS, and the loss of augmentation features and some HUD information at the moment of failure, which may have influenced the results of the longitudinal study. It was proposed that:

a. The present study should be extended, so as to establish a basis for low-speed lateral/directional handling qualities guidelines for transport aircraft. This study should include the effects of reversion to a backup system on failure.

b. A full HUD format and associated head-down displays should be developed as an integrated part of the FCS for use with (a) above.

c. Following the above, a return should be made to a further longitudinal handling qualities study including an improved autothrottle. This work would resolve some of the questions left unanswered by the present study. In particular a broader variation of system parameters would provide the basis for more well defined design guidelines.

The GARTEUR organisation has recently approved a new lateral/directional handling qualities Exploratory Group to establish, with Industry, whether there are clear grounds for a further study to cover some of these items.

In conclusion, at the end of the old Action Group's work, the members considered that their conclusions and guidelines represented a fruitful cooperation between research institutes of the four European countries involved.

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TABLE 1

EXISTING LONGITUDINAL HANDLING QUALITIES CRITERIA

Criteria using frequency-domain characteristics

- Aircraft transfer function
- Neal-Smith
- Inferred closed-loop
- Bandwidth
- NLR precision flightpath control
- Attitude/flightpath consonance

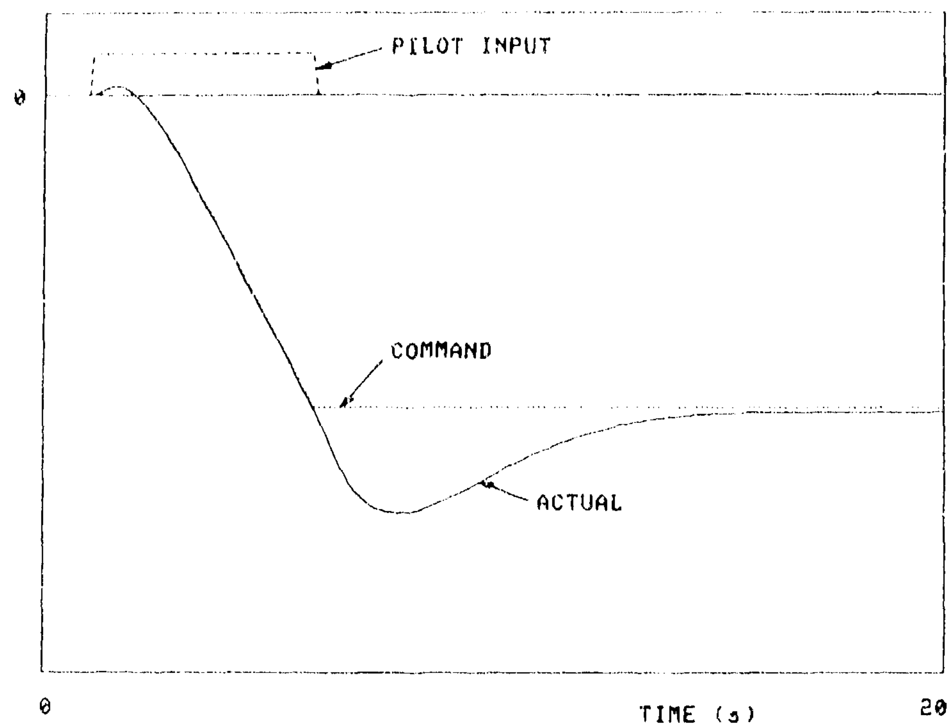
Criteria using time-domain characteristics

- Aircraft open-loop time response
- C*time-history envelope
- Large supersonic aircraft
- Shuttle pitch rate time-history envelope
- Dropback
- NLR rise-time and settling-time

Miscellaneous criteria

- Longitudinal (speed) static stability
- Steady manipulator forces
- Dynamic manipulator forces
- Steady manipulator forces vs pitch acceleration

SYSTEM A



SYSTEM B

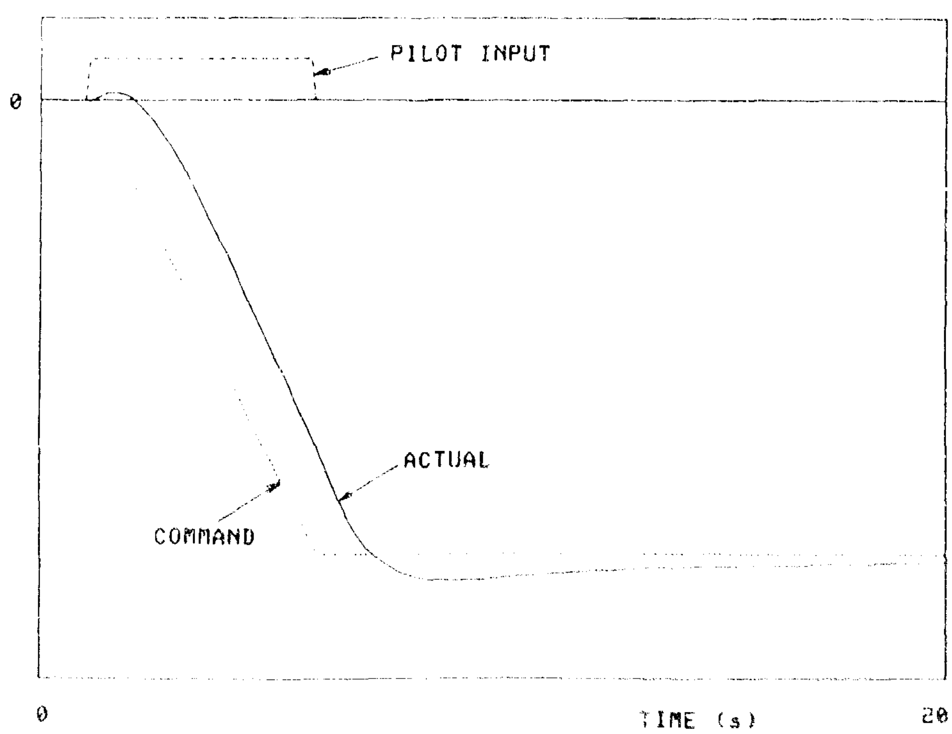
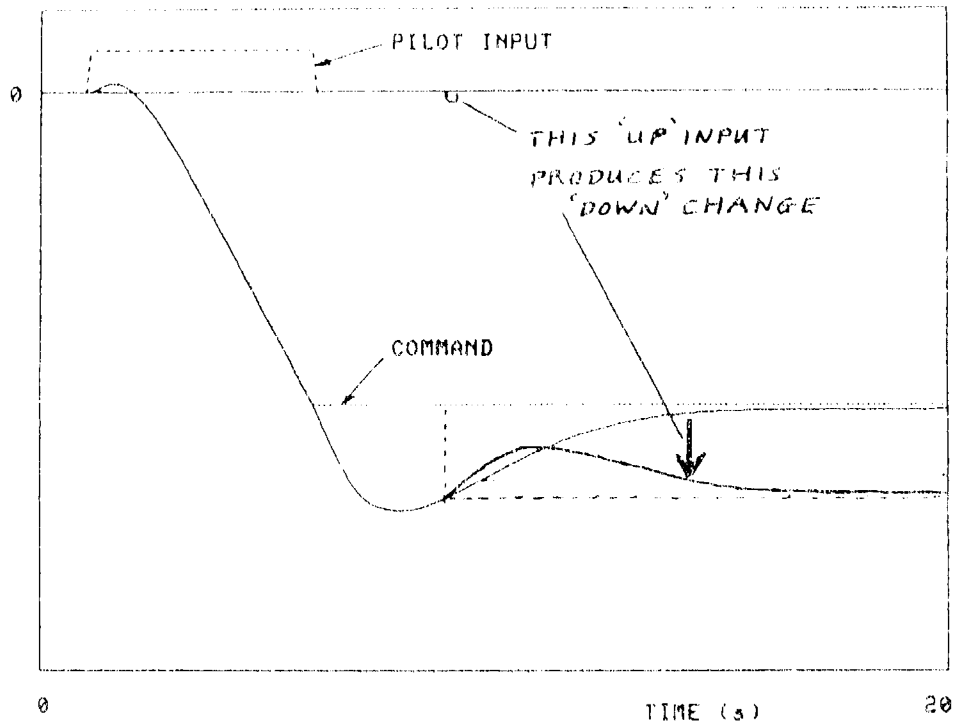


FIGURE 1 FLIGHTPATH RESPONSE OF SYSTEMS A AND B

SYSTEM A



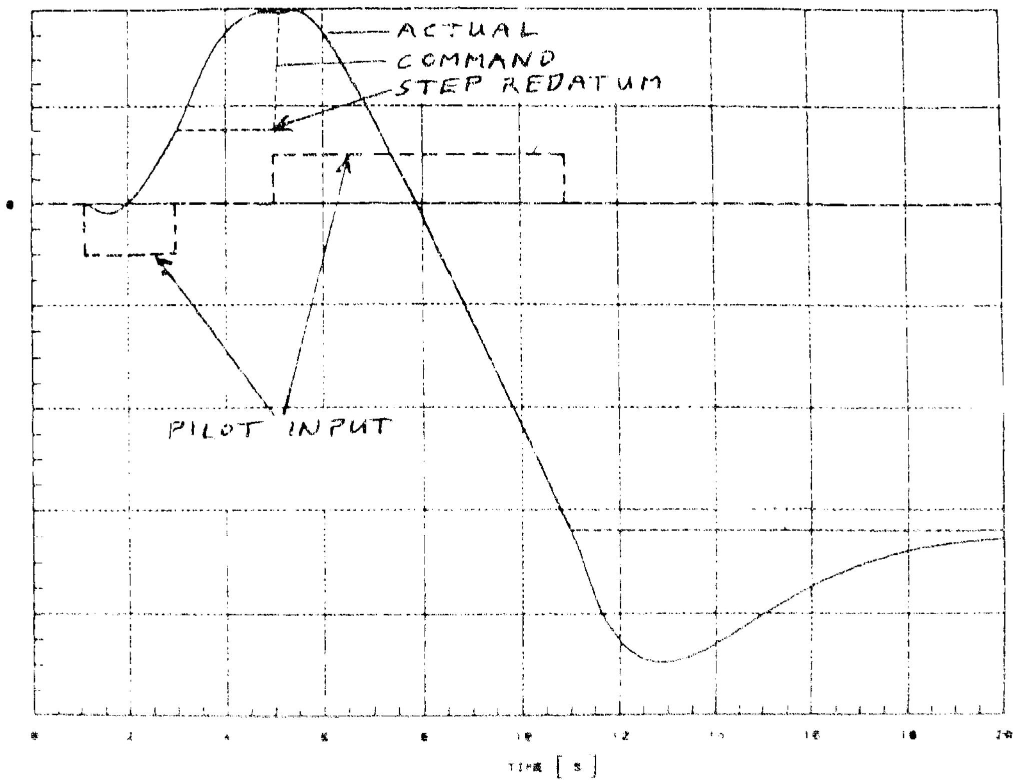


FIGURE 2 FLIGHTPATH RESPONSE OF SYSTEM A

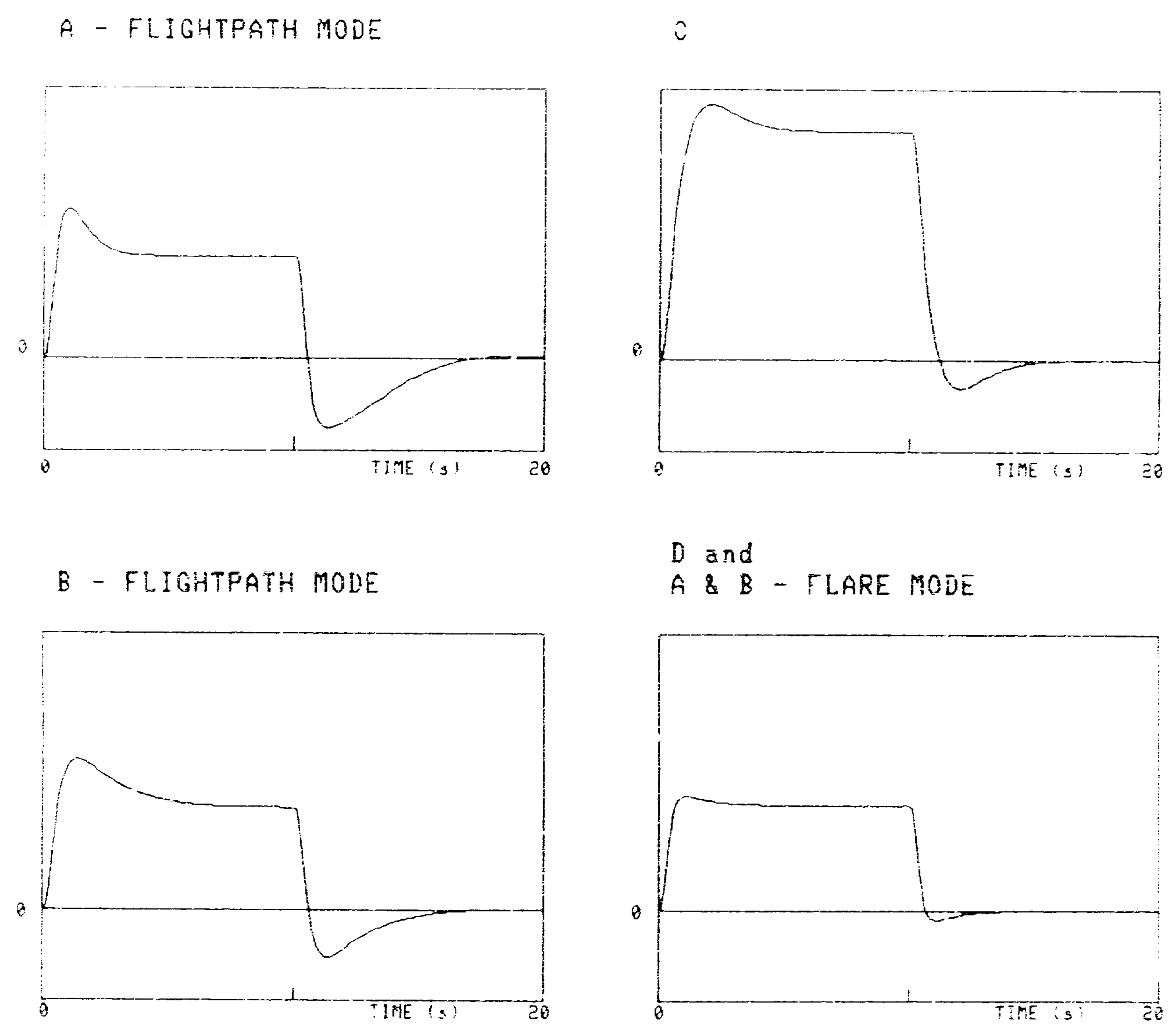


FIGURE 3 PITCH RATE RESPONSE TO BLOCK INPUT

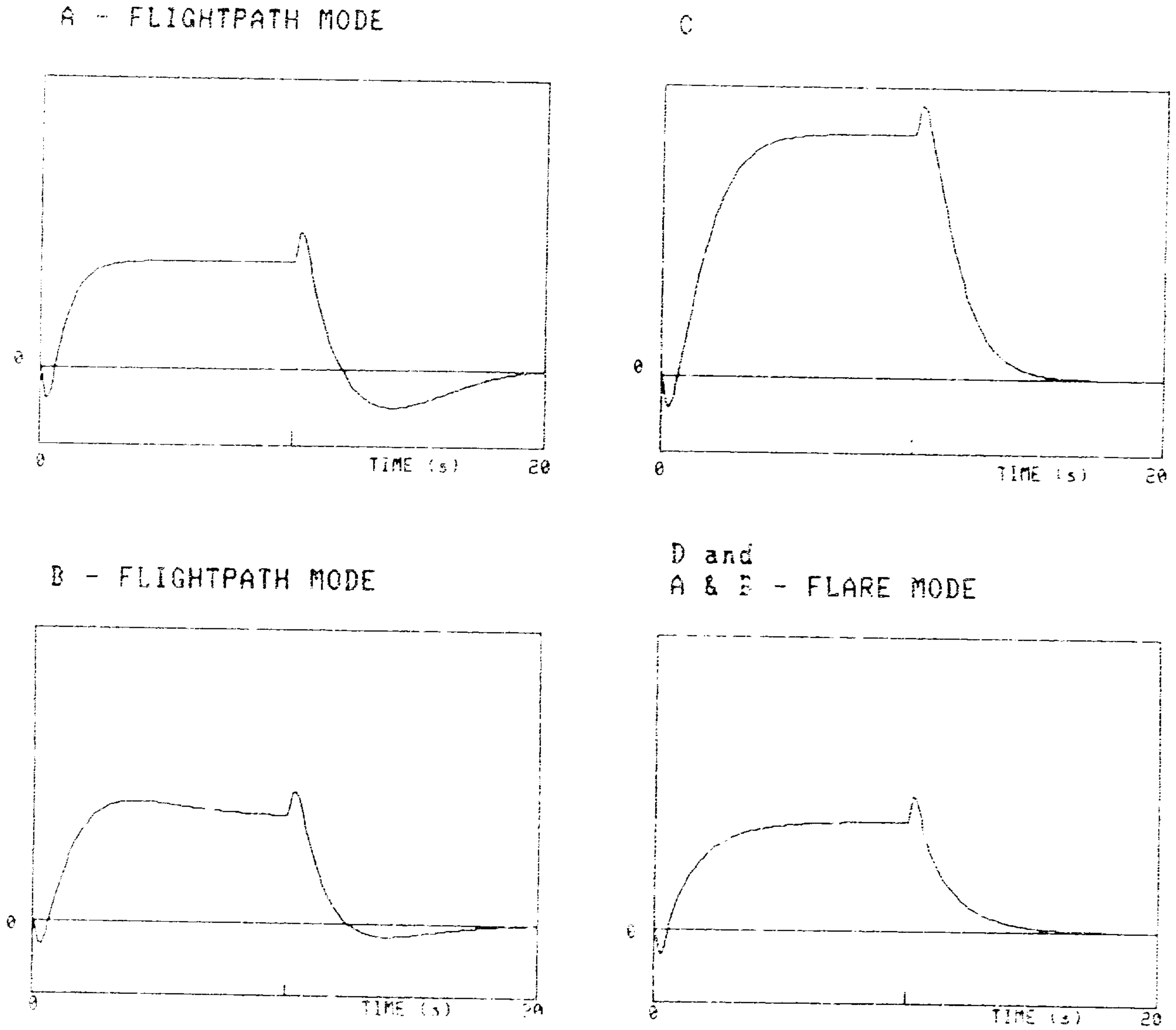


FIGURE 4 NORMAL ACCELERATION RESPONSE TO BLOCK INPUT

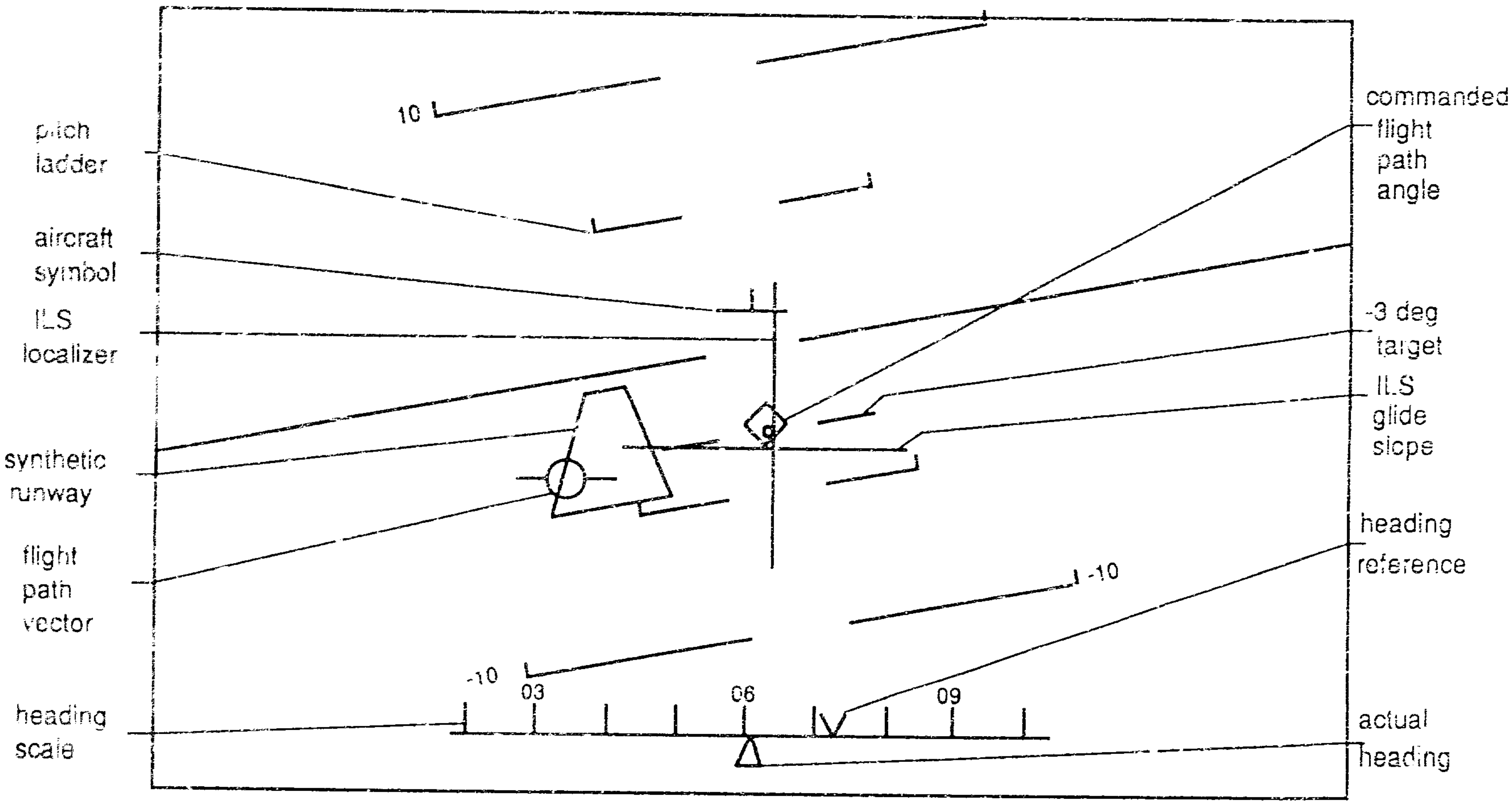


FIGURE 5 HEAD-UP DISPLAY

THE FLYING QUALITIES INFLUENCE OF DELAY IN THE FIGHTER PILOT'S CUING ENVIRONMENT

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SUMMARY

Flight testing has amply demonstrated the serious flying qualities deficiencies that can occur from excessive control system delay. Delay outside of the control system, yet within the pilot's cuing environment, can be potentially as deleterious as control system delay effects. This paper summarizes the results of flight tests to evaluate the effect on flying qualities of time delay in the pilot's cuing environment introduced outside the flight control system. These delays were introduced in the tactile cuing, head-up display visual cuing, and the motion and visual cuing during simulation of fighter aircraft.

NOMENCLATURE

ALT	Altitude
F_{as}	Roll stick force
F_{es}	Pitch stick force
FS	Feel system
HUD	Head-up display
KIAS	Knots, indicated air speed
km	Kilometer
K_{PF}	Prefilter static gain value
K_{FS}	Feel system static gain value
L'_{δ}	Primed roll control effectiveness stability derivative
mils	Milliradian
msec	Millisecond
n_z	Aircraft normal acceleration
p	Aircraft roll rate
$\dot{p}$	Aircraft roll angular acceleration
PF	Prefilter
q	Aircraft pitch rate
rms	Root-mean squared
SP	Short period
θ	Aircraft pitch attitude
θ_{cmmd}	Pitch attitude command
θ_{error}	Pitch attitude error
ϕ	Aircraft roll attitude
ϕ_{cmmd}	Roll attitude command
ϕ_{error}	Roll attitude error
τ_e	Equivalent time delay
τ_{eff}	Effective time delay
ζ	Damping ratio
ω	Natural frequency, rad/sec

INTRODUCTION

The influence of time delay on fighter flying qualities has been a subject of considerable concern because the effects are so deleterious. Flight test results of the YF-16, F-18, and Tornado aircraft (Reference 1) have shown that delay in the path between pilot control input and aircraft response creates imprecision and unpredictability in the pilot's ability to control the vehicle. The outcome is increased pilot workload for control and, if severe enough, pilot-induced oscillations with their concomitant potential for disaster.

Research has been undertaken to understand the effects of control system time delay and higher order control system effects, in general (Reference 2 and 3). Flight control system design criteria, governing allowable control system delay levels, have evolved (Reference 1, 4, and 5). These criteria, however, are by no means complete or comprehensive. For instance, considerable disagreement exists as to whether time delay criteria such as those of MIL-F-8785C or MIL-STD-1797 are applicable to all classes of aircraft (i.e., both fighter-type and transport-type aircraft) since data substantiating the time delay requirements were primarily developed from fighter-class aircraft. The MIL-F-8785C (and MIL-STD-1797) requirements for time delay are:

LEVEL 1: $\tau = 100$ msec
LEVEL 2: $\tau = 200$ msec
LEVEL 3: $\tau = 250$ msec

Despite the understanding of control system time delay effects that has been evolving, available design criteria and research are inadequate or inappropriate when the delay is not resident in the control system. These other sources of time delay can occur in the motion, visual, or tactile cue feedbacks to the pilot. This can be illustrated using the pilot-vehicle dynamic system diagram of Figure 1 (Reference 10). The pilot is the controller of the augmented aircraft in this closed-loop system. Control system delay equally affects both the motion and visual cues feedback to the pilot. Delays outside of the control system but within the pilot's cuing environment may affect only one feedback element, yet its influence can be as deleterious as control system delay effects. For instance, delay is introduced by the display processing requirements for head-up displays, helmet-mounted displays, and others. The capabilities and sophistication of these devices increase their importance in mission success and thus, enhance their predominance in the pilot cuing environment. Therefore, the delay in the visual cuing response introduced by these devices can significantly impact flying qualities as it affects the pilot-vehicle dynamic system. However, design criteria, governing control system delay requirements, are not appropriate for the delay requirements for visual-only, motion-only, and tactile cuing feedbacks. Application of control system delay criteria in these instances can severely and incorrectly impact a vehicle design.

In this paper, the results of several research programs are presented where the influences of delay or high order dynamics in the pilot cuing environment were evaluated. These data and analyses provide guidance into the effect that delays introduced by elements other than the control system have on fighter aircraft flying qualities.

The definitions of time delay in this paper are "equivalent" delay measures; that is, the delay components include both pure digital delay (i.e., from digital computing elements) and "high order" phase delay (i.e., the delay is caused by phase lag added to the "nominal" response). This concept follows the basis provided in Reference 1. The term "equivalent" time delay (τ_e) refers to the sum of the digital and high order delays measured in the frequency domain using a technique such as that proposed in Reference 4. "Effective" time delay (τ_{eff}) refers to the sum of the digital and high order delays measured in the time domain by the maximum slope intercept method (Reference 1). These two metrics should yield similar delay quantities for the same system although important differences arise that demand attention. In either case, these measures attempt to quantify the delay between pilot control input and the response perceived by the pilot.

FEEL SYSTEM INFLUENCE ON FIGHTER AIRCRAFT FLYING QUALITIES

In the F/A-18 and X-29A aircraft developments, important design questions were raised as to the role that the pilot's tactile cuing played in the aircraft flying qualities. In particular, the questions concerned the significance that the delay introduced by the primary cockpit controller feel system (a centerstick in both cases) had on the aircraft handling qualities. Data generated previous to these aircraft developments by such research aircraft as the USAF variable stability NT-33 employed feel system designs that were of sufficiently high frequency and damping that the position response of the controller to a pilot-applied force input was essentially instantaneous. Consequently, the displacement of the controller was dynamically "transparent" to the pilot and unobtrusive to this control actions. Control system time delay criteria developed from data produced by the vehicles, indicated that the delay should be measured from control force input to aircraft angular rate response. Thus, the delay introduced by the feel system dynamics should be included in the overall time delay "account." This procedure was adopted in MIL-F-8785C (Reference 8) with a maximum of 100 msec allowable for Level 1 flying qualities.

The early design of the F/A-18 aircraft utilized a force command control system architec-ture. The primary advantage of this command type is its immunity to loss of aircraft control due to damage or immobilization of the cockpit controller. In addition, this architecture theoretically reduces the overall delay between control force input and aircraft response since the feel system dynamic elements are placed in a parallel instead of serial path between pilot control input and the aircraft control command. This difference in architecture is illustrated using the simplified schematic diagram for a roll flight control system in Figure 2. A pilot-applied step force command is input to both the force and position command control system architectures. For these same step inputs, 90 msec more effective time delay results from a position command system compared to a force command system.

This simple example does not, however, consider the practical implementation of a force command system and, as demonstrated in the F/A-18 development, contradicts this hypothetical time delay reduction. The force signal sensed at the controller is inherently more noisy than the position signal due to transducer differences and the mechanical smoothing that occurs from the feel system spring, mass, damper mechanics. The F/A-18 force signal, therefore, required significant filtering to attenuate noise and, combined with forward path structural notches filters on the stick input, the "equivalent delay" due to these "high order" filters, more than offset the delay produced by the feel system dynamics. The F/A-18 actually changed from a force command to a position command control system architecture (Reference 11) in an effort to reduce the overall response lag between pilot control input and aircraft response.

The X-29A control law mechanization utilizes a stick position command architecture. In the X-29A design, the question of whether the feel system should be included in the time delay account for MIL-F-8785C requirements took special significance because the feel system natural frequency was only 13 rad/sec. With a 0.7 damping ratio, over 100 msec of equivalent time delay is contributed by the feel system alone in an equivalent system analysis between aircraft response and pilot-applied force input. Consequently, Level 1 delay requirements could not possibly be met with the X-29 feel system design. During the design stage, this situation created considerable turmoil. In the end, X-29A flight test results did not reveal any flying qualities deficiencies that were related to excessive delay between pilot control input and aircraft response (Reference 13).

These two designs highlight the powerful and unique role that the tactile cuing provided by the feel system response feedback to the pilot plays in aircraft flying qualities. The feel system provides vital information feedback to the pilot regarding his control actions, yet it also acts as the interface between the pilot and vehicle control system. This interface is unique, however, as both the input (force) and output (position) of this element are directly sensed by the pilot. To further investigate the effect of the feel system on flying qualities, the USAF variable stability NT-33 was used as the testbed for an in-flight flying qualities research program evaluating the influence of the feel system on lateral fighter flying qualities.

The NT-33A aircraft was modified by Calspan and is now operated by Calspan under USAF contract as an in-flight simulator. The vehicle is an extensively modified Lockheed T-33 jet trainer. The evaluation pilot, who sits in the front cockpit, controls the aircraft through a standard centerstick or sidestick and rudder pedal arrangement. The front seat control system of the NT-33A has been replaced by a full authority fly-by-wire flight control system and a variable response artificial feel system. A fully programmable head-up display (HUD) is installed in the front cockpit. The system operator in the rear cockpit, who also acts as safety pilot, controls the simulated HUD and aircraft configuration characteristics.

To meet the test objectives, variations were made to the generic roll flight control system architectures shown in Figure 2. Evaluations of fighter-type (Class IV) roll flying qualities were made with variations primarily in:

- Augmented Aircraft Configuration (combinations of τ_R, L'_δ)
- Control System Command Architecture (position or force command)
- Feel System Dynamics (ω_{FS} = 26, 13, or 8 rad/sec)
- Control System Prefilter (ω_{PF} = 26, 13, or 8 rad/sec)

Evaluations primarily utilized a centerstick controller although some evaluations with a side-stick controller were conducted. The augmented aircraft roll mode time constant variations spanned the lower Level 1 range of fighter aircraft values (τ_R = .45 to .15 sec). The effective command gain (L'_δ) was adjusted to maintain a steady-state roll rate-per-unit roll stick force for each τ_R variation. A linear command gradient was used exclusively with essentially no breakout or hysteresis force in the feel system. Nominally, a 4 lbs/inch roll force-deflection gradient was simulated.

Evaluations were conducted in power approach (Flight Phase Category C) and up-and-away (Flight Phase Category A) tasks. Up-and-away evaluations included formation flying, gun tracking, and computer-generated HUD-displayed roll attitude compensatory tracking tasks. The nominal flight condition was 280 KIAS (518 km/hr indicated) at 7500 ft (2290m) altitude. The power approach task was a visual approach with a 300 ft (91m) lateral offset and the lateral line-up correction initiated at 200 ft (61m) AGL (above ground level) concluding with a flared landing to touchdown. Stringent desired and adequate performance standards were established for each task. Details of the experiment and its results are contained in Reference 12.

Three engineering test pilot's served as evaluators. Based on their pilot rating and comment data, the feel system characteristics and the tactile cuing were evaluated as being somewhat, but not completely, independent of the configuration response characteristics. This could occur because both the input and output of this "system element" were sensed directly by the pilot and these characteristics potentially dominated the aircraft flying qualities irrespective of the aircraft roll response due to these control inputs. The influence that these characteristics had on flying qualities was often "separated" from the augmented aircraft roll response flying qualities. (The evaluation pilots had no knowledge of the configurations they were evaluating nor of whether a force or position command control system was being employed.)

With this result, the feel system influence could be described almost independently of the augmented aircraft configuration or control law architecture. For instance, pilot rating data for three configurations are shown in Figure 3. The 26 rad/sec feel system was essentially "transparent" to the pilots; that is, the dynamic response of the stick position due to a force input was not consciously noticeable nor obtrusive to his control actions. With a 0.7 damping ratio maintained, a centerstick feel system frequency change from 26 rad/sec to 13 rad/sec was occasionally, but not often, noticeable by the pilots and/or influencial of the aircraft flying qualities. The 13 rad/sec frequency was apparently just at the threshold above which the stick dynamics are transparent and below which the dynamic stick response became noticeable. Whether the 13 rad/sec feel system was noticeable and influenced flying qualities depended upon the required control actions for the task, the aircraft configuration, and to a certain degree, on the individual piloting control technique. If large amplitude or rapid inputs were employed, the feel system dynamics were noticed, and degraded flying qualities could occur due to the poor controller feedback cues. For feel system frequencies below 13 rad/sec, (i.e. 8 rad/sec), the feel system was almost always noticed by the pilots and the feel system characteristics significantly influenced the closed-loop pilot-vehicle dynamic system. The pilots could often tell that the stick dynamics were slow or sluggish, although sometimes the pilots just noted a peculiar aircraft response to control inputs. The 8 rad/sec stick feel system effect was sometimes described as a "bobweight effect" in the stick itself. This characteristic was typically rated as Level 3 (deficiency that requires improvement) although it was rated as a Level 2 characteristic by some pilots. Significant scatter in pilot ratings occurred for these slower feel system configurations since there was an apparent dependency on whether the feel system dynamics were more noticeable for individual piloting technique.

One pre-experiment hypothesis was that a slower, more sluggish feel system could physically filter the pilot's control inputs and hence, ameliorate the poor up-and-away flying qualities characteristics of a high roll damping (short roll mode time constant), high roll command authority configuration. As shown in Figure 3 in a force command architecture, the 13 rad/sec feel system produced little change in roll flying qualities from the 26 rad/sec feel system configuration. With an 8 rad/sec feel system, a significant degradation in flying qualities was produced and "roll ratchet" tendencies were exacerbated rather than ameliorated. In Figure 4, tracking task time histories for the same aircraft configuration with different feel system natural frequencies are compared. The tracking task was compensatory roll attitude tracking displayed on the HUD. The PIO or roll ratchet tendencies with the slow feel systems are clearly more pronounced. This example illustrates the importance of tactile cuing. A decrease in feel system frequency, introduced in a dynamic element outside of (parallel to) the direct command path from pilot input into the control system, degraded handling in the pilot's ability to control roll.

Using these data and others, the influence of delay or high order dynamics in the pilot's tactile cuing was examined with respect to the flying qualities design and flying qualities specification problems. Specifically, the flying qualities effects of higher order dynamics or "equivalent delay" in the tactile cuing to the pilot were evaluated using several position command control system configurations which exhibited identical transfer function characteristics between aircraft roll rate response and pilot stick force input. However, these configurations were very different in where the dynamic elements or equivalent delays were located throughout the control system. The flying qualities results, summarized in Figure 5, highlight important design considerations that are <u>not</u> always reflected properly by their equivalent systems representations. These results also highlight the uniqueness of the feel system element in this control system. For instance:

- The "baseline" configuration exhibited Level 2 flying qualities. Adequate or even desired task performance could be attained but the roll response was overly abrupt and objectionable. Moderate pilot compensation was required for desired or adequate performance so Cooper-Harper pilot ratings of 5 or 4 were given. When measured for a stick force input, this configuration exhibits 100 msec of equivalent time delay (equal to the MIL-F-8785C Level 1 upper limit for allowable delay).

- When an 8 rad/sec feel system was used, Level 2 and Level 3 pilot ratings were given to reflect the general degradation in handling qualities. Objectionable stick characteristics predominated the ratings by one pilot whereas another pilot did not object to the feel system change. The equivalent time delay for this configuration is 220 msec when measured from a stick force input. This configuration, just as the baseline configuration, exhibits only 40 msec of equivalent delay if the transfer function model uses the position signal as the input instead of the stick force signal.

- The addition of 110 msec digital delay to the baseline configuration created PIO-prone flying qualities characteristics. The abrupt initial response of the baseline configuration was compounded by initial delay. Level 3 flying qualities ratings were given accordingly. This configuration has nearly the same equivalent delay from stick force inputs as the previous configuration but much more delay when measured from the stick position input. Hence, this delay is downstream of the pilot's direct perception.

- A fast feel system combined with a moderate prefilter lag (ω_{PF} = 13 rad/sec) provided some smoothing of the abrupt initial response of the "baseline" configuration. However, the delay that this filter introduces negates its benefits and Level 2/Level 3 ratings were given. This configuration exhibits the same equivalent delay measured from either the force or position signal input as the previous configuration but the flying qualities are very different.

- The interchange of the feel system and flight control system dynamics from the previous situation created PIO tendencies similar to the addition of 110 msec of pure digital delay. The 26 rad/sec prefilter frequency does not attenuate any of the configuration roll abruptness yet it introduces detrimental phase lag or "equivalent" delay between the control position input and aircraft response. The 13 rad/sec feel system is on the edge of being "transparent" or unobtrusive to the pilot control actions; nonetheless, its dynamic response influence is accessible to the pilot. These last three configurations highlighted significant flying qualities differences yet identical equivalent, low order transfer functions when the input is defined as the force signal.

As this illustration highlights, the "location" of the various dynamic elements in the control laws can produce tremendous flying qualities differences. Descriptions of these flying qualities effects using methods such as the equivalent systems technique are not necessarily adequate to define roll flying qualities. For flying qualities design specification, this result produces a dilemma. MIL-F-8785C (Reference 8) encouraged application of a low-order, equivalent system modeling technique to demonstrate compliance to this delay requirement but the stick force input was exclusively defined as the pilot input for this model. The update of MIL-F-8785C, MIL-STD-1797 (Reference 9), recommends that the force signal be used with a "force" command architecture and the "position" signal should be used for the input definition in a position command system for equivalent systems modeling. The two alternative methods for flying qualities specification are compared in Figure 6. These data were obtained in a visual, lateral offset landing task. Similar results were obtained in the up-and-away tasks, such as those summarized in Figure 5. The correlation of pilot rating data and the two proposed specifications show:

- Using a force input definition for equivalent delay measurement (Figure 6a) exclusively as in MIL-F-8785C results in a conservative requirement in the sense that better flying qualities ratings were predominantly given to configurations that are "predicted" to be worse than actual by the specification requirement. The primary source of this discrepancy is the slower feel system dynamics configurations in a position command system architecture. These systems exhibit commensurately high equivalent time delays but without a "significant" flying qualities penalty.

- Using a force or position input definition depending upon the control system command type (Figure 6b) as in MIL-STD-1797, provides a better correlation of the "predicted" and actual Level 2 and Level 3 configuration flying qualities. However, numerous configurations are "predicted" to be Level 1 but in actuality, exhibit worse flying qualities because the influence of the slowest feel systems on flying qualities is not accounted for in the requirements. The requirement focuses only on delay introduced by the flight control system.

Including and treating the feel system in a manner analogous to a control system element, such as proposed in MIL-F-8785C, is misleading to a designer and potentially erroneous for specification. These data show that the "accessibility" of the feel system by the pilot is unique. F/A-18 and X-29A flight test data substantiate this conclusion. Requirements or design criteria for tactile cuing are, nonetheless, required. For instance, in Reference 14, a technique to account for the flying qualities influence of the feel system was proposed by simplistically modeling the pilot neuromuscular system. In the analysis, an equivalent limb/manipulator dynamic system and hence, an equivalent time delay contribution was computed and when used, showed good correlation to MIL-F-8785C allowable delay requirements.

Additional requirements for tactile cuing and flying qualities are warranted to supplement the requirements of MIL-STD-1797. The NT-33 research was limited in scope, since only second-order feel system dynamics were evaluated and damping ratios were maintained at 0.7. For instance, a minimum allowable natural frequency of 13 rad/sec could be proposed from these results, however, this may be overly restrictive. An alternative requirement is desirable because a minimum frequency requirement may be inappropriate. For instance in Reference 15, it was concluded that considering only the natural frequency and damping ratio of the feel system dynamics results in an adequate quantification of their effects on handling qualities. This reference suggest that the physical parameters (inertia, damping and spring gradient) of the stick should be specified since they directly relate to the forces the pilot feels. Forces for abrupt stick movements result from inertia, forces for large rapid movements result from damping, and forces required to hold constant deflection result from the spring gradient. Thus, the pilot can perceive the independent effects of inertia, damping and spring gradient.

Finally, portions of the NT-33 in-flight research program were repeated in a fixed-based ground simulation (Reference 16). The comparison of these results to the in-flight experiment vividly highlight the problems with ground-based simulation "answers" for fighter aircraft flying qualities. From the ground-based simulation experiment, the following conclusions were drawn:

- "Sensitivity (of flying qualities degradation) to τ_e (equivalent time delay) decreased as (roll) command gain increased."
  - During the in-flight experiment, the sensitivity of flying qualities degradation to τ_e decreased as roll command gain <u>decreased</u>.
- "The distribution of principal lag between feel and command prefilter has essentially no influence on flying quality rating."
  - In-flight, the lag location significantly changed the configuration handling qualities.

Finally, this in-flight/ground simulation comparison is further highlighted by considering that the "baseline" roll configuration was evaluated as Level 1 in the ground simulation but was evaluated in-flight to be Level 3 with excessive roll acceleration to pilot control inputs with high frequency PIO (roll "ratcheting") tendencies.

EFFECT OF DELAY DURING SIMULATION

The vivid differences between ground-based simulation and in-flight evaluation of fighter flying qualities have been learned through aircraft developments and loose replications of in-flight experiments in ground-based simulation facilities, such as the one discussed above (e.g., see References 16-18). These experiences and experiments highlight ground simulation deficiencies but, because of the many differences between the ground and in-flight studies, these works do not sufficiently pinpoint the pilot-vehicle dynamic system differences. To help focus this understanding an extensive in-flight and ground simulation study was performed using the NT-33 as a flight vehicle and also as a ground simulator cab. In this way, the cockpit environment, feel system, and field-of-view were duplicated for both a fixed-based, no-motion ground simulation and in a full-motion, in-flight simulation. Also, by simulating instrument meteorological conditions and using the head-up display for evaluation task generation, the visual cues were replicated between ground simulation and flight. Consequently, the vast majority of the pilot-vehicle dynamic system elements were duplicated. Time delay was introduced into the two simulation conditions to investigate its influence on flying qualities and simulation fidelity with and without motion cues.

For this program, a broad-based experimental approach was used. The NT-33 simulated four generic aircraft "types" ranging from a highly responsive, fighter configuration to a less responsive, transport aircraft configuration. Evaluation tasks were fashioned in concert with the aircraft type and hypothetical missions. The results of the fighter aircraft evaluations are presented here, while details of the experiment and the results for the other aircraft evaluations are contained in Reference 19.

The fighter aircraft characteristics were designed to be highly agile yet, to produce Level 1 flying qualities in the in-flight environment with no control system delay added experimentally. The lateral-directional characteristics of the configuration were tailored to be good and unobtrusive; thus, "feet-on-the floor" roll maneuvers could be performed. This allowed the pilot to concentrate on pitch and roll control without objectionable yaw or sideslip. The centerstick feel system dynamic characteristics were of high frequency with a force command control system architecture in both pitch and roll.

The experimental set-up is sketched in the schematic diagram of Figure 7. The simulated control system was identical for both the in-flight and ground-based simulations. Pure digital delay was added experimentally. The "baseline" condition included no added delay ($\tau = 0$ msec). Delay effects were evaluated in both pitch and roll flying qualities with equal amounts of delay added in both control axes simultaneously.

The fixed-based ground simulation capability was provided by appropriate interfaces to the NT-33A aircraft. The ground simulation utilizes the actual aircraft hardware and cockpit with the exception that the NT-33A motion responses are computed by a real-time computer simulation and are interfaced into the fly-by-wire flight control system through the NT-33A sensor system. Thus, the ground-based NT-33A simulation system is operated the same as it is in flight, with the exception of aircraft motion. The real-time computer simulation operated at an 80 hz update rate. In Table I, the pitch (θ) and roll attitude (ϕ) transfer functions are shown for both the "HUD-displayed" (i.e., visual states) and the "motion" states. Naturally, the motion response states were only available as a piloting cue in the in-flight simulation. The other motion state transfer functions are not shown in Table I as they follow naturally from the tabulated transfer functions. Equivalent systems transfer functions are used to simply represent the configuration response. Nonetheless, the models are good approximations. The 55 msec equivalent delay from pilot stick input to the motion state response occurs from the phase lags produced by necessary high order control system filters and control surface actuators. The attitude response displayed on the HUD lags the motion response by 45 msec due to the sensor, signal conditioning, and digital display processing components. Hence, a total of 100 msec equivalent delay exists between pilot stick force input and HUD-displayed attitude response. In the ground simulation, the same equivalent delay of 100 msec occurs between pilot stick input force and HUD attitude response.

The head-up display was programmed to generate the piloting evaluation tasks. Instrument meteorological conditions (IMC) were simulated using a blue/amber vision restriction system to limit the visual cues available to the pilot. The evaluation tasks were thus repeatable, known quantities that could easily be replicated in ground simulation. The instantaneous field-of-view (FOV) of the HUD and hence, the total visual FOV available to the pilot for task cuing was approximately 20 degrees. Airspeed control was maintained by the rear seat safety pilot so that the evaluation pilot could concentrate on pitch and roll control.

In this set-up, the visual evaluation task in-flight and in ground simulation is identical. Consequently, the primary difference between the in-flight and ground simulation is the absence of motion cuing although, to the evaluation pilots, other differences also exist, such as the absence of aural cues and flight stresses. The in-flight and ground-based simulations were mechanized such that the equivalent time delay between the cockpit control input and the HUD (visual) attitude response was identical during in-flight or ground-based simulation. Without any experimentally added delay, this was a constant 100 msec. During in-flight simulation, the visual (HUD) response lagged the motion response by 45 msec equivalent delay.

Three types of HUD-displayed evaluation tasks were used for this experiment. The format of the HUD and the computer-generated tracking task are presented in Figure 8. The primary evaluation task was step-and-ramp compensatory tracking of pitch and roll attitude display on the HUD. This "discrete" tracking task is shown in Figure 9 where the command bar moved in a series of attitude commands in both pitch and roll to lead the pilot through a coordinated maneuvering flight profile. In addition, compensatory attitude tracking tasks were also performed using a psuedo-random sum-of-sines generated command. Three engineering test pilots served as the initial evaluation subjects although their results were substantiated by data from eight additional engineering test pilots.

The influence of delay on simulator effectiveness and flying qualities can be "measured" by pilot ratings. Using the Cooper-Harper pilot rating (PR) scale (Reference 10), Level 1 flying qualities are defined as being satisfactory without improvement with pilot compensation not a factor ($PR < 3.5$); conversely, Level 2 flying qualities exhibit deficiencies which warrant improvement. Pilot compensation is, at least, moderate and vehicle performance is adequate or desired at best ($3.5 < PR < 6.5$). The maximum tolerable delay introduced in a simulation can be defined as the maximum delay before which flying qualities degrade to Level 2 (given that Level 1 flying qualities existed without added delay). Implicit in this rating difference is a change in pilot control strategy, behavior, and/or workload. The other specific simulation effectiveness measure, transfer-of-training, was not addressed in this experiment.

In Figure 10, the effects of time delay on fighter aircraft flying qualities are shown during full-motion, in-flight simulation and during no-motion, ground-based simulation. Flying qualities are shown using averaged Cooper-Harper pilot ratings, and the extreme ratings that were given for each configuration. The in-flight data are plotted in two ways. In one case, the equivalent time delay was measured from the pilot stick force input to the HUD display response (i.e., the "visual delay"). In the other case, the delay was measured from the same stick input but only to the aircraft motion response (i.e., the "motion delay"). The motion response lead the visual response by 45 msec. Of course, no motion cues were available to the pilot in the ground simulation.

The data of Figure 10 indicates the following:

- As control system time delay was introduced, the flying qualities of the in-flight simulated aircraft degraded at a rate of approximately 1 PR unit for each 30 msec of delay added above a 130 msec threshold. This trend is based on close comparison to the data of Reference 1 using the "motion delay" values. Pilot comments support the pilot rating trends, that is, as the delay was increased beyond 130 msec, PIO tendencies became more prevalent and overshoots in target tracking were more pronounced. Good correlation of the in-flight pilot rating data using the "motion" delay substantiates the importance of the motion cuing influence and the effects of time delay.

- During no-motion, ground-based simulation, the lowest simulated delay configuration was judged as being Level 2. Only adequate tracking performance could be attained although sometimes desired performance was obtained but with moderate or considerable pilot compensation. As delay was added experimentally, flying qualities degraded but at less severity than that demonstrated in-flight. The in-flight/-ground simulation differences were primarily attributed to the different cuing environments.

Tracking performance and hence, flying qualities in the no-motion ground simulation were poor in comparison to the identical configuration simulated in-flight. The lack of motion cues changed the pilot-vehicle dynamic system to such a degree that bobbling tendencies and an inability to steadily track the target occurred even with the lowest values of time delay. Tracking problems were most evident during gross acquisition maneuvers. For large amplitude maneuvers, the lack of "g-cuing" was felt by the pilots to deteriorate their ability to judge target closure rates. In Figure 11, two time histories of step-and-ramp HUD tracking task are overlaid for the identical configuration and evaluation pilot. This configuration only exhibited 100 msec delay between stick input and HUD, "visual" response. The different motion cues, however, are clearly affecting tracking performance. In the fixed-based simulation, significant pitch bobble is evident and tremendous "over-g" is being commanded by the pilot. In-flight, this configuration was rated as a 3 whereas, in the ground-simulation, a pilot rating of 5 was given.

In this program, very good consistency in pilot rating data were achieved both within-subjects and across-subjects. Pilot ratings, of course, reflect pilot-vehicle performance and the attendant pilot compensation or workload to achieve that performance. The consistency in pilot rating data is remarkable particularly when considering the fairly significant differences in pilot control techniques and tracking performance. For example, tracking statistics from the step-and-ramp HUD tracking task are shown in Figure 12 for three values of control system time delay during in-flight simulation. The statistics are the "time-on-target" (the cumulative time that the pitch tracking error was less than 5 mils) and the normalized root-mean-squared (rms) pitch tracking error.

Pilot A demonstrated the best performance (i.e., highest time-on-target and lowest normalized error) with the lowest delay configuration. As delay was added, his tracking performance degraded. The closed-loop tracking performance by Pilot B exhibited a similar degradation but overall his performance was not as good as Pilot A. On the other hand, Pilot C with the lowest delay configuration,was not as precise as Pilot A, but Pilot C was extraordinarily adaptative, in that, as delay was increased his pilot-vehicle performance remained essentially constant. The pilot rating data, despite these tracking performance differences, were very consistent between each subject.

Without motion cuing, degradation in tracking performance occurred and consistency in tracking performance and flying qualities (i.e., pilot ratings) could not be maintained. For instance, the HUD tracking task statistics for the same configuration as previously shown are presented in the ground simulation environment in Figure 13. The normalized rms error, which could be related to pitch bobbling or PIO tendencies, was consistently higher in all cases of ground simulation than for in-flight simulation. Inconsistencies in performance as control system delay was added occurred frequently. For example, Pilot B achieved an increase in time-on-target and a reduction in normalized tracking error in one case with 280 msec of control system delay compared to just 190 msec of delay.

Measurement of the tracking performance degradation with increased control system delay was also made during the sum-of-sines command tracking tasks. The normalize pitch and roll rms tracking errors are presented in Figure 14. The sum-of-sines tasks were not demanding flying qualities evaluation "tasks" but they provided good data from which to analyze pilot control action during random, small amplitude tracking. The data in Figure 14 shows fairly consistent degradation in tracking performance as delay is increased. Motion cuing effects (i.e., the difference between the in-flight and ground simulation environments) produced an almost constant, tracking error performance difference at each value of control system delay. Unlike the pilot rating data for increasing control system delay, the rate of change of the sum-of-sines tracking error with respect to the delay did not change when the motion cues were deleted. The standard deviation measures, particularly for the roll tracking task, are fairly large and reflect the aforementioned differences in individual piloting performance.

EFFECT OF DISPLAY SYSTEM DELAY ON FIGHTER FLYING QUALITIES

The role of the head-up display in fighter aircraft has increased tremendously in recent years. Night and adverse weather mission rely significantly on information provided to the pilot for navigation and situational awareness. This reliance is certainly not going to subside but will undoubtedly increase. Since flight in this scenario is conducted primarily with visual reference to the HUD, the display system dynamic response may be critical to flying qualities. The current U.S. Military standards do not address temporal response requirements (Reference 20) and applicable research has been primarily limited to V/STOL aircraft and their intended missions (Reference 21). Consequently, a research program was conducted using the USAF NT-33 variable stability aircraft and its programmable head-up display.

The HUD was used as the primary flight instrument reference in both visual (VMC) and instrument meteorological conditions (IMC) tasks. Instrument meteorological conditions were simulated using a "blue/amber" system. A display format similar to that of the F/A-18 aircraft was used exclusively. The format, symbology, and content of the HUD are important factors that affect the pilot's ability (speed and accuracy) to process displayed information; however, these were not evaluated because the intent of the program was to determine what influence the temporal distortion of the display information had on pilot-vehicle performance and pilot workload (i.e., flying qualities). Both up-and-away and power approach tasks were investigated. In the power approach task, a conformal runway display symbology was drawn for landing guidance, analogous to the display system developed by Gilbert Klopfstein (Reference 22). Details of the program are contained in Reference 23.

The schematic diagram of Figure 15 outlines the experimental variation. The aircraft flight control was designed for good flying qualities characteristics. The equivalent delay between stick force input and aircraft motion response was 80 msec. (In the position command control system architecture, 50 msec of that delay was due to the centerstick feel system dynamics.) The aircraft "motion" response from stick input commands remained unchanged in this evaluation of display system delay effects.

An irreducible delay of 65 msec was contained in the display system response characteristics. This delay is an equivalent delay composed of analog dynamic elements (such as sensors, signal conditioning filters) and digital processing elements (analog anti-aliasing filters, sampling delays, computational time delay). Pure digital delay was added on top of the 65 msec irreducible delay, rather than sampling effects, under the assumption that multiple processors in a "pipeline" architecture would more likely be used than one processor running at a reduced sample rate. The delay was introduced into the HUD input signals uniformly.

A variety of up-and-away evaluation tasks were flown including simulated gun tracking using a HUD-programmed pitch-and-roll tracking (identical task to that shown in Figure 9) simulated air-to-ground weapons delivery, and acrobatic maneuvers. The power approach task was an IMC approach starting on downwind and concluding with a 50 to 200 ft (15 to 60 m) breakout for a visual landing.

Pilot rating data are shown as a function of the display time delay in Figure 16 for the up-and-away tasks. Averaged pilot ratings and standard deviations were calculated and are used to illustrate flying qualities. A trend line is drawn showing an estimate of the flying qualities degradation with display time delay based on a least-squares fit to the raw rating data.

For the up-and-away configurations, the pilot rating and comment data indicated the following:

1) As the computational delay for head-up display processing increased, flying qualities reflected by the Cooper-Harper pilot ratings (PR) were essentially unaffected until after 250 msec of equivalent "display system" delay. Some degradation occurred below 250 msec of delay but the degradation was not dramatic. Flying qualities degraded markedly beyond the 250 msec display delay value. Level 3 ratings (PR > 6.5) were given for 385 msec of display delay.

2) The overall task ratings were primarily based on the pilot compensation/workload and task performance in the simulated air-to-air gun tracking. To a lesser degree, the ratings were also based on the subjective opinion as to the degree by which the display characteristics were deficient. As display time delay increased, "bounce" in the pitch ladder and lag in the flight-path marker became more noticeable. These deficiencies were noticed primarily during precise control maneuvers or those maneuvers that had clearly defined attitude or flight path performance "end points."

3) The added pure delay in the head-up display did not significantly affect performance or flying qualities for the large-amplitude maneuvering tasks (modified cloverleaf and pop-up weapons delivery). Because precise and aggressive pilot control was not required in these tasks, the effect of display time delay was relatively transparent to the pilot, except for pitch ladder bobbling and flight-path marker lag at the "end points" of these maneuvers.

4) Control was not in question for any of the configurations. Thus, the pilot rating range was essentially limited to a maximum of 7. Added display time delay, unlike added control system time delay, did not evoke pilot-vehicle dynamic system instabilities.

These results suggest that flying qualities are not significantly affected until 250 ms delay exists in the display system. This delay is in addition to the 80 msec of delay between the cockpit control input and aircraft motion response. The 80 msec "motion" delay was held constant and is below the Level 1 MIL-F-8785C or MIL-STD-1797 allowable control system delay requirements.

The pilot rating data for the power approach task are shown in Figure 17 as a function of display time delay. Averaged pilot ratings are plotted with standard deviations about the mean illustrating the rating trends. A trend line is drawn based on these data. These data indicate the following:

1) As display system delay was added, flying qualities degraded after 300 msec of display time delay, as indicated by the rating data. The averaged pilot rating below 300 msec of display time delay was essentially Level 1 (PR < 3.5).

2) The degradation of flying qualities with display time delay was less severe in power approach than up-and-away flight. The pilot comments indicate that the pilot ratings were primarily determined by the pilot workload demands as display time delay increased. Added display delay

amplified the "bouncing" and "wobbling" of the contact analog runway projection, thus, increasing the pilot workload to keep the display centered and to maintain the approach course.

The influence of the evaluation task is evident in both the power approach and up-and-away task data. The power approach evaluation task was the instrument approach with a visual landing made after instrument breakout. The visual delay experiment was, in actuality, only evaluated in the IMC approach phase. After IMC breakout, a visual landing was made without significant reference to the HUD. In Reference 3, it was noted that the effects of added (motion) delay were most dramatic in the final 50 ft (15m) to landing when the pilot's control task becomes critical (high stress). In this power approach task, the effects of delay added to the display were evaluated only on the relatively benign approach course flight task to breakout. Consequently, the flying qualities degradation with display delay would not be expected to be as severe as a more demanding task. Similar conclusions could be drawn from the up-and-away task.

Turbulence was another factor that was significant but it could not be experimentally controlled. In up-and-away flight tasks, turbulence was not typically encountered. In power approach tasks, turbulence was sometimes encountered and turbulence produced a significant degradation in flying qualities when the display delay was greater than 100 msec. The pilot ratings that were influenced by turbulence are not shown in Figure 17. In turbulence, the temporally-delayed display characteristics created significant pilot workload for compensation and objections by the pilot regarding the continual bobbling of the display. Turbulence affects will likely reduce the amount of display delay that is acceptable based on flying qualities. Unfortunately, sufficient data were not gathered to demonstrate this result conclusively,.

The apparent 250 msec threshold in allowable display system time delay for the up-and away evaluations and 300 msec in power approach tasks seemingly contrasts with the previous works from which the 100 msec MIL-F-8785C and MIL-STD-1797 time delay requirements evolved. Control system time delay has been shown to significantly impact the pilot-vehicle dynamic system. Flying qualities have been demonstrated to degrade from Level 1 to Level 2 by 100-150 msec of control system time delay with pilot-vehicle instabilities occurring from 250 msec delay. Much larger delays were tolerated in this program when delay was added to the display system. The significant difference between control system and display system time delay effects can be demonstrated by examining the pilot-vehicle dynamic system.

The results of this experiment indicate that with constant Level 1 motion flying qualities, a substantial threshold exists where added visual delay does not affect flying qualities. It could be deduced because of the different flying qualities effects between control system delay and display system delay that the pilot, as the closed-loop controller in the pilot-vehicle dynamic system, was either unaffected by the visual delay, or he could easily compensate (without workload or performance penalty) for its influences. From a physiological viewpoint (Reference 24), it could be logically defended that the visual cue feedback to the pilot was not a critical cue in task performance in the presence of a visual delay. Since time delay adds deleterious phase lag proportionally to frequency, the high-frequency spectrum of the cue feedback to pilot would be most influenced by delay. However, the visual senses are primarily used during piloting tasks for low-frequency cuing or steady-state reference. Conversely, the motion response primarily provides the high frequency response cue such as onset or acceleration cuing. The pilot blends the motion and visual senses, in a complementary fashion, to provide a full-fidelity, broad-banded frequency response estimate of the aircraft states. Hence, significantly greater time delays in the visual feedback would be required to affect the low frequency visual cue feedback than the delay permitted in the high frequency motion feedback due to the physiological nature of the human pilot. This was shown experimentally. Motion cue effects must, therefore, always be referenced in assessing the influence of display system dynamics.

CONCLUDING REMARKS

The influence of delay outside of the control system but within the perceptual cuing environment of the fighter pilot can significantly affect flying qualities. The influence is, however, different than the effects due to control system delay. The pilot-vehicle dynamic system provides the necessary framework from which these influences can be understood.

Directly and indirectly, the results summarized here imply that motion cues are a critical component in determining the flying qualities influence of the delay and in why control system delay effects are a clearly detrimental flying qualities factor. Control system delay equally lags both motion and visual cue feedbacks to the pilot. The flying qualities degradation with control system delay (i.e., delay in both the visual and motion cuing) was shown to be approximately 1.0 PR unit per 30 msec of delay after a threshold of 130 msec (Figure 10). In the evaluation of display system time delay, flying qualities are not influenced by delay in the visual cue feedback path until over 250 msec of visual delay is added (Figure 16). Motion cues were present in this case but they were unaffected by the experimental addition of delay. In a no-motion, ground-based simulation, flying qualities degrade at less than 1 PR unit per 100 msec of added delay in the visual cue feedback. In this case, the visual cue was essentially the only aircraft response feedback to the pilot. These data suggest that if the motion cues are absent or unaffected by the delay, the effect of the delay on flying qualities will be less detrimental than that shown in flight for control system time delay effects. For the HUD-visual delay in flight and the ground simulation delay evaluations, the motion cues were unaffected by the delay or the motion cues were absent, respectively.

An explanation for the less severe degradation in flying qualities with visual-only delay has a basis in human physiology. Delay in the visual cue is not as deleterious as motion cuing delay since the pilot does not physiologically rely on the visual cue feedback predominantly for high frequency cues. Visual cuing for the pilot primarily provides low frequency, steady-state situational cues, not high frequency acceleration or response onset cues. The effect of time delay is to introduce phase lag proportionally to frequency and distort the response onset cues. Thus, significantly more delay is required in the low frequency, visual cue feedback than in the high frequency, motion cue feedback to produce equal amounts of phase distortion in their respective cues. Consequently, smaller values of delay in the motion cue will produce greater flying qualities degradation than that same delay added to visual cues.

Similarly, delay or high order dynamics in the fighter pilot's tactile cuing may significantly influence flying qualities but the analysis is different than a control system element. The tactile cuing evaluations investigated variations in the natural frequency of the second-order feel system dynamic system. The same dynamic elements, when placed in the control system, produced different flying qualities effects since the pilot has "perceptual access" to both the input (force) and output (position) of the tactile feel system cue and the dynamics of the feel system. Time delay criteria and equivalent systems modeling cannot be used blindly and without regard to the location of the dynamic element since these play critical roles in the definition of a vehicle's flying qualities.

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Table I
Fighter Aircraft Configuration Characteristics

IN-FLIGHT SIMULATION:

"HUD-DISPLAYED" STATES		"MOTION" STATES	
(θ_{HUD}/F_{es})	(ϕ_{HUD}/F_{as})	(θ/F_{es})	(ϕ/F_{as})
$\frac{.25(1.25)e^{-.10s}}{(0.)[0.7;6.3]}$	$\frac{.56e^{-.10s}}{(0.)(2.85)}$	$\frac{.25(1.25)e^{-.055s}}{(0.)[0.7;6.3]}$	$\frac{.56e^{-.055s}}{(0.)(2.85)}$

GROUND-BASED SIMULATION:

"HUD-DISPLAYED" STATES		"MOTION" STATES	
(θ_{HUD}/F_{es})	(ϕ_{HUD}/F_{as})	(θ/F_{es})	(ϕ/F_{as})
$\frac{.25(1.25)e^{-.10s}}{(0.)[0.7;6.3]}$	$\frac{.56e^{-.10s}}{(0.)(2.85)}$	NONE	NONE

Nominal Flight Conditions: 250 KIAS, 7500 ft ALT (463 km/hr, 2286 meters ALT)

Short Hand Notation Used: $(a) = 1/a(s+a)$; $[\zeta;\omega] = 1/\omega^2[s^2 + 2\zeta\omega s + \omega^2]$

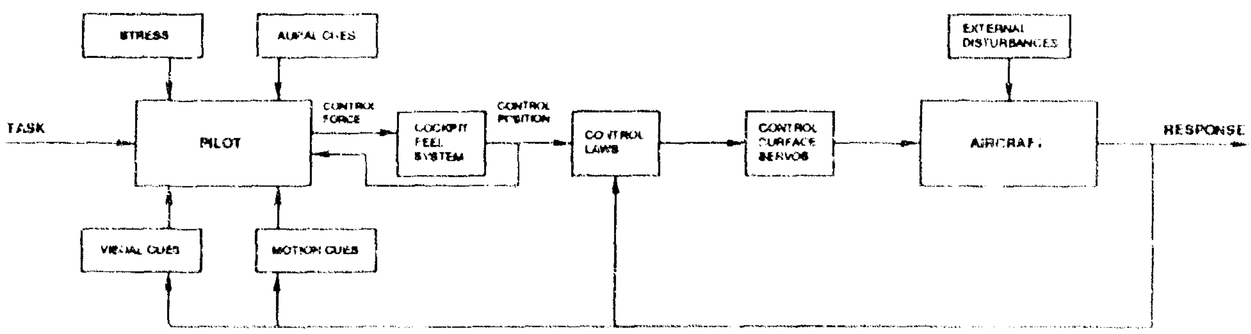


Figure 1 Pilot-Vehicle Dynamic System

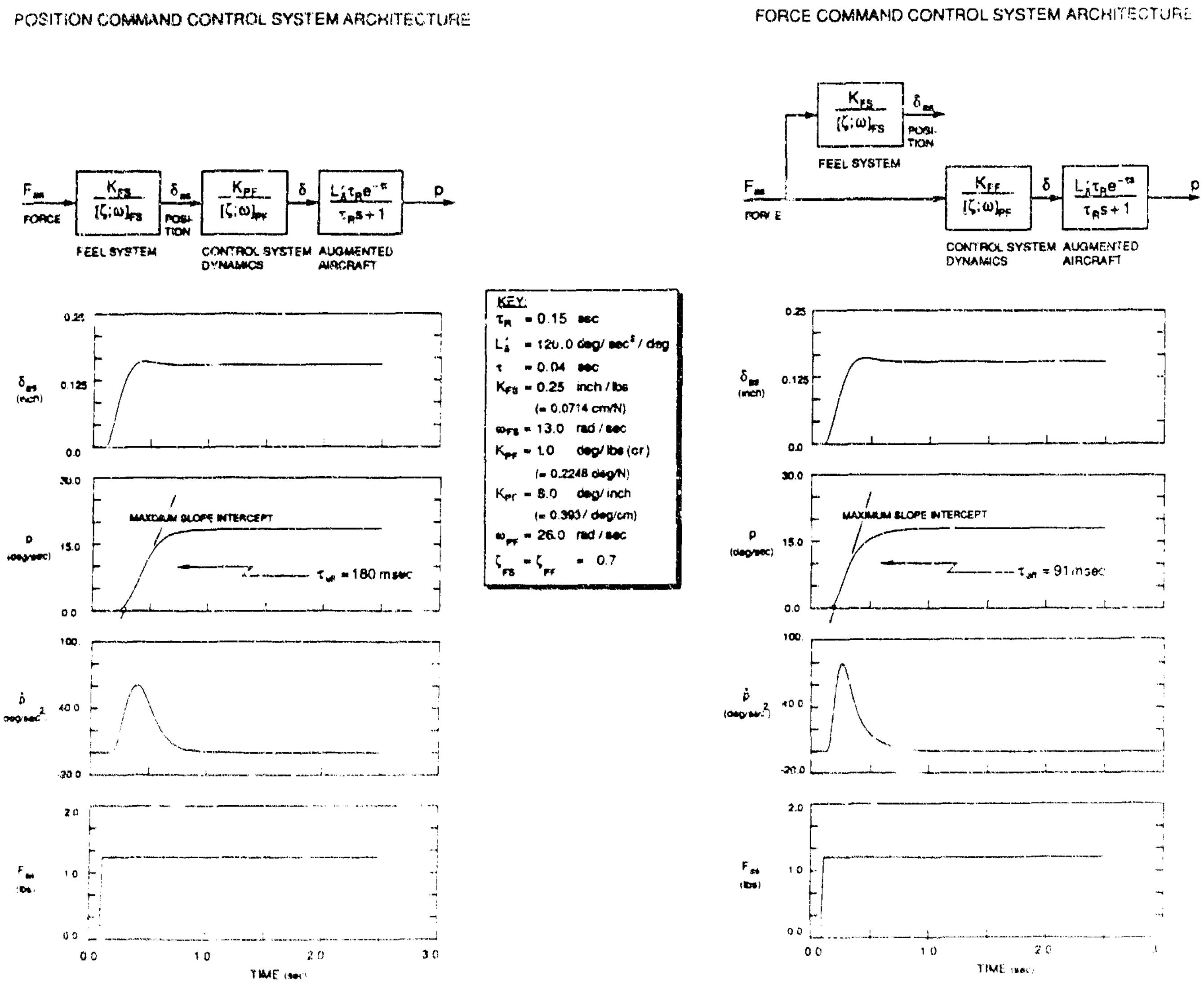


Figure 2 Position and Force Command Control System Comparison

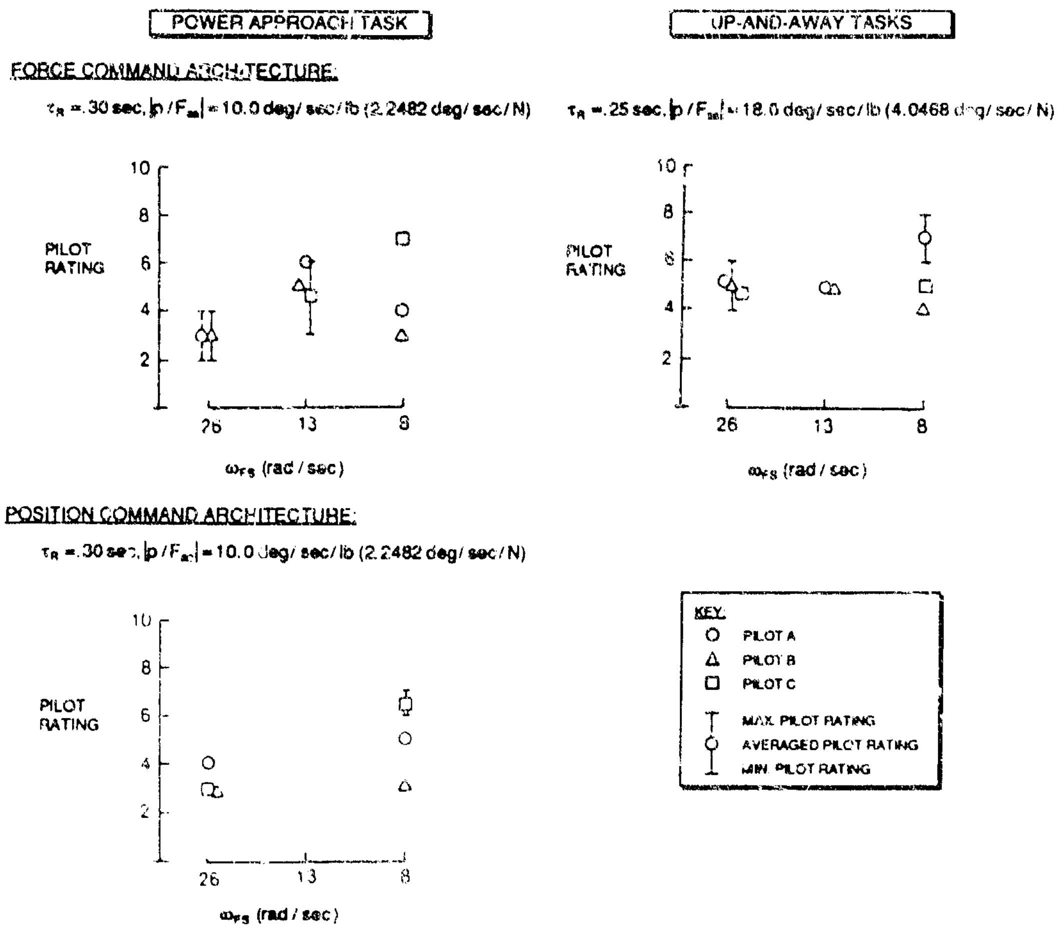


Figure 3 Pilot Rating Data for Feel System Frequency Variations

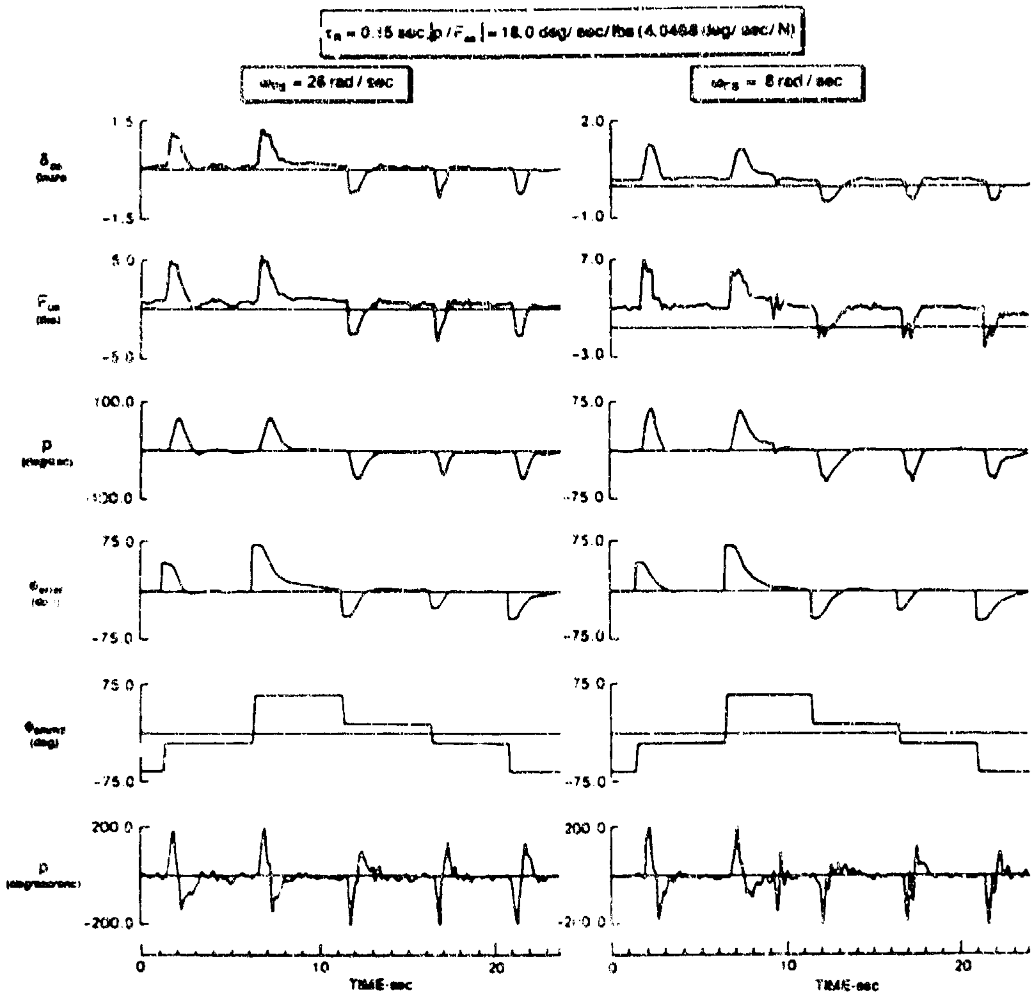
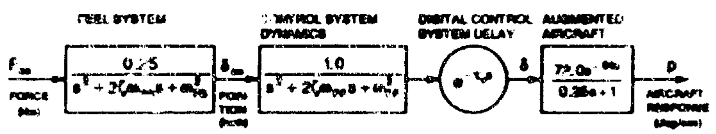


Figure 4 Time History of HUD-displayed Roll Tracking Task



FEEL SYSTEM FREQ. ω_{FS}	PREFILTER FREQ. ω_{pp}	DIGITAL DELAY $e^{-\tau_d s}$	EQUIVALENT SYSTEM* τ_R (sec)	EQUIVALENT SYSTEM* τ_e (msec)	EQUIVALENT SYSTEM** τ_R (sec)	EQUIVALENT SYSTEM** τ_e (msec)	PILOT RATINGS	PILOT COMMENTS
26.0 rad/sec	NONE	NONE	0.20	100	0.25	40	5,5,4	"BASELINE" CONFIG. SATISFACTORY BUT QUICK ABRUPT INITIAL RESPONSE. COMPENSATION REQ'D FOR PERFORMANCE
8.0 rad/sec	NONE	NONE	0.20	150	0.25	40	4,5,5,6	STRANGE FEEL IN STICK. MOVEMENT EFFECT IN HAND. QUICK INITIAL RESPONSE, BOBBLING, PIO TENDENCY WITH POOR PREDICTABILITY
26.0 rad/sec	NONE	110 msec	0.20	210	0.25	150	7,7	ABRUPT AND UNPREDICTABLE. PIO TENDENCY WITH DELAYED AND ABRUPT INITIAL RESPONSE. HIGH FREQ. PIO, "RATCHET"
26.0 rad/sec	13.0 rad/sec	NONE	0.76	210	0.25	105	7,5,5,7	MODERATELY QUICK INITIAL RESPONSE. SOME "BOBBLING" TENDENCY
13.0 rad/sec	26.0 rad/sec	NONE	0.25	210	0.20	100	7,10,4,7	SENSITIVE AIRPLANE WITH [illegible]. TENDENCY TO OVERSHOOT, SUSTAINED PIO

* Using p/F_{as} transfer function for equivalent system model ** Using p/δ_{as} transfer function for equivalent system model

Figure 5 Experimental Results of Feel System and Flight Control System Dynamics Variation

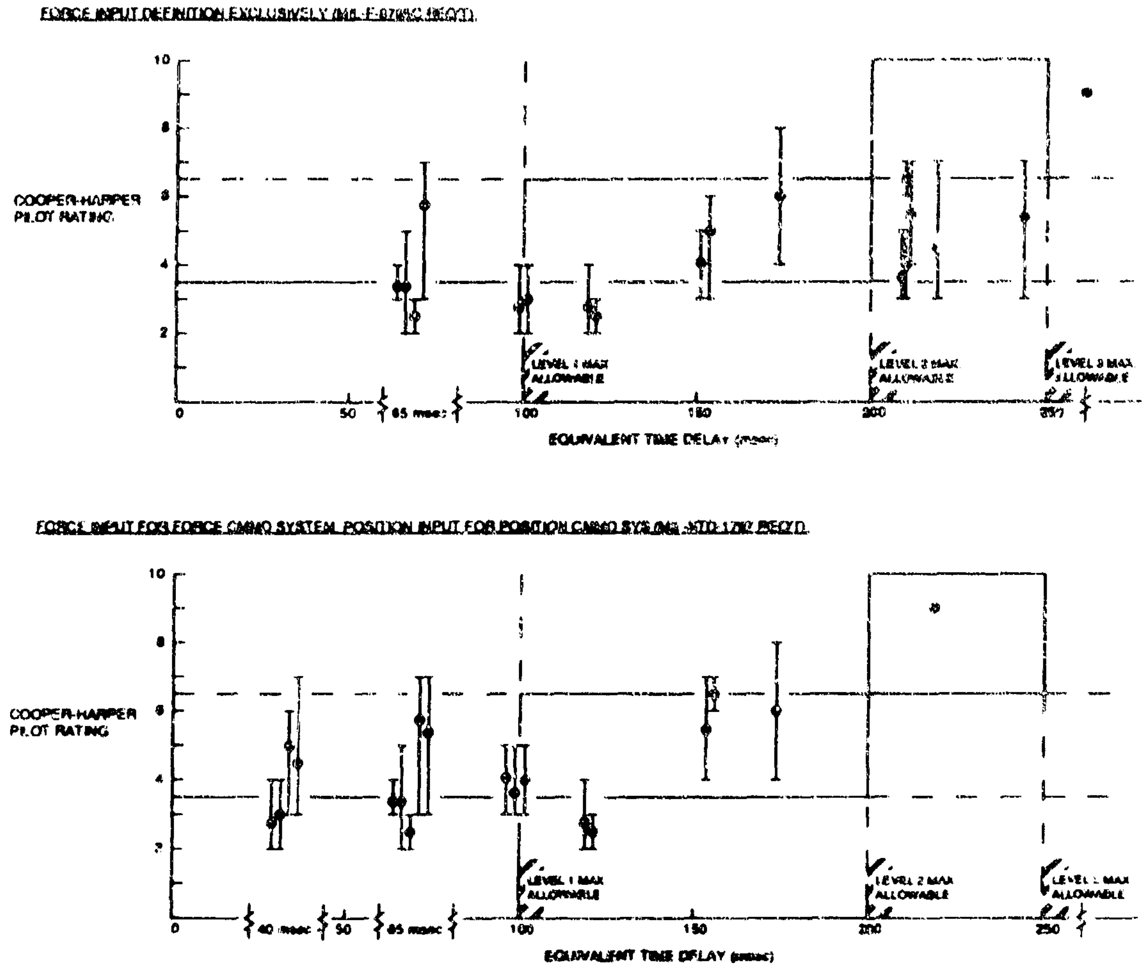


Figure 6 Pilot Rating Data Comparison Against MIL-F-8785C and MIL-STD-1797 Allowable Equivalent Time Delay Requirements

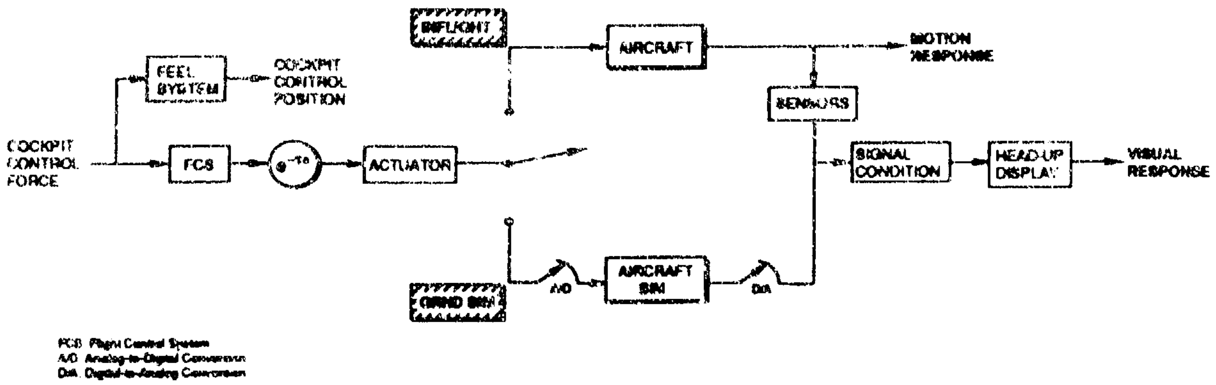


Figure 7 Simulation Time Delay Experimental Set-Up

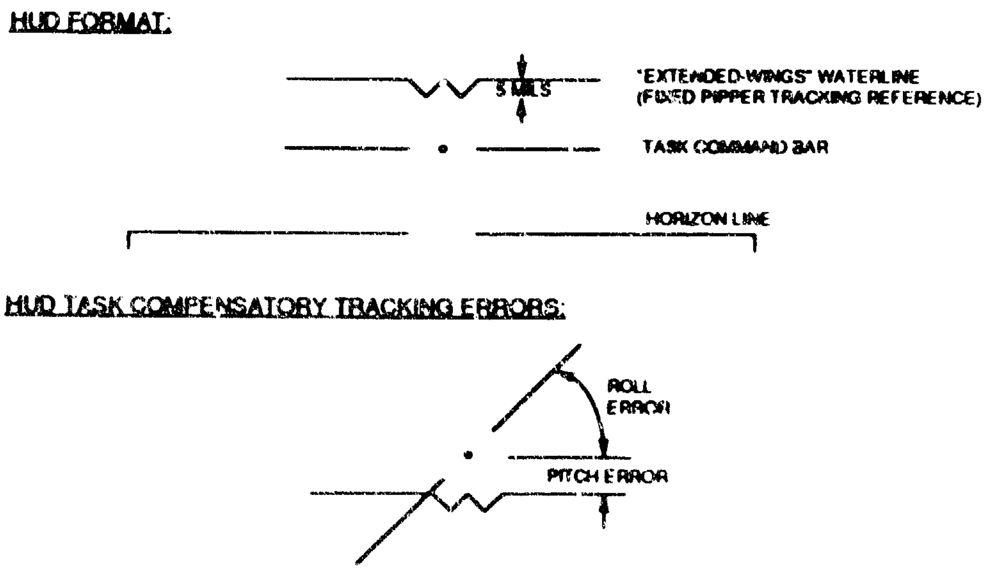


Figure 8 HUD-based Tracking Display Format

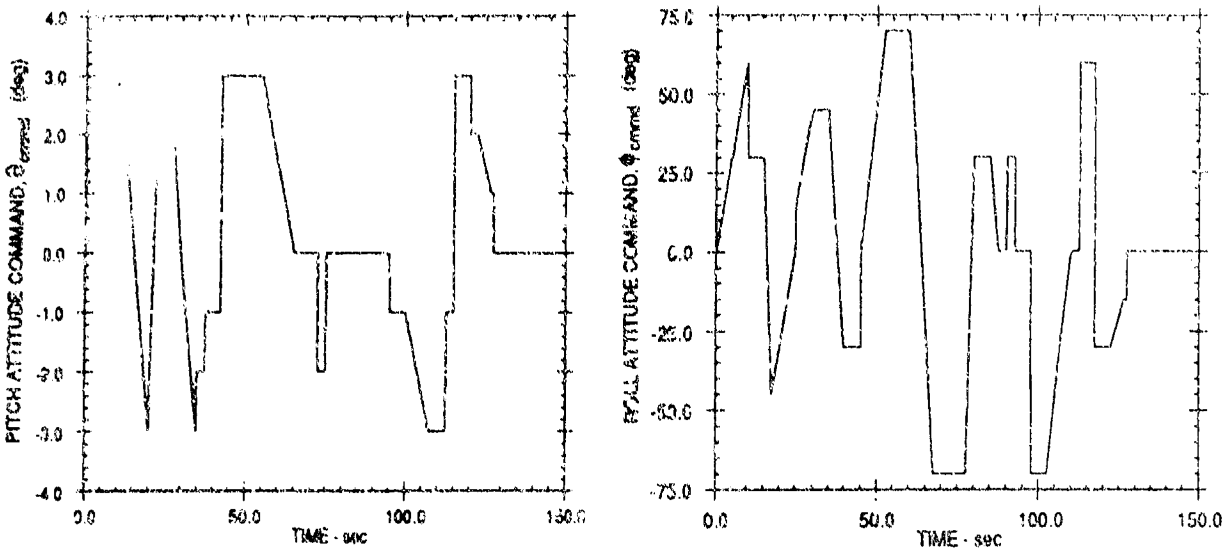


Figure 9 Step-and-Ramp HUD Tracking Task

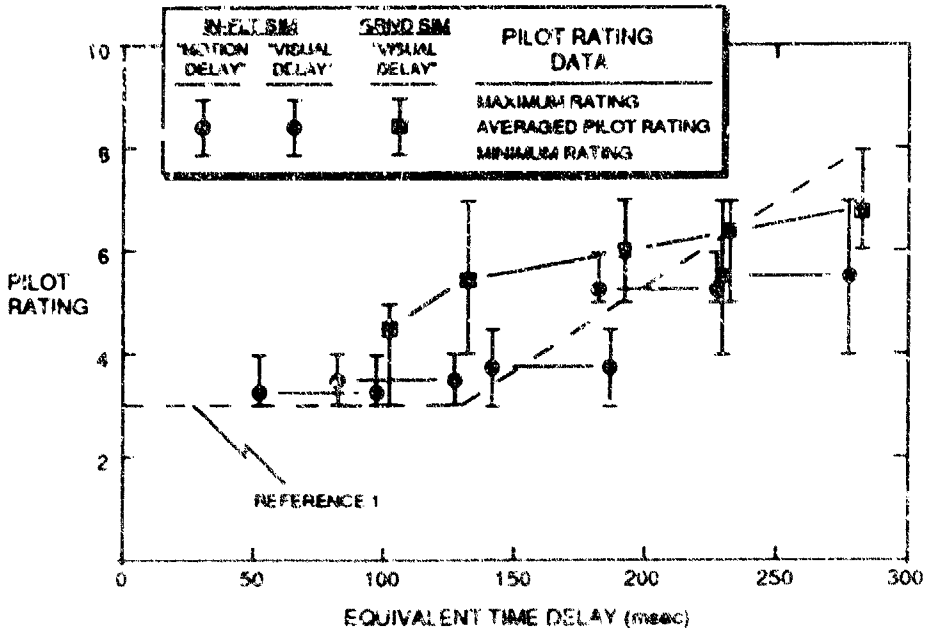


Figure 10 Comparison of Time Delay Effects on Flying Qualities During In-Flight and No-Motion Ground-Based Simulation

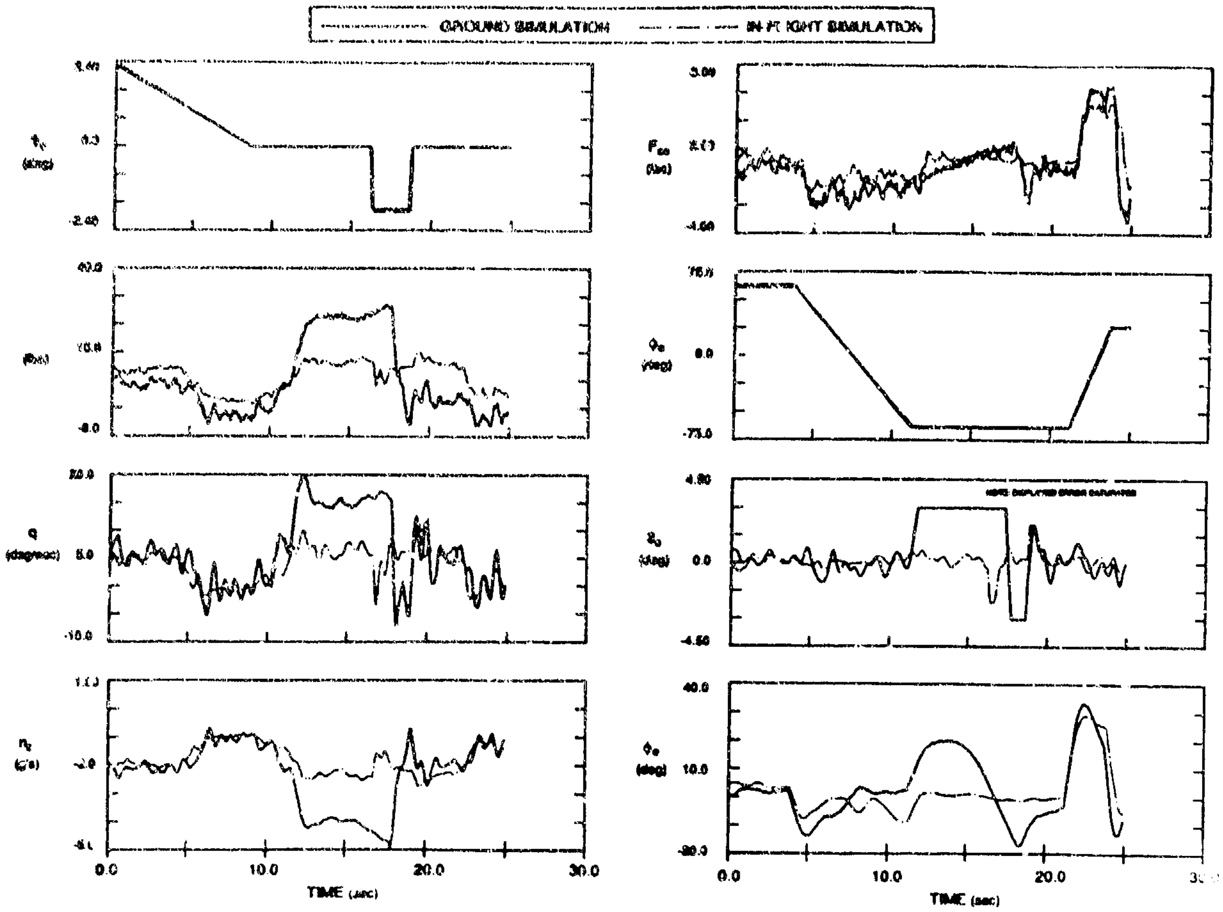


Figure 11 Effect of Motion Cuing on Fighter Aircraft, HUD Tracking Task

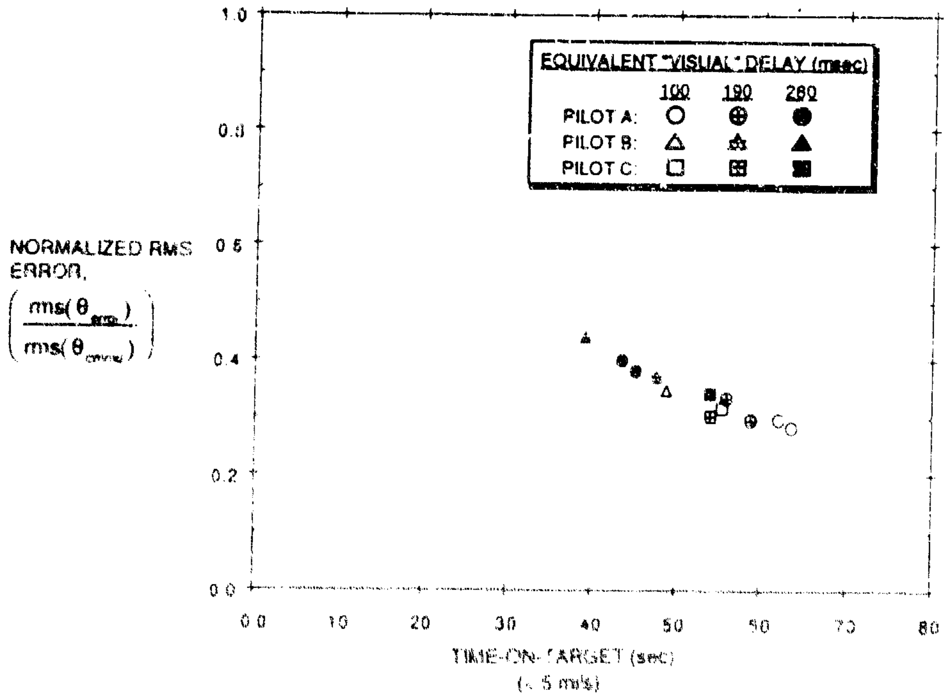


Figure 12 Pitch Time-on-Target and Normalized rms Error During in-Flight Simulation

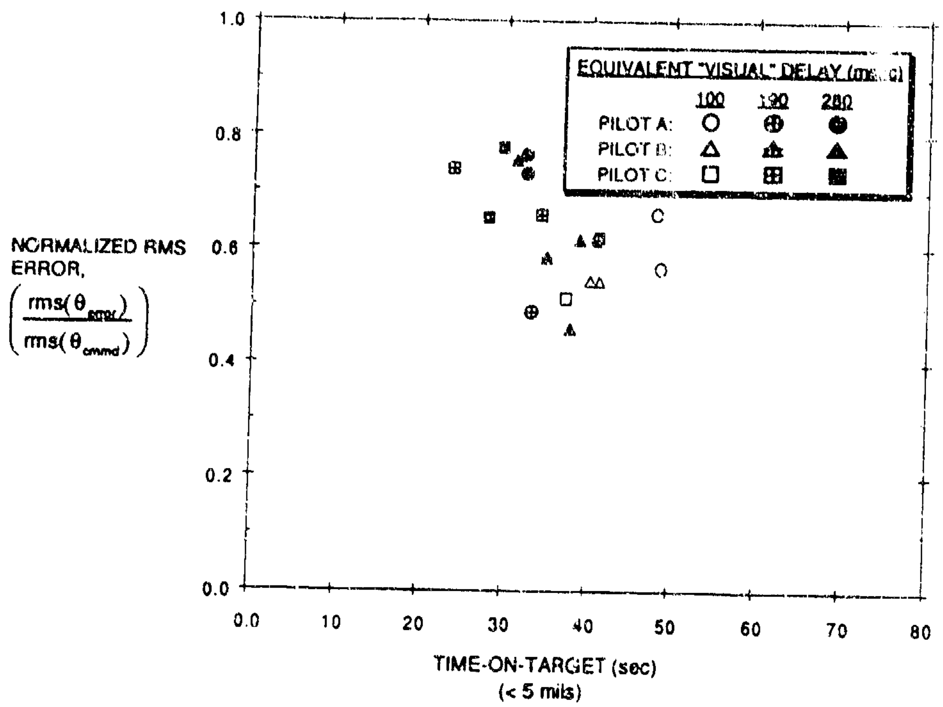


Figure 13 Pitch Time-on-Target and Normalized rms Error During Ground-Based Simulation

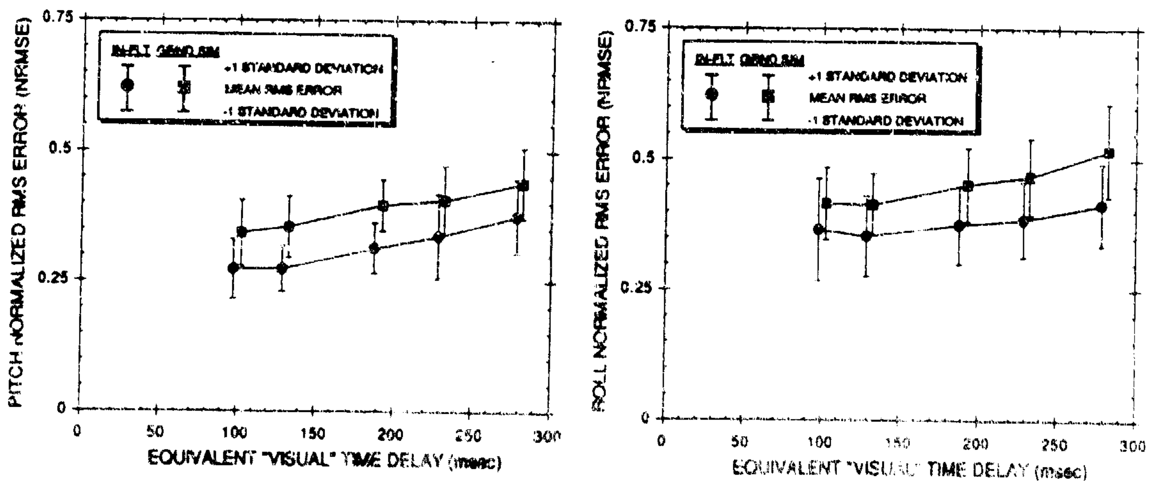


Figure 14 Sum-of-Sines Task Tracking Performance

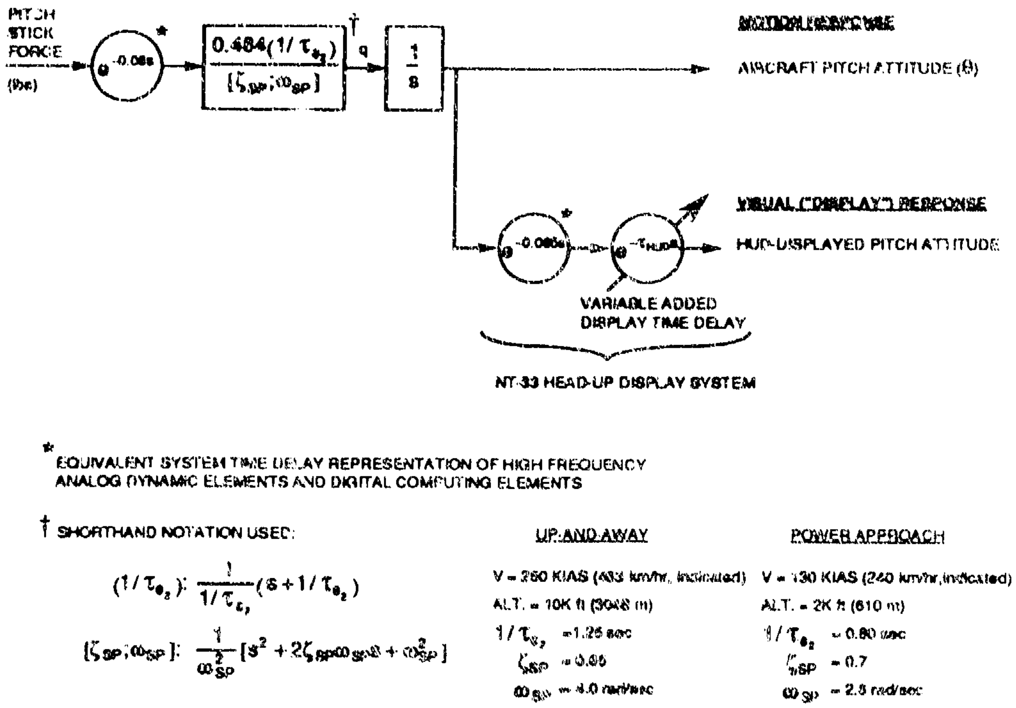


Figure 15 Schematic Diagram for Experimental Evaluation of Display System Delay Effects

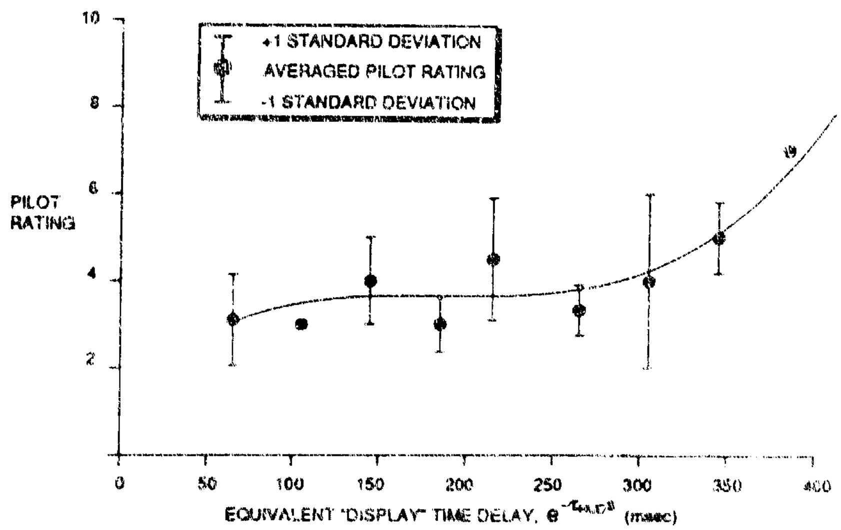


Figure 16 Effect of Display System Time Delay on Fighter Aircraft Flying Qualities During Up-and-Away Tasks

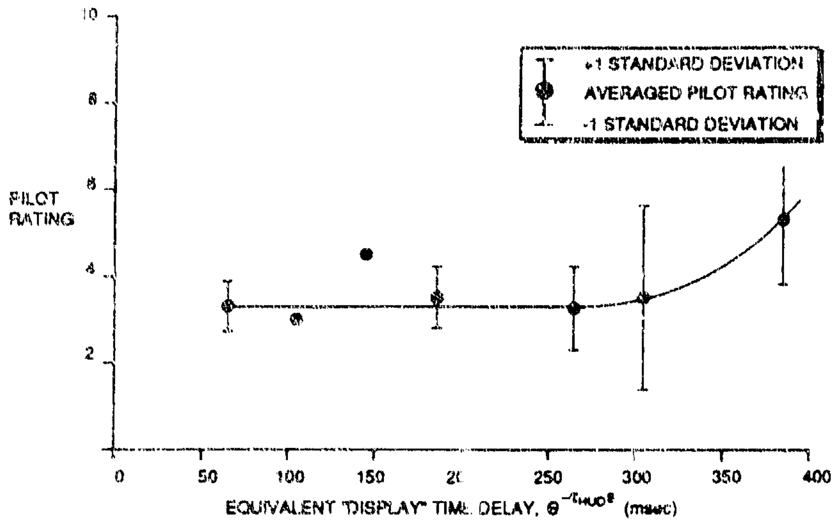


Figure 17 Effect of Display System Time Delay on Fighter Aircraft Flying Qualities During Power Approach Task

ESTIMATION OF PILOT RATINGS VIA PILOT MODELING

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SUMMARY

The dynamic behavior of pilots engaged in closed-loop, full-attention piloting can be estimated with some confidence using either the well-known "Crossover Model," a structural-isomorphic model, or an algorithmic (optimal control) model. Data and interpretations of the analyses made with these models can also be factors in the estimation of pilot ratings. In one technique a pseudo-pilot commentary in technical terms is developed employing results obtained from analysis with the classical models which is then converted to a pilot rating estimate for each axis of control. Ratings for multi-axis control are then determined with a special multiplicative rule, which has recently been validated with comprehensive multi-axis pilot dynamic measurements. While the algorithmic model results can be used in a similar way, it can also sometimes deliver pilot rating estimates directly. When suitable pilot-rating calibration data are available, the performance index which the OCM minimizes can be used to accomplish these estimates. Both the classical and algorithmic models are now suitable for application to deliver multiaxis rating pilot rating estimates.

INTRODUCTION

Flying qualities and piloting considerations have always been central issues in manned aircraft. The development of understanding and appreciation of flying qualities has relied on combinations of experiment and analysis in which the most important "metrics" have been subjective assessments and commenta . For undemanding task combinatio these measures can be almost delphically obscure and poor descriminators. Fortunately, many piloting tasks critical to flying qualities are closed-loop in that the pilot's behavior is both constrained by the vehicle and task-centered dynamics, and conditioned on the aircraft's state relative to a desired condition. For these tasks closer connections between dynamic behavior and subjective assessments are possible.

A major part of the analysis activities in the flying qualities art has been focused on control theory applications of pilot models which intrinsically summarize and characterize vast amounts of experimental data. These models serve as the mathematical attornies in what are, in many ways, typical applications of controls analyses akin to those conducted for automatic systems. But there are uniquely human attributes that must be embodied in the models. The most important by far is the enormously adaptive character of the human as a controller. Many modes of adaptive and plastic behavior are possible; most are well-recognized and understood, and can be represented with appropriate models (see e.g., Ref. 1). Another unique attribute is the self-assessment capability inherent in the overall system — the pilot's role as a vocal as well as an adaptive controller. In well-designed flying qualities experiments pilot ratings and commentaries become fundamental indicators of pilot and system dynamic behavior, system performance, and pilot workload. Accordingly, they are intrinsically associated with pilot models. The primary purpose of this paper is to summarize and indicate some of these connections.

Detailed descriptions of models for pilot dynamics, as such, are not the theme here. Current and comprehensive summaries of such models suitable for flying qualities applications appear in a recent AGARD Lecture Series (Item 14 on the "Surveys" list) and elsewhere (e.g., Refs. 2, 8). But models for dynamic analysis are necessary as the first step in an analysis procedure intended to provide ratings /comments estimates. Accordingly, the first section of the paper provides a brief review of appropriate pilot dynamic models.

The second section introduces pilot ratings in general and notes that there are basically two approaches which have been used to connect the pilot's assessments tc pilot and pilot-vehicle system dynamics. The first is to associate pilot and system dynamic and performance characteristics with ratings using functional relationships. Representative forms of pilot rating functionals are given in the next sections, illustrated by an extensive cross-section of examples. The second approach is "clinical," and its treatment is deferred to the last section.

One means of combining single axis ratings into a multi-axis estimate is developed in the next section The development relies on workload and attentional demand concepts, although the ultimate combination rules have a validity based on experiment which transcends any such basis.

The final section covers a clinical approach to rating estimates which is more general and does not rely directly on empirical correlations. Instead, it takes into account the pilot and pilot-vehicle system characteristics as revealed by analysis, and considers their implications for control.

PILOT DYNAMIC MODELS

The human pilot is the archetype hierarchical, adaptive, optimalizing, decision-making controller. In accomplishing these functions the pilot exhibits a bewildering variety of behavior which defies quantitative description when considered in the large. Nonetheless, since World War II scientists and engineers have attempted to describe specific elements within this functional list in terms of quantitative models. The major source of paradigms for quantitative descriptions derive from control theory. Control theories can also be classified using similar adjectives, so it is not surprising that almost every new advance in control theory has led to attempts to better understand aspects of human behavior in the perspective of this advance. When these attempts have been fruitful, a control theory paradigm has emerged which is useful in quantifying the human's operations. In the process theory has been used to "explain" experiment, and unexplained experimental results have motivated new theory. The results of this widespread synergistic activity have been documented in hundreds of research papers and in a series of summary surveys which have appeared periodically. (A chronological listing of surveys is given at the end of this paper, following the reference list). As a consequence, much of the successful art is now mature. Furthermore, it has become a fundamental mode of thinking on the part of technical practitioners in the fields of pilot-vehicle systems, flying qualities, operator/vehicle control system integration and many aspects of interactive man-machine systems. From this rich variety of man-machine control models that have been addressed, the emphasis here will be confined to models particularly pertinent to flying qualities situations. The models of interest are quite comprehensive (as encapsulations of experimental data) as well as broadly representative of useful theory (in that both classical and modern control viewpoints are presented).

Flying qualities in general can be divided into "unattended" (and trim), large amplitude maneuvering, and "closed-loop" operations. All three categories have some degree of pilot interaction, and pertinent models exist for all types (Ref. 1). The pilot-vehicle systems most relevant for the exposure of critical flying qualities involve operations in which the pilot controls the effective aircraft dynamics in a closed-loop fashion. "Closed-loop" in this sense means operations wherein at least part of the pilot's control actions are conditioned by the differences between the aircraft's desired and actual outputs. The kinds of piloting covered include precision control, regulation, and stabilization tasks; the types of flying qualities tests represented include "flying qualities while tracking". For these cases the core pilot-vehicle system, and the associated human pilot behavior, are referred to as "compensatory". Fortunately, compensatory operations are the most definitive in disclosing critical flying qualities deficiencies, and the associated pilot models are the most extensive and advanced.

There are currently three predominant types of human operator models used to describe compensatory behavior. Reference 2 is an up-to-date summary which includes both full-and divided-attention operations. By far the simplest model describes the human pilot-vehicle system dynamics in the crossover frequency region (Fig. 1a). (The crossover frequency occurs where the open-loop amplitude ratio of the pilot-vehicle system is unity.) It is often sufficient for flying qualities analyses intended to elicit the governing vehicle parameters, key variations, and basic relationships. Because of its overall importance and simplicity, the implications of the Crossover Model and some simple rubrics for pilot-vehicle analysis using the model are summarized in the Appendix.

The most elaborate description of human dynamic properties as a controller is the structural-isomorphic model. This is an expansion of the crossover model which attempts to account for many of the subsystem aspects of the human controller as well as the total input-output behavior. A somewhat simplified version of the pilot's transfer characteristic is shown in Fig. 1b. The third type of pilot model — the algorithmic or optimal control model (Ref. 2-8) — stems from a quite different control theory perspective (Fig. 1c). The primary purpose of this model is to mimic the human operator's total response by appropriate specialization of modern control computational procedures. Because the "Crossover Model" is the most broadly applicable and best understood of human dynamic descriptions, the behavior predicted by either the structural-isomorphic or the algorithmic models must "reduce" to this form in the crossover frequency region. Thus the more elaborate models must inevitably return to the crossover model as a necessary limiting case "consequence".

In order to exercise the algorithmic model, a new formulation of the computational steps involved in the optimal control model has been developed in the context of a commercially available control system analysis program (Program CC). Besides the PC compatible format, this new formulation includes additional sequences which allow the analyst to determine the actual estimated pilot characteristics from the optimal controller solution. These steps should improve the understanding and interpretation of algorithmic model-based estimates, and should broaden the use of the OCM by making it available as a PC compatible routine. The new formulation is documented in Reference 8.

In modern high performance aircraft the pilot is no longer primarily a controller. Instead communications, monitoring and management of automated equipment, planning, re-adjusting to adapt to changing circumstances, etc., place increasingly arduous demands on the pilot. Thus, flying stressful, high workload mission phases may require the pilot to divide his attention between control and managerial tasks. The dynamic models for the pilot must take these divided attention operations into account. This is done for both classical and algorithmic models in Ref. 2.

The pilot models briefly reviewed above focus on the dynamics of pilots

a) CROSSOVER MODEL

$$Y_p Y_c \doteq \frac{\omega_c e^{-j\omega\tau_1}}{j\omega}, \quad \omega \sim \omega_c; \quad Y_p \doteq \frac{\omega_c e^{-j\omega\tau_1}}{j\omega Y_c}$$

b) STRUCTURAL-ISOMORPHIC MODEL

$$Y_p \doteq \frac{K_p (T_L j\omega + 1) e^{-j\omega\tau_2}}{(T_I j\omega + 1)(T_N j\omega + 1)\left[\left(\frac{j\omega}{\omega_{NM}}\right)^2 + 2\zeta_{NM}\left(\frac{j\omega}{\omega_{NM}}\right) + 1\right]}$$

c) ALGORITHMIC (LINEAR OPTIMAL CONTROL) MODEL

$$J(u) = E\left\{\lim_{T\to\infty}\frac{1}{T}\int_0^T (e^2 + G\dot{u}^2)dt\right\}$$

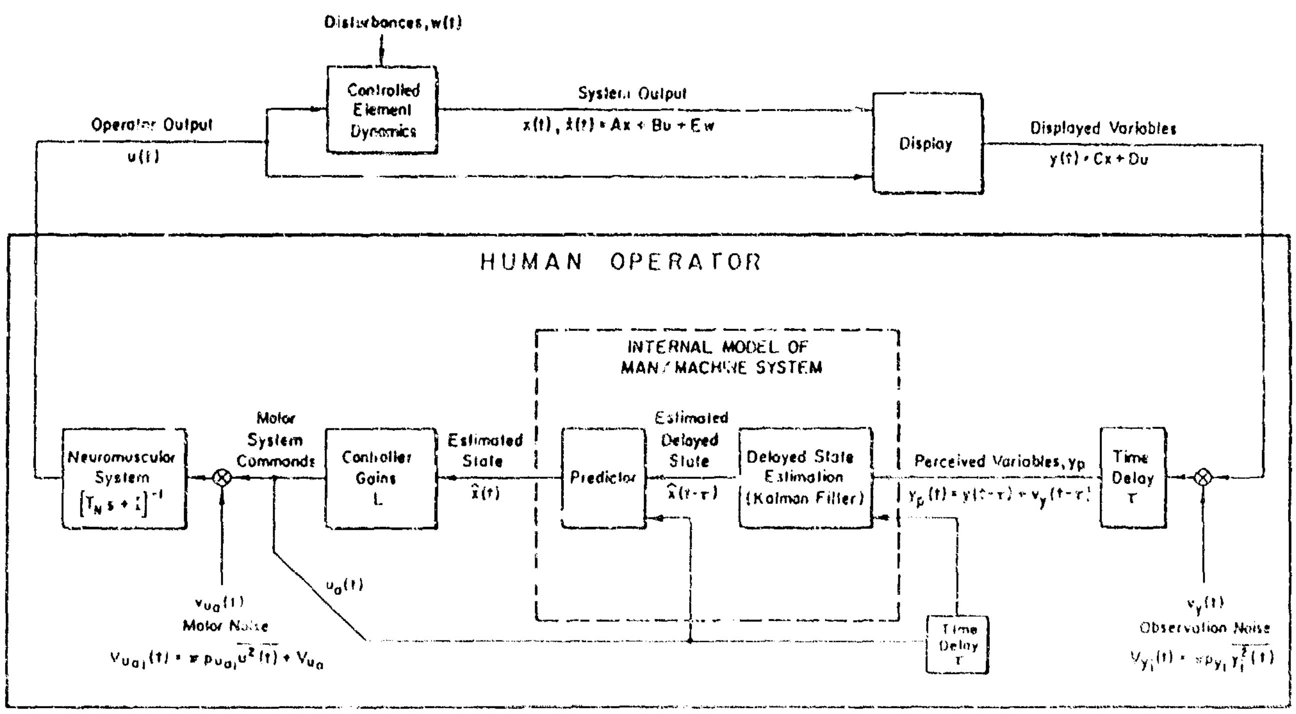


Figure 1. Pilot Models

in flying qualities tasks for both full and divided attention conditions. This paper concentrates on the workload associated with generating these dynamics and accomplishing the control task as measured by subjective impressions delivered as pilot ratings. Ideally we would like to predict the workload (pilot commentaries and ratings) along with the prediction of the underlying pilot dynamics. This paper summarizes the extent to which this can be done.

PILOT RATINGS

The previous section briefly listed techniques for the estimation of pilot dynamics in closed-loop tasks. Because aggressively performed closed-loop tasks are ordinarily critical from the standpoint of pilot compensation or skill required and are usually high workload flight phases, they tend to be dominant discriminators in flying qualities assessments. The assessments themselves are provided by pilot comments and associated ratings, such as the Cooper-Harper scale (Ref. 9), part of which is shown in Fig. 2. It is apparent from the "Demands on the Pilot..." column that pilot compensation (equalization) and effort (workload) and task performance are major constituents of the rating scheme. When the task variables (effective vehicle dynamics, forcing functions and disturbances, etc.) are well-defined, the pilot-vehicle system dynamic models described in Refs. 2 and 8 can be used to make quantitative estimates of pilot compensation and task performance. Workload, on the other hand, is much more difficult to quantify. Still, the subjective pilot ratings and comments, which are subjective workload indices, should have some connections with the pilot and pilot-vehicle system dynamics and performance. These connections are intrinsically empirical. They are also awkward theoretically because the rating scale is ordinal. Consequently, averages, standard deviations, etc., are not legitimate statistics, although this has never stopped flying qualities engineers from using them! (Fortunately, the scales seem to be close to interval in some ranges or, for the purist, data can be converted to an underlying interval scale wherein all the parametric statistics can be applied and then converted back — see Ref. 10.)

Aircraft Characteristics	Demands on the Pilot in Selected Task or Required Operation*	Pilot Rating
Excellent Highly Desirable	Pilot Compensation Not a Factor for Desired Performance	1
Good Negligible Deficiencies	Pilot Compensation Not a Factor for Desired Performance	2
Fair – Some Mildly Unpleasant Deficiencies	Minimal Pilot Compensation Required for Desired Performance	3
Minor But Annoying Deficiencies	Desired Performance Requires Moderate Pilot Compensation	4
Moderately Objectionable Deficiencies	Adequate Performance Requires Considerable Pilot Compensation	5
Very Objectionable But Tolerable Deficiencies	Adequate Performance Requires Extensive Pilot Compensation	6
Major Deficiencies	Adequate Performance Not Attainable With Maximum Tolerable Pilot Compensation Controllability Not in Question	7
Major Deficiencies	Considerable Pilot Compensation Is Required for Control	8
Major Deficiencies	Intense Pilot Compensation Is Required to Retain Control	9
Major Deficiencies	Control Will Be Lost During Some Portion of Required Operation	10

*Definition of required operation involves designation of flight phase and/or subphase with accompanying conditions.

Figure 2. Cooper-Harper Handling Qualities Rating Scale

The goal of this paper is to summarize the available connections between pilot and pilot-vehicle system dynamics and pilot ratings. There are fundamentally two approaches which have been used with some success. The first directly associates pilot and system dynamic and performance characteristics with the pilot rating via a functional relationship. Such functionals have been developed for use with both the classical and OCM versions of pilot models. The actual connections which have been established are based on specific tasks and circumstances.

The second approach is more clinical in style. It takes into account the pilot and pilot-vehicle system characteristics in terms of their implications for control. A list of assessment features is considered in order to reveal symptoms of flying qualities problems. Some quantitative aspects can be set forth, but others are only qualitative. Consequently, this approach is more the basis for a pseudo-pilot commentary rather than a means to make numerical rating esti-mates directly. Of course, if the "commentary" is sufficiently complete it can be converted to a rating by working through Fig. 2. The clinical technique is especially useful to define possible flying qualities prob-lems and key effective airplane dynamic parameters, or as a means of interpreting experimental data.

The two approaches described are currently most highly developed for single axis situations. Multi-axis rating estimates can be developed from single axis results using a "product rule," and OCM-based multi-axis results can be the basis for direct estimates of multi-axis ratings. Both approaches are described below.

PILOT RATING FUNCTIONALS

A direct approach is to formulate a functional which incorporates the pilot and system dynamic and performance quantities which are presumed to underlie the pilot rating. A general form which explicitly contains some and implicitly contains all of the desired features is given by,

$$R = \mathcal{R}\Bigg[\ \underbrace{\overbrace{\lambda_i \overline{q_i^2}}^{\substack{\text{Dominant Aircraft}\\ \text{Motion Quantities}\\ \text{and Task Measures}}}}_{}^{\substack{\text{MISSION/TASK}\\ \text{PERFORMANCE}}}\ ,\ \overbrace{\overline{\delta_j^2}\ ,\ \overline{\dot{\delta}_j^2}}^{\substack{\text{Pilot Activity}\\ \text{(Scale of}\\ \text{Pilot Effort)}}}\ ,\ \overbrace{\left.\frac{d\,|Y_p|_{dB}}{d(\log\omega)}\right|_{\omega_{I_k}},\ (T_L, T_I)_k}^{\substack{\text{PILOT WORKLOAD}\\[4pt] \text{Pilot Equalization}\\ \text{(Dynamic Quality}\\ \text{of Pilot Effort)}}}\ \Bigg]_{i,j,k} \qquad (1)$$

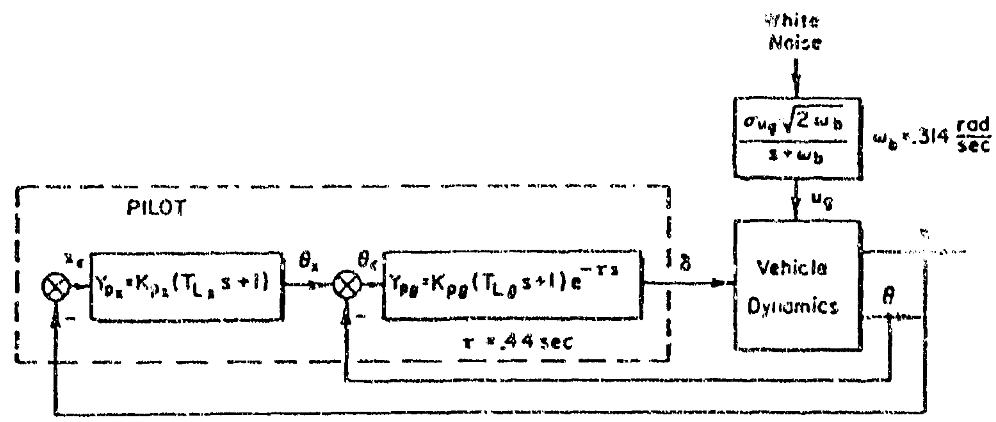
the subscript notation used is intended to imply that $i \epsilon I$ motion and task measures are controlled by $k \epsilon K$ pilot loops actuating $j \epsilon J$ control points. This functional form is general enough to include the existing (e.g., Refs. 1, 2, 10-23) approaches to quantitative flying qualities rating functions. The key closed-loop system quantities in the rating functional are measures of mission/task performance. These are conveniently described by a set of dominant weighted aircraft motion deviations and total task accuracy or error indications (represented by the q_i).

The pilot activity component of pilot effort, $\overline{\delta_j^2}$ (either force or displacement, as pertinent to the manipulator involved) and $\overline{\dot{\delta}_j^2}$ are particularly dependent on the level of pilot gain. For a given gain, these will increase directly with gust disturbance spectrum amplitude and pilot-induced noise (remnant) amplitude. Accordingly, both the mission/task and pilot activity quantities will reflect turbulence and remnant levels.

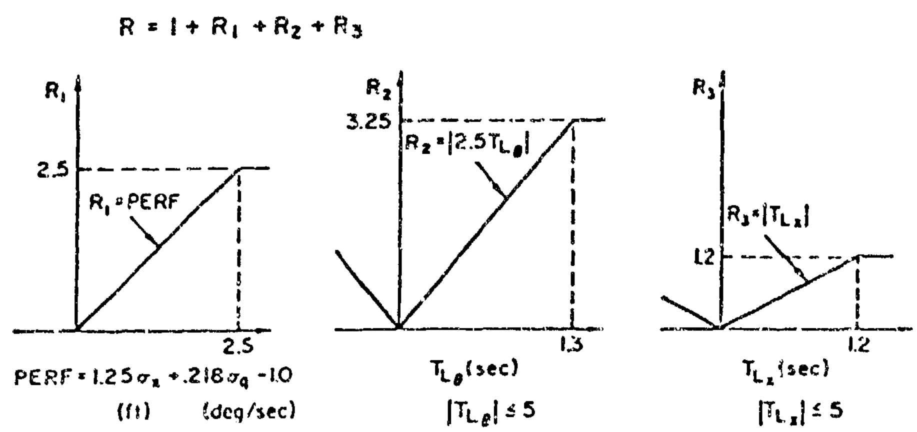
The pilot equalization component of pilot workload is generally represented in Eq. 1 by the slope (in dB per octave or decade) of the pilot's amplitude ratio evaluated at a particular frequency (generally near crossover). This is by no means the only measure available to describe the dynamic quality of the pilot's effort; others (e.g., Refs. 12-16) use pilot lead time constants (the T_{L_k} shown in Eq. 1) which are a desirable alternative for particular situations with a sufficient data base. Then the rating functional takes the very useful form illustrated in Fig. 3. At present, adequate functions of this form exist for precision hover tasks (Refs. 12 13), pitch attitude control (Ref. 14), and roll attitude control (based on Refs. 17 and 18). In addition, the Ref. 19 data provide a base for a multiloop functional

The technique pioneered in Ref. 12 actually used the pilot rating functional as a performance index, as well as a rating estimator. That is, the pilot model parameters (Fig. 3a) were adjusted to minimize R, the pilot rating functional.

The follow-on work of Ref. 16, which was dedicated to experimentally verifying the Ref. 12 result, produced a "modified" pilot rating functional for the $R_2 + R_3$ component, as shown in Fig. 4. The correlation between predicted and actual ratings, shown in Fig. 5 are reasonably good.



a) Pilot / Vehicle Model



b) Pilot Rating Functional

$J = 1 + PERF + R_2 + R_3$

PERF, R_2, R_3 – as defined in b)

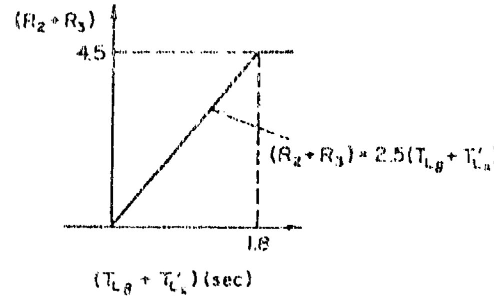
1) K_{p_x}, T_{L_x}, K_{p_θ}, T_{L_θ} found to minimize J

2) K_{p_x}, K_{p_θ} simultaneously increased by 20%. If system asymptotically stable, performance and rating computed based on values found in 1). If system not stable, go to 3).

3) T_{L_θ}, T_{L_x} held fixed at values found in 1). K_{p_x}, K_{p_θ} simultaneously decreased until values found for which a 20.3% increase would result in an unstable system while a 20% increase would result in a stable system. These values are used to compute performance and rating.

c) Minimum Pilot Rating Concept For Pilot Model Parameter Adjustment

Figure 3. The Elements of the "Paper Pilot" for the Hover (Ref. 12)



Modification for Hover Paper Pilot Rating Functionals (Ref 16)

Figure 4. "Paper Pilot" Rating Functionals

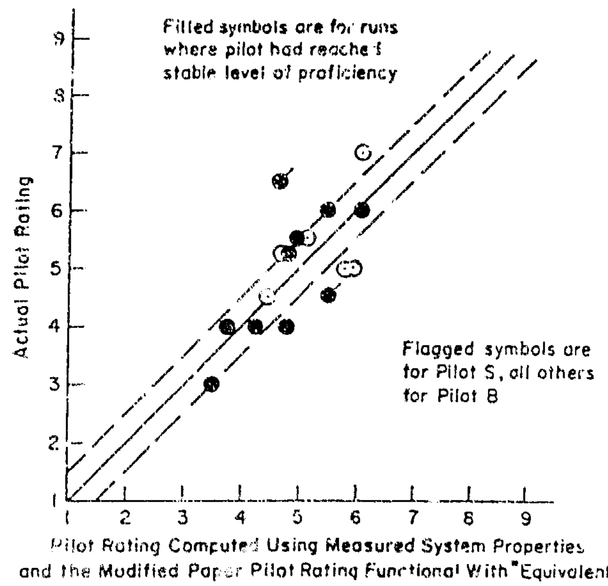


Figure 5. Comparison of Actual Ratings with Ratings Computed Using Modified Hover Paper Pilot Rating Functional (Ref. 16)

In Ref. 14 the task was changed to pitch attitude control and the resulting pilot rating functional evolved to:

$$R = \underbrace{\frac{0.1}{.974 - \sigma}}_{R_1} + \underbrace{2.5T_L + 1.0}_{R_2}$$

where

$\sigma = \sigma_e/\sigma_1$ = ratio of rms error to rms input

T_L = pilot lead, seconds

and

$$R \leq 10 \quad , \quad 0 \leq R_2 \leq 3.25 \quad , \quad 0 \leq R_1$$

(If a value of $\sigma < .974$ cannot be obtained, $R = 10$)

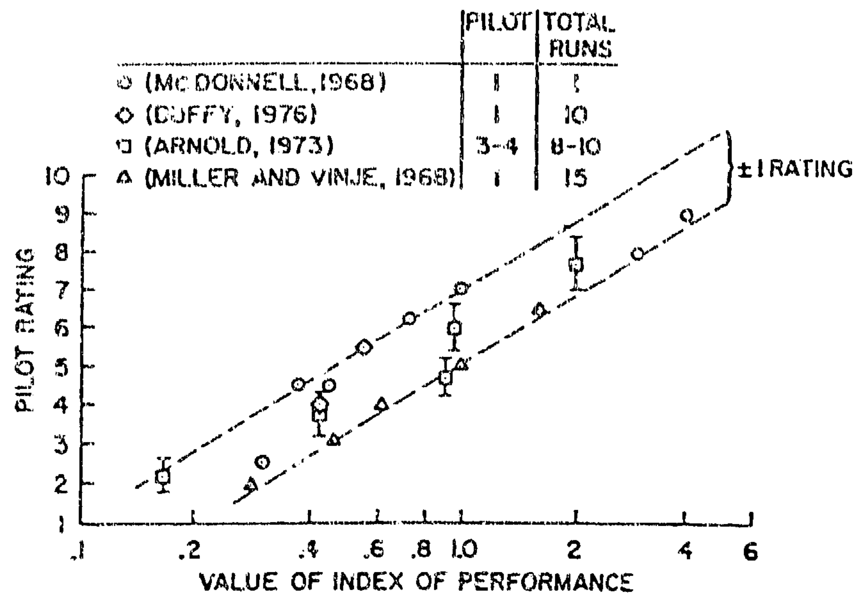
Yet another way to estimate pilot ratings is to use correlations developed for the algorithmic pilot model (Refs. 20-22, 24, 25). This pilot rating estimation procedure is based on the hypothesis that the pilot rating for a particular task and set of vehicle dynamics can be correlated with the numerical value of the index of performance (minimum values of the OCM Cost Function) resulting from the optimal pilot modeling procedure. As indicated in Fig. 6, this has worked fairly well for some single-axis cases (e.g., helicopter hover and longitudinal approach).

The extension of the OCM performance-index-based pilot rating estimating procedure to the multi-axis case has recently been addressed. The basic developments are given in Refs. 8 and 22. Reference 26 is the primary source for these studies of connected single- and multi-axis rating data, although no associated pilot dynamic information is available. Consequently, the OCM was used to establish pilot and system dynamics estimates. For the rating estimates only the performance index is needed. The appropriate performance index for each single axis was chosen to be,

$$J_{axis_i} = \left[\frac{\sigma^2_{\epsilon_i}}{\sigma^2_{c_i}} + \frac{\sigma^2_{\dot{\delta}_i}}{g_i}\right] \qquad (2)$$

And for a multi-axis task, the objective function used was

$$J_{task} = \sum_i^{N_{axes}} J_{axis\ i} \qquad (3)$$



Model Parameters for Hover Task of Miller and Vinje (1968)

Parameter	*Value*
Time delay τ	0.2 s
Neuromuscular time constant T_N	0.2 s
Visual thresholds	None
"Full-attention" noise-signal ratio for observation noise	0.0025
Fraction of attention on control task f_c	0.25 → 1.0 (configuration dependent)
Noise-signal ratio for motor noise	0.003
Index of Performance	

$$J = E\left\{ \lim_{T\to\infty} \frac{1}{T} \int_0^T \left[y^T(t) Q y(t) + u^T(t) R u(t) \right] dt \right\}$$

$y_1 = \dot{\theta}$	$q_{11} = (1/0.0873)^2\ s^2/rad^2$
$y_2 = x$	$q_{22} = (1/3.5)^2/ft^2$
$y_3 = \dot{x}$	$q_{33} = (1/3.5)^2\ s^2/ft^2$
u_1 = commanded control	$r_{11} = (1/0.138)^2/ft^2$

Figure 6. Pilot Rating vs. Value of Model Index of Performance (Ref. 20)

The justification of this selection involves three considerations. The first relates to the selection of equal (unity) weighting on each J_{axis_i} in the definition of J_{task} in multi-axis tasks. This decision was based on the instructions given to the subjects in the Ref. 26 experiment. They were to attempt to minimize the errors in all controlled axes. That is, they were instructed that no axis was to be given preference, which would then define primary and secondary sub-tasks.

Secondly, the normalization of the mean-square error with the mean-square command deals nicely with the fact that different units and different command-signal strengths were used in several axes.

Finally, the interpretation of g_i requires some discussion. In the OCM, the selection of g_i defines the frequency range over which the open-loop system amplitude ratio approximates a K/s-like form. In connection with the OCM it is often cited that g_i is selected to yield a desired neuromotor time constant, T_N, in the pilot's describing function obtained from the model. But, as indicated in Refs. 2 and 8, when the total pilot describing function, Y_n, is actually constructed from its various elements in the OCM, the T_n established in this fashion is canceled by a directly compensating lead, leaving the actual estimated Y_p with no $(T_n s + 1)^{-1}$ lag. Still, it has been convenient to adjust g_i in this fashion even though the lag will later disappear. In this vein, the value of the desired neuromotor time constant used is either 0.1 sec, or the T_n that yields the lowest error (e.g., best performance), whichever is greater. Notice that after T_n is determined in the above fashion, this "operatin point" is associated with some we g_i in $J_{axis\ i}$. This value may alsc fer the subject's subjective tra between performance (σ_ϵ) and workload (σ_δ^2). And since pilot lead and σ_δ^2 are correlated, this procedure maximizes the possibility of relating the resulting value of J_{task} to the subjective rating of the task.

Shown in Fig. 7 is the correlation between J_{task}, as modeled, and the subjective ratings of the task. The correlation between J_{task} and POR from the single-axis results appears to hold for the multi-axis results as well. This result seems to indicate that the ratings reflect the

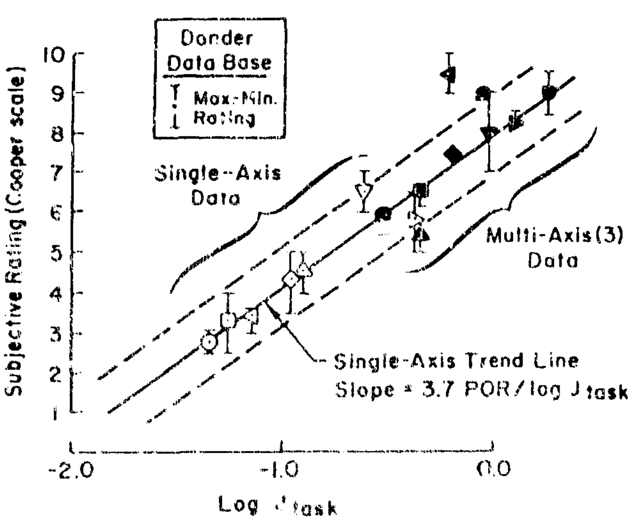


Figure 7. Pilot Rating vs. Performance Index for Dander Single- and Multi-Axis Tasks (see Ref. 22)

actual performance and workload (stick) rate) in the overall task. The results also tend to support the hypothesis that determining the weightings g_i in the manner discussed leads to approximately "correct" relative weightings on control rate in the axis, and the relative weight between control rate and normalized error. Because the multi-axis correlations follow the same trend as the single-axis data, this study indicates that the objective function for multi-axis situations can be extrapolated (or calibrated) from single axis correlations.

WORKLOAD, ATTENTIONAL DEMANDS, AND THE PRODUCT RULE FOR MULTI-AXIS RATINGS

There is a strong connotation of increasing pilot effort and workload in the phrases of the Cooper-Harper Scale (Ref. 9) which invoke levels of "pilot compensation," but workload is difficult to define and, consequently, to quantify. A general definition that can be measured and predicted is workload margin, defined as the ability (or capacity) to accomplish additional (expected or unexpected) tasks. The pilot opinion rating scale satisfies this definition up to its "uncontrollable" limit point. It is, therefore, a key workload measure, easy to obtain in some experimental circumstances.

Auxiliary tasks have been developed that satisfy the workload margin definition given above and that permit more objective measurements. One such task provides a complementary pair of measures suitable for integrating many workload concepts and factors into one basic context. These are the "attentional demand" and the "excess control capacity."

The attentional demand and excess control capacity measures have been connected with pilot rating in a multi-multi-axis experiment using the so-called cross-coupled subcritical task (see, for example, Refs. 10 and 27). A block diagram of the general experimental setup is shown in Fig. 8. The pilot first performs the primary task alone, attempts to achieve

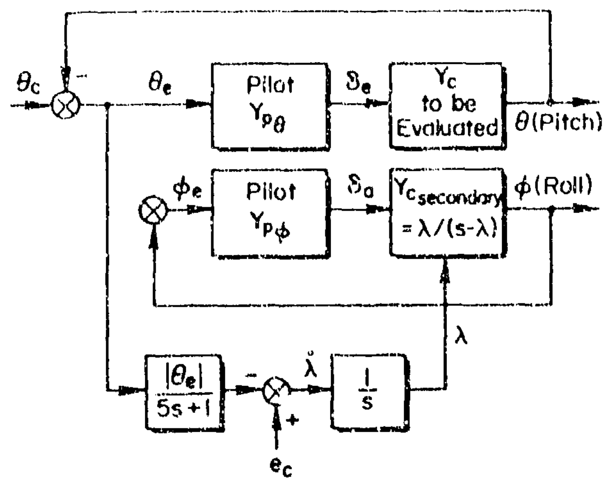


Figure 8. Single-Loop Primary Task with Secondary Cross-Coupled Loading Task

satisfactory levels of performance, and provides a Cooper-Harper pilot rating.

The secondary subcritical tracking task is then connected in order to "load" the pilot. The difficulty of the secondary task is made proportional to primary task performance via the cross-coupling. Thus, when the pilot keeps primary task performance less than a criterion value (based on the runs with the primary task alone), the secondary task difficulty is automatically increased by increasing the rate of divergence of the secondary task instability. Conversely, when the pilot becomes so busy with the secondary task that the primary task error becomes larger than the criterion value, the secondary task difficulty is automatically decreased. The final "score" is λ_s, the stationary value of the secondary unstable pole (λ) in rad/sec. The scores obtained from this cross-coupled secondary task represent its difficulty; consequently, they also represent the "degree of ease" of the primary task or the excess control capacity available with respect to the primary task. The λ_s scores can be appropriately scaled into proportional workload indices by normalizing them with respect to the maximum sidetask score attainable under full attention conditions (no primary task). In this case, λ_s approaches λ_c, the "critical task" score. The attentional demand of the primary task is then given by

$$AD = 1 - \frac{\lambda_s}{\lambda_c} \qquad (4)$$

The "Attentional Demand," AD, is a dimensionless fraction that can be equated with the average primary control task attentional dwell fraction, η. Its complement, the "Excess Control Capacity," which measures the average fraction of time available for other than the primary task, is

$$XSCC = \frac{\lambda_s}{\lambda_c} \qquad (5)$$

If the side task is taken to be a surrogate for all of the managerial functions, XSCC will be just the average managerial task dwell time fraction $1 - \eta$.

Achieving the critical limiting score in the cross-coupled secondary task indicates a condition of maximum available excess control capacity; the secondary task is a "critical" task in this limiting case. The critical task provides a divergent controlled element form that tightly constrains allowable pilot equalization near the region of gain crossover so that the pilot's effective time delay, τ_e, is the sole determinant of system stability. Thus, pilot activity that demands an increase in τ_e on the <u>whole</u> task will prevent the attainment of the pilot's critical limiting score on the cross-coupled secondary task.

Secondary scores obtained for a variety of primary controlled elements are presented in Ref. 10. Figure 9 shows how the scores for the best gain configurations of each controlled element compare with the Cooper-Harper ratings. In Fig. 9 a score of $\lambda_s = 0$ corresponds to 100 percent of the pilot's attention being devoted to the primary task or <u>no</u> excess control capacity; whereas, a limiting score ($\lambda_s \doteq 5.5$) means that no attention is

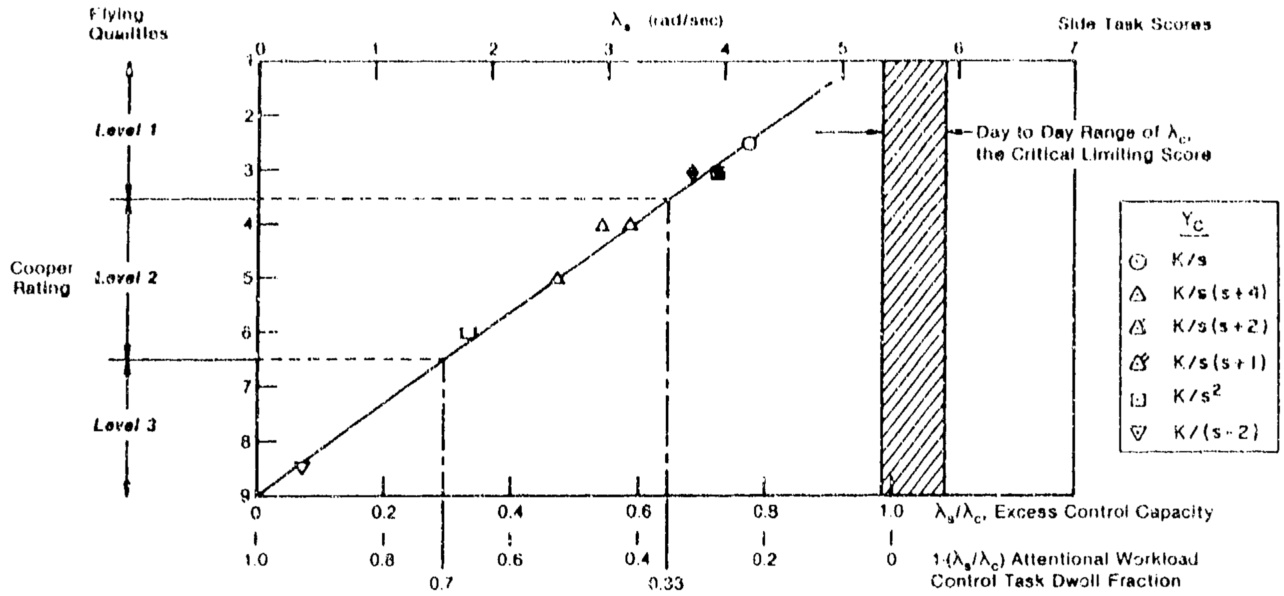


Figure 9. Calibration of Pilot Rating with Attentional Workload and Excess Control Capacity (adapted from Ref. 10)

required to maintain primary task performance, or that 100 percent of excess control capacity is available.

These relationships suggest that subjective pilot ratings can be associated closely with the objective measures of workload provided by the attentional demand and the excess control capacity. The lower (better) values of pilot rating correspond to low attentional demands and large excess capacity to perform other functions. More difficult effective vehicle dynamics that receive poorer pilot ratings of their flying qualities require much more of the pilot's attention and hence leave less capacity for other tasks.

The excess control capacity concept also provides a potential basis for estimating ratings for multiloop situations (Ref. 28). First, assume that the relationship between pilot rating and excess control capacity, $\lambda_n = \lambda_s/\lambda_c$ given by Fig. 9, is applicable to each axis in a multiaxis situation. Then, single-axis capacity or attention, values can be combined to yield the combined axis value by a multiplication process, i.e., the multiaxis excess capacity, λ_{n_m}, is given by the product of the excess capacities for the individual axes:

$$\lambda_{n_m} = \prod^{m} \lambda_{n_i} \tag{6}$$

For $R = A + B\lambda_n$ as a linear fit of the Fig. 9 data, the multiaxis rating R_m will be given by,

$$R_m = A + B\lambda_{n_m} = A + B\prod^{m}\lambda_{n_i}$$

$$= A + B\prod^{m}\left(\frac{R_i - A}{B}\right) \tag{7}$$

$$R_m = A + \frac{1}{B^{m-1}}\prod^{m}(R_i - A)$$

Combined ratings are always greater than (or equal to) individual ratings, since combined λ_n's are always less than any individual λ_n. Also, the maximum value of R_m never exceeds A, i.e., for large $R_i < A$, $\prod^{m}(R_i - A) \to 0$.

The logical value for A is 10.0, and B was determined, using the empirical data, to be equal to −8.3. As depicted in Fig. 10, this results in a good, overall fit to the multi-axis

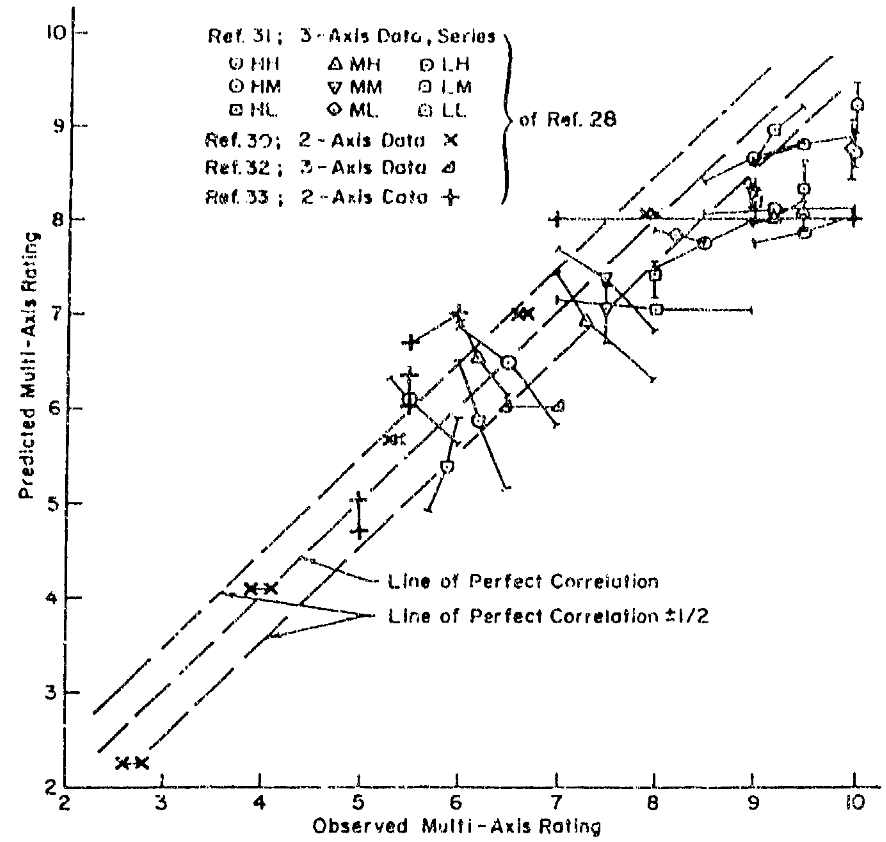


Figure 10. Correlations Obtained with Product Method (Ref. 28)

rating data of Ref. 28. Notice that in its final form the multi-axis rating, R_m, can be computed directly from the single-axis ratings, R_i. Measures or computation of excess control capacity or attentional demand are not required.

For far too many years (1962-1990) the Ref. 26 ("Dander") data were unique as the only generally available experiments in which single and multiple axis ratings were systematically taken and compared. They are the basis of the developments leading to Figs. 7 and 10. A brand new data set (Ref. 29) are now available which enormously extends the data base on which to build and assess new correlation and rating-model building possibilities. The data are unique in including both fixed and moving base multi-axis conditions with comprehensive measures of pilot dynamics (e.g., describing functions) as well as pilot ratings. Reference 29 provides a good start on the development of more elaborate functional relationships between single and multiple axes, but we fully expect many more to follow in the course of time. For the present, the fixed-base data shown in Figs. 11 and 12 serve to further refine and validate the product rule for combining single axis data into multi-axis estimates. Figure 11 illustrates that the product rule continues to work well with these data. Figure 12 provides a useful diagram with boundaries suitable for combinations of ratings in flying qualities "Levels" terms. Among other things, the Level boundaries pertinent to conditions wherein one of the axes is highly rated (the horizontal and vertical boundary lines) implicitly recognize that the combined axis ratings can never be superior to single axis ratings. This is a possibility if the basic formula is used for these cases, e.g., a rating of 1 in one axis combined with a rating of 4 in the other yields a combined rating of 3.5. Reference 29 provides alternative product rules based on various regressions with several parameters, including in some

$$R_m = 10 + \frac{-1^{(m+1)}}{8.3^{(m-1)}} \prod^{m} (R_i - 10)$$

$m = 2$

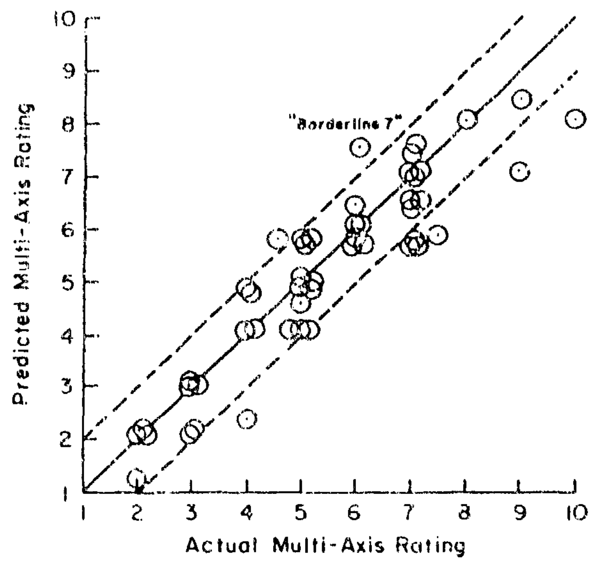


Figure 11. Comparison of Product Rule with Single- and Multi-Axis Pilot Ratings from STI Simulation (Ref. 29)

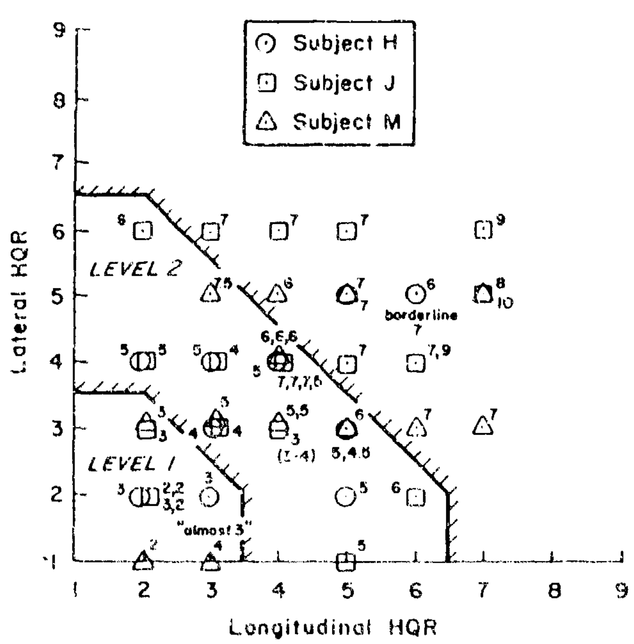


Figure 12. Comparison of Single-Axis (Pitch and Roll) HQRs with Multi-Axis HQRs from STI Simulation (Ref. 29)

cases such quantities as aircraft bandwidth measures. These more extensive and refined formulas can be particularly useful for specific roll pitch control situations.

THE CLINICAL APPROACH TO RATING ESTIMATION

The treatment above has the great merit that, when appropriate measures and experimental correlates are available, a set of pilot rating estimates can be made using relatively simple formulas. The detailed reasons for the rating estimates are inherently buried in the empirical data which serve as bases for the correlations. In other words, the pilot commentary and reasons behind whatever the rating estimate comes out may be quite obscure. To alleviate this difficulty, and to provide an alternative for situations where the data base is insufficient or non-existent, a clinical approach is indicated. Here the characteristics exhibited by the pilot and pilot-vehicle system dynamics are examined for "symptoms" of potential problems. These are then reflected into a summary of properties which amount to a pilot commentary expressed in technical terms.

Consider, for the most elementary situation, that the crossover model is used to accomplish a pilot-vehicle analysis for a given set of effective aircraft dynamics. The data directly available from the analysis includes an estimate of:

- the stability-limited (zero phase margin) maximum crossover frequency, ω_u;
- pilot lead equalization required in the region of crossover to make good the crossover law (measured in terms of pilot amplitude ratio slope, $[d|Y_p|_{dB}/d \log \omega]_{\omega_c}$);
- the nominal full-attention crossover frequency, ω_c.

As developed in Ref. 2, the two crossover frequencies are closely related, i.e.,

	ω_c/ω_u
No pilot lead	0.78
Low-frequency pilot lead	0.66

If, in addition, equivalent forcing function information is available the system steady-state performance can be determined easily (see e.g., Ref. 2, Figs. 9 or 13a).

As might be expected, the most important pilot dynamics correlates with pilot rating are pilot gain and pilot lead. Empirical connections between these are given in Fig. 13. For a particular controlled element there is an optimum controlled element gain which depends on the manipulator dynamics, controller sensitivity, control harmony among axes, etc. No theory yet exists to establish this optimum gain, so it must be determined empirically. Then, curves such as those shown in Fig. 13 can be used to assess any rating decrements from the optimum. By virtue of the ω_c-K_c independence property (Refs. 1, 2) any change in the effective aircraft gain, K_c, will be countered by a change in pilot gain, K_p, to keep the pilot-vehicle system crossover frequency approximately constant. However, either too-sluggish (K_c too small, K_p too large) or too-sensitive conditions can give rise to major decrements. This can be greater than 6 rating points even for the $Y_c = K_c/s$ controlled element dynamics. As can be appreciated from Fig. 13 the optimum is quite broad (changes of plus or minus 50% in either direction are less than 1 rating point for even the narrowest U-shaped curve), so once the controlled element sensitivity is properly adjusted minor controlled element gain changes are not major factors in pilot rating.

The pilot lead equalization required to make good the crossover model has a major effect on the pilot rating. For example, Fig. 13 indicates that the difference between a $Y_c = K_c/s$ controlled element, which requires no pilot lead, and $Y_c = K_c/s^2$, which demands +1 lead units, is a pilot rating decrement of about 3 Cooper-Harper rating points. Considered as idealized systems these correspond, respectively, to "rate command" and "acceleration command" effective vehicle characteristics. Re-examining Fig. 9, the rate command system data point shows a pilot rating, PR = 2-1/2 with an attentional demand of 0.2 while the acceleration command system data point has PR = 6 and a control task dwell fraction of about 0.65. The primary reasons for rating shifts for these data are the amount of lead required and the reduction in system performance (the attainable crossover frequency for the acceleration case is less than that for the rate command situation because of the increased τ_e due to the need for lead generation). In any event, even a best gain acceleration command system will be Level 2 (3-1/2 < PR < 6-1/2) from a flying qualities standpoint. From the descriptive adjectival phrases of Fig. 2 this level of low-frequency lead generation would therefore be interpreted as "considerable pilot compensation" required to achieve adequate performance.

The effects of pilot low-frequency lag equalization have been more difficult to quantify. When the controlled element is a pure gain, the pilot will introduce a very-low-frequency lag to create crossover model properties in the region of pilot-vehicle crossover (Ref. 1). In this case there appears to be no particular rating penalty associated with the lag or trim-like control features. Similar consequences have been seen with high-bandwidth attitude command /attitude hold systems. On the other hand, pitch attitude dynamics in which a conspicuous shelf is present between the $1/T_{\theta 2}$ lead and a lightly damped effective short-period undamped natural frequency, require the pilot to generate a lag or lag-lead feature. This is needed to cope with the lightly damped short-period mode as well as to make the open-loop pilot-vehicle system approach the crossover model. The pilot response data of Ref. 29 demonstrate that the crossover model applies to these situations, and that the pilot does indeed generate a lag or lag-lead. Consideration of the rating data associated with the Ref. 29 experiments, reported in Ref. 23, indicate that, in this case, there is a distinct rating penalty associated with the pilot-generated lag. Also, when the "shelf" length is

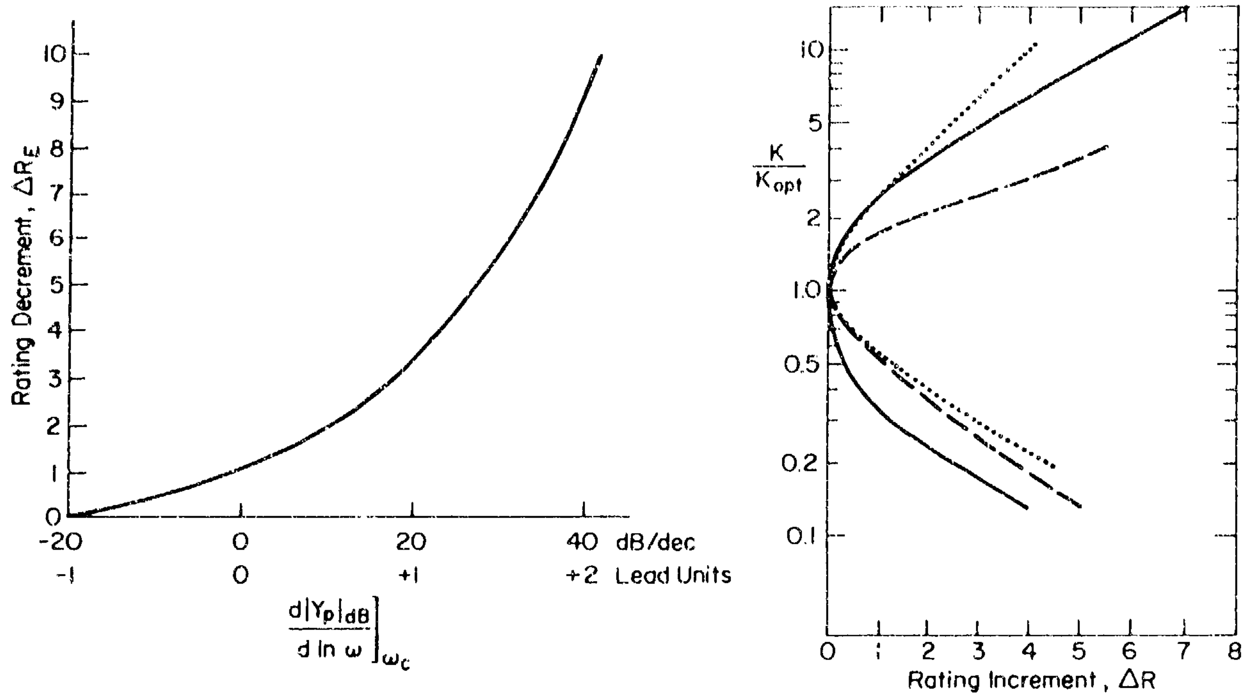


Figure 13. Pilot Rating Decrements as Functions of Lead Equalization and Gain Tracking with Full Attention, Single Axis (Ref. 1)

extreme and/or the effective short period damping ratio too small, the pitch attitude control transfer function becomes a "PIO Syndrome" situation, which can be very poorly rated indeed.

There are, of course, factors other than pilot lead and gain adjustment that affect the pilot rating. In general, flying qualities ratings tend to be given on a global basis which may include several maneuvers in a task complex. Both open-loop (unattended) and closed-loop (attended) piloting operations will be considered in the rating. Concern here is, of course, primarily with the closed-loop piloting aspects. In fact, for stability and control flight testing the important connection is with "flying qualities while tracking" aspects and other precision and/or aggressive tasks which involve tight closed-loop pilot-vehicle control.

The unattended category can be the major factor in determining the acceptable values of very low frequency divergences such as the spiral or a divergent phugoid. It can also be decisive in setting the nature of the "hold" characteristic built into the stability augmentation system. For example, for most flying tasks the attended longitudinal pitch attitude system should, ideally, require no pilot lead equalization. For this to be true the ideal effective airplane dynamics would approximate $Y_c = K_c/s$ in the region of crossover. But for unattended operations a rate command system is not ideal in that an attitude-stable platform is desired. Thus a rate-command/attitude-hold system has superiorr pilot ratings to rate-command/rate-hold.

Another major facet in some nearly unattended or divided-attention operations is "commandability", the ability of the airplane to respond in a precise, orderly, and predictable manner to highly skilled, precognitive pilot command inputs. These inputs are pure commands, functions of time alone, and, as such, are basically open-loop in character. Typical examples are turn entries, step-like (for attitude command systems) or pulse-like (for rate command systems) inputs to adjust attitude, precognitive landing flares, etc. These maneuvers may have to be fine-tuned at the end via closed-loop control, but for an ideal vehicle and a skilled pilot this will not ordinarily be necessary. Again, to the extent that this feature of the airplane's characteristics enter into the rating game, closed-loop dynamics considerations are not explicitly involved.

An important distinction about the unattended factors in the current context is that they may set a base for the pilot rating which does not depend on closed-loop factors. This base can itself shift as the divided attention requirements shift. For example, if managerial tasks take up almost all the available time the effective vehicle dynamics in the unattended state may have to be highly automated, even including path, altitude, or position control. In any event, the closed-loop effects should be thought of as increments from the base level determined from the unattended operation requirements.

Table 1 presents a list of primary factors to which the pilot is sensitive and which, accordingly, underlie pilot rating. Except for pilot lead and gain variation from optimum these factors are not yet individually quantifiable in ratings terms. On the other hand, with modern flight control system technology most of them can be modified by design. Consequently, these system aspects can be profitably compared in competing system studies, and also serve as a useful checklist for interpreting manned simulation or flight test results. As remarked earlier, a pseudo pilot commentary can be constructed by considering them.

In Table 1 both items under "Unattended Operations" and the "Pilot Lead" and "Pilot Gain/Optimum" parts of the "Attended Operations" list have been covered above. The remaining items will be discussed below. Some of the considerations can be developed from the crossover model, while others will require application of the structural-isomorphic pilot model in some form or other.

The "Urgency Adjustment Gain Tolerance" factor can best be understood by considering two limiting cases of controlled element. For the first, consider a K_c/s controlled element form. From the crossover model the pilot dynamic characteristic for this system will be a pure gain plus effective time delay. The closed-loop system for this case can support a range of pilot gains which correspond to crossover frequencies from zero to an octave or so below ω_u with only minor changes in the basic dynamic form of the closed-loop system. In terms of pilot-vehicle system input/output characteristics this will be approximately,

$$\frac{M(s)}{I(s)} \doteq \frac{1}{(s/\omega_c + 1)}$$

As the pilot attention level, urgency, or aggressiveness modifies his gain, ω_c will increase or decrease, with the dominant closed-loop system time constant constant, $1/\omega_c$, waxing and waning in corresponding fashion. Thus, there is a very wide range of excellent closed-loop dynamic response properties available to the pilot which is easily adjusted in direct proportion to his effort. In the words of Fig. 1, "pilot compensation is not a factor for desired performance", and the configuration will be highly rated. For the other extreme imagine a set of effective airplane characteristics which has dynamics in the region of crossover which require precise adjustment of the pilot's lead-lag equalization and gain to make good the crossover model and to close the loop in a stable manner. The pilot can exert closed-loop control, but the dynamic quality and even closed-loop system stability require that his describing function be precisely tuned to offset the controlled element deficiencies. The pilot's compensation in this case will range from "considerable" to "intense", and the configuration configuration will be rated very poorly.

TABLE 1. PILOT-VEHICLE SYSTEM FACTORS IN PILOT RATING

- ATTENDED OPERATIONS
  - Pilot Lead/Lag
  - Pilot Gain/Optimum
  - Urgency Ajustment Gain Tolerance Without Changing Closed-Loop Dynamic Form
  - Stability Margin Gain Tolerances Including Total Available Gain Range
  - PIO Syndrome
  - Neuromuscular System Coupling
  - Attentional Demands/Excess Control Capacity
  - Closed-Loop System Performance
- UNATTENDED OPERATIONS
  - Allowable Fluctuations in Pilot-Control-Precision Demands
  - Equilibrium/Trim Properties (Effective "Hold" Characteristics)

The "Stability Margin Gain Tolerances" factor is most easily described when the pilot-vehicle system is conditionally stable. In this situation the system becomes unstable if the gain is either too low or too high. When the pilot lead-lag equalization is adjusted to maximize this range (which will ordinarily provide crossover model-like features in the nominal crossover region), there is a "total available gain range" (TAGR) through which the pilot can maintain some semblance of closed-loop control. Clearly, the more narrow this range becomes the more difficult the pilot's adjustment and the worse his rating will become.

The "PIO Syndrome" has been introduced in connection with pilot lag equalization. When the pitch attitude transfer function exhibits a shelf and a lightly damped short period the pilot has to introduce a precisely located and tuned lag or lag-lead characteristic to accomplish tight closed-loop control. If he inadvertently drops the precision adjustment of his equalization and switches to a proportional control characteristic, as can happen when triggered by an unexpected major upset or need for divided attention, the pilot-vehicle system may then become unstable. This is a relatively common condition for a PIO, thus the appellation.

The "Neuromuscular System Coupling" factor can become important when the low-frequency effective airplane dynamics are excellent but the closed-loop system gain margin in the region of the neuromuscular actuation mode (ω_{NM} in Fig. 1) is reduced. This is covered in Refs. 30 and 31. The resulting closed-loop system instability is high frequency, 2-3 hz. It is one explanation for "roll ratchet".

The "Attentional Demands/Excess Control Capacity" factor is primarily related to divided attention operations. When the control task itself is responsible for using most of the pilot's excess control capacity the reasons for this are invariably due to factors already covered. When the managerial, communication, planning, and other non-control tasks consume too much of the pilot's available attention, pilot ratings will suffer. The obverse of this is that the effective vehicle dynamics must be very good in order to require a minimum of attention. The ratings for control alone should, in general, be superior, and the unattended operations factors would be good as well.

The last Table 1 factor to be discussed is "Closed-Loop Performance". Many facets of task performance stem directly from mission requirements and are hence mission-specific. Using the complete Cooper-Harper rating sequence (Ref. 9) the status of the pilot-vehicle system relative to mission requirements is the very first thing the pilot assesses before more detailed ratings are established. Average error performance in command and regulation tasks can be calculated with all the pilot models once these inputs are defined. These estimates can serve as one basis for "Closed-Loop Performance" in flying qualities assessments.

There are other, more general, closed-loop dynamic performance aspects which should also be considered in flying qualities assessments. These are listed in Table 2. The first two are simple statements of closed-loop dynamic response quality

TABLE 2. DESIRABLE CLOSED LOOP DYNAMIC FEATURES

- ADEQUATE CLOSED-LOOP DAMPING, $\zeta_{CL} \geq 0.35\text{-}0.50$
- AVOIDANCE OF CLOSED-LOOP MID-FREQUENCY DROOP, NUISANCE OR SUBSIDIARY MODES
- MULTILOOP CONTROL VIA SERIES STRUCTURE FOR SINGLE CONTROL
- FREQUENCY SEPARATION OF INNER, OUTER LOOPS
- COOPERATIVE/DECOUPLED CONTROL EFFECTORS
- SIMPLE CROSSFEEDS TO DIRECTLY NEGATE SUBSIDIARY RESPONSES
- CONTROL HARMONY

They, in essence, suggest that any second-order dominant closed-loop pilot-vehicle system mode have a damping ratio greater than 0.35 to 0.5. The requirement to avoid a closed-loop mid-frequency droop is tantamount to a prescription of one dominant mode per axis, for the droop will show up as an additional minor mode with a longer time constant. The Neal-Smith criteria (Ref. 32), for example, call specific attention to the mid-frequency droop and require that it be less than 3 dB to achieve Level 1 ratings. By way of example, a 3 dB mid-frequency droop can be associated with the presence of a minor mode comprising a single dipole pair in the closed-loop pilot-vehicle system (with the |zero/pole| < 1.41) supplementing the major dominant mode.

The remaining two desirable closed-loop dynamic features are associated with multi-loop, single control axis situations. Common examples of this include: 1) the control of altitude wherein altitude error is the outer loop feedback and pitch angle is an inner loop; and 2) however control, as shown in Fig. 3. "Desirable" aspects of such systems include the qualitative feature that a "series" (rather than parallel) closure of the outer loop is possible in the presence of an inner loop system which serves both independently and as a means to equalize the outer loop. Thus the pilot closure of a pitch attitude loop satisfies an attitude control function and gives rise to an effective outer, altitude control, loop which needs very little if any further pilot equalization. This is supported in a more quantitative sense by the suggestion for the separation of crossover frequencies for multiloop systems with series pilot elements.

Interaxis considerations apply to the last three items listed in Table 2. Ideally the control effectors, as seen by the pilot, should be cooperative or actually decoupled. Potentially cooperative cases lead to the desirable crossfeed feature listed. This accounts for the possibility possibility of pilot-induced crossfeeds to reduce or eliminate subsidiary modes or response quantities. A common example is an aileron to rudder crossfeed for turn coordination. The last feature on the list of desires is control harmony, which relates primarily to multi-axis control conditions with a common manipulator. Force and position gradients, pre-loads and centering springs and other manipulator features between elevator and aileron need to be in proper balance so that the effective controlled element gains in each axis are near optima, interaxis crosstalk is minimized, etc. Just as with the setting of controlled element optimum gain, control harmony is a subject of experimental determination.

As a consequence of flying qualities analyses using pilot-vehicle analysis to examine the factors of Tables 1 and 2, the analyst can develop a set of conclusions and arrive at a wide variety of issues and possible problems. Table 3 illustrates the type of problems that might be uncovered by such examinations for the case of either lateral-directional or longitudinal attitude and path control.

TABLE 3. TYPICAL PILOT CENTERED REGULATION PROBLEMS

- CLOSED-LOOP DOMINANT MODE INADEQUACIES
  - Too Slow
  - Lightly Damped
  - Low/Mid Frequency Droop
  - Too Many Modes Dominant
- ATTITUDE CONTROL
  - Inadequate Bandwidth
  - Inner-Outer Loop Equalization Conflict/Interaction
  - Low Static Gain
  - Over-Sensitivity to Gain/Equalization
- PATH CONTROL
  - Performance Reversals
  - Inadequate Bandwidth
  - Inadequate Separation of Path and Attitude Responses
  - Difficult or Conflicting Crossfeeds/Coupling Interaction
  - Excessive Depletion of Safety Margins
  - Low (High) Effective Path Gains

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APPENDIX

SIMPLE RUBRICS FOR PILOT-VEHICLE ANALYSIS

- **CROSSOVER LAW**

$$Y_p Y_c \doteq \frac{\omega_c e^{-\tau_e s}}{s} \text{ Near } \omega_c$$

- **MAXIMUM FULL-ATTENTION ω_c ($\phi_m \doteq 0$)**

$$\omega_u \doteq \frac{\pi}{2\tau_{eff}}$$

- **DIVIDED ATTENTION CONTROL DWELL FRACTIONS, η**

LEVEL	η	EFFECTIVE Y_c MUST SUPPORT ϕ_M (DEGREES)
1	$< 1/3$	> 60
2	$1/3 < \eta < 0.7$	$25 < \phi_M < 60$

- **PURE GAIN ($Y_p = K_p e^{-\tau_p s}$) CLOSURES**

If Closed-Loop Dynamics are Good over Wide K_p Range

— Effective Aircraft Dynamics OK *if* K_c OK

If Closed-Loop Dynamics Exhibit Difficulties

— Aircraft Dynamics which inhibit good closures are revealed as potential flying qualities problems and parameters

IMPLICATIONS OF THE CROSSOVER MODEL

- **DUALITY OF Y_c AND Y_p FOR CLOSED-LOOP CONTROL**

  ∴ Can specify open-loop effective aircraft dynamics needed for "Good" pilot dynamics

- **FULL-ATTENTION ω_c > DIVIDED ATTENTION ω_c**

  ∴ Single Loop $\omega_c \geq$ Multi-Loop ω_c's

- **DIVIDED ATTENTION —> FULL ATTENTION OPERATIONS IMPLY**

  — Wide Range of ω_c, From $0 \longrightarrow \omega_u = \dfrac{\pi}{2\tau_{eff}}$

  — Effective aircraft dynamics which support large phase margins

- **GOOD PILOT RATINGS REQUIRE MINIMUM PILOT ANTICIPATION**

  ∴ Desired $Y_p \sim K_p e^{-\tau_p s}$

  ∴ Desired Crossover Region $Y_c \sim K_c/s$;

  For Divided Attention K_c/s Region should extend from $\omega_{c_{Divided}} \longrightarrow \omega_{c_{Full}}$

- **METRICS FOR $\omega_{c_{MAX}}$ PURE GAIN ($Y_p \sim K_p e^{-\tau_p s}$ CLOSURES ARE "AIRPLANE BANDWIDTH" PARAMETERS**

AN INITIAL STUDY INTO THE INFLUENCE OF CONTROL STICK CHARACTERISTICS ON THE HANDLING QUALITIES OF A FLY-BY-WIRE HELICOPTER

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SUMMARY

A piloted experiment was flown using the Institute for Aerospace Research Bell 205A variable stability helicopter. The experimental variables were the static and dynamic characteristics of a conventional centre-mounted cyclic controller. The cyclic controller characteristics were changed by varying the mass and spring gradient to provide five basic cases, while for each case the dynamics of the stick were varied to provide critically damped, underdamped and overdamped models. Two pilots were asked to fly a variety of tasks designed to exercise three fundamental modes of helicopter flight, high frequency stabilisation, gross single axis tasks with off axis stabilisation and simultaneous multi axis control. The stick sensitivity was adjusted in proportion to the spring gradient to give constant static sensitivity with respect to applied force. A first order filter was incorporated on an optional basis to reduce the command response bandwidth of the roll channel to the Level 1/Level 2 boundary of the ADS-33C criterion for divided attention operation. The results achieved indicate that cyclic stick characteristics are of considerably less importance than had been previously thought, that large values of overdamping can be tolerated even in low frequency sticks, but that underdamped sticks should be avoided especially if the resonant frequency of the stick is close to an undesirable and easily excited aircraft mode. There was a suggestion that a boundary based on undamped natural frequency also existed. The results did not support the contention that inertia alone is enough to specify an acceptable/unacceptable boundary for stick design.

LIST OF SYMBOLS

a	Disturbance function weighting coefficient	
F_a	Pilot's applied force	lbf
G_p	Gain on roll rate error	
G_s	Sensitivity gain	
G_{ff}	Forward feed function	
G_φ	Gain on roll attitude error	
I	Inertia	slugs
K_s	Spring gradient	lbf/ft
K_v	Damping coefficient	lbf/ft/sec
L_δ	Roll control power derivative	rad/sec^2
p	Body axis roll rate	rad/sec
p_c	Commanded roll rate	rad/sec
$\dot{p}_c$	Commanded rolling acceleration	rad/sec^2
s	The Laplace operator	
t	Time	sec
x	Generic linear displacement	ft
δ	Generic control displacement	in
τ_e	Equivalent time delay	ms
φ	Roll attitude	deg
φ_c	Commanded roll attitude	deg
φ_d	Attitude disturbing function	deg
θ	Pitch attitude	deg
ζ	Damping coefficient	1/sec
ω_n	Undamped natural frequency	rad/sec

INTRODUCTION

A physical manipulator, operating a conventional control run to move a control or signal an actuator, may be characterised by a wide variety of parameters associated with, and to a large extent determined by, the mechanical design of the system. In the early days of the helicopter the physical control runs acted directly on the rotor blades and the forces reflected back to the pilot depended on such things as the inertia in the system, the friction characteristics it displayed, the aerodynamic forces generated by the rotor blades and the relative positions of the feathering, flapping and drag hinges. Even in small machines, these forces could become very large under some flight conditions and moreover, unlike the situation found in fixed wing aircraft, they had no fundamental relationship to the stability of the machine or its departure from trim. This situation required the designer to give the pilot some form of assistance in changing the main rotor blade incidence. Initially, a variety of aerodynamic devices were to be seen and very early in the life of the helicopter, powered or power assisted controls became almost universal.

Lacking a basic aerodynamic relationship between force and stability early designers tended to leave the cyclic controller with no force feel whatsoever (the classic *limp noodle)*. However, as helicopter design matured and particularly with the advent of Stability Augmentation Systems and Automatic Flight Control Systems, a need arose to have a controller which had reasonably well defined self-centring and which could be abandoned for short periods without its falling away from its trimmed position. Various force feel systems were introduced, yet it is still possible to find pilots who will immediately disable such systems and continue to fly the limp noodle until such time as they wish the release the stick.

Progressing to the future, and anticipating the production of a full fly-by-wire helicopter, it is an appropriate time to address the characteristics of the cyclic controller for such applications. This activity is particularly needed since the guidance concerning static characteristics given in the recently published ADS-33C[1] appears to be based on very old data, and while it may be appropriate for simple unaugmented helicopters, there is no evidence that it is applicable for the coming generations of machines. Moreover, this document gives no guidance whatsoever as to the acceptable or unacceptable dynamic behaviour for such controllers. Dynamics are mentioned not at all in the section on Controller Characteristics (3.6.1) and one finds only a passing mention of them in the discussion of the small amplitude response bandwidth criteria thus:

> *"It is desirable to meet this criterion for both controller force and position inputs. If the bandwidth for force inputs falls outside the specified limits, flight testing should be conducted to determine that the force feel system is not excessively sluggish"*

Recent experimental evidence[2],[3], obtained at the Institute for Aerospace Research (IAR) as secondary observations in other bandwidth related studies, indicates that the force to attitude bandwidth has less significance than was previously thought. This evidence was sufficiently powerful to suggest that a formal study of centre mounted cyclic controller characteristics should be undertaken. This paper will describe the initial exploratory investigation in this study. While a complete analysis of the experimental results has not yet been completed, the initial findings are presented together with the body of in-flight data.

GENERAL DISCUSSION

If one considers Figure 1 to be a generic representation of the full closed loop task facing a helicopter pilot, the stick appears to be a discrete dynamic element in the system. However, little is known in general terms about the exact function of this element, or how the human pilot interacts with it. It is not even clear whether the pilot and stick should be considered separate elements or whether they behave as a single entity.

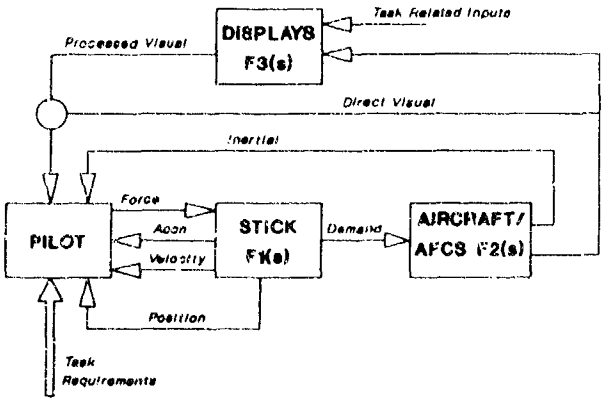

Figure 1: Generic Helicopter Piloting Task

There exists a considerable body of work relating to manipulators for fixed wing applications. This ranges from the early work of McRuer and Magdelano[4], and McRuer and Krendel[5] on human pilot dynamics, through the experimental work performed by a variety of investigators on the Calspan NT-33A to the comprehensive analysis of Johnston and Aponso[6]. The majority of this work concentrates on roll tracking performance, generally as a single axis task and while it provides excellent insight into the behaviour of the human neuromuscular sub-system in a task of great importance in fixed wing flying, it does not address those open loop or pre-cognitive inputs used by pilots to perform, or at least initiate gross manoeuvring tasks.

The practice in the fixed wing world of considering the roll and pitch axes separately is quite understandable and well established as a valid procedure. In fixed wing aircraft the two axes are almost completely de-coupled, have quite different natural response types and control responses that can vary one from the other by over an order of magnitude. The helicopter, using the same force or moment generators in both pitch and roll has the same response type in each axis and response magnitudes that vary only by the ratio of the moments of inertia about the X and Y axes, typically 1:3 in a single rotor helicopter.

The piloting task in the helicopter is quite different from that in a fixed wing aircraft; it is frequently required to control pitch and roll simultaneously in similar and coordinated ways. Moreover, the helicopter pilot at and around the hover is rarely faced either with a single axis task, or with the need for sustained tracking (excepting that precision hover has many of the attributes of the high frequency portions of a tracking task). These considerations may explain why there is a relative paucity of previous work on manipulator characteristics relating to helicopters. However, during the course of this experiment a new work, Reference [7], was published, which contained flight and ground based simulation data relating to roll axis stick characteristics in a classic single axis experiment. Since the data acquired at the IAR to that point did not seem to correlate with those published in that paper, it was decided to add a roll regulation task to this study. The purpose in so doing was to try to determine if the tasks used were responsible for any divergence in observations or whether other factors were involved.

THE AIRBORNE SIMULATOR

The Flight Research Laboratory (FRL) of the IAR operates a variable stability Bell 205A, known as the Airborne Simulator. The aircraft has been extensively modified for this role in such a way that in its fly-by-wire mode the right seat pilot creates inputs to a high speed computing system which in turn drives full authority high bandwidth dual-mode actuators. A comprehensive set of state sensors is used to derive feed-back signals to create the desired aircraft responses in all axes. A safety pilot in the left seat has conventional controls mechanically linked to the same actuators. The flight control computers operate at 64 Hz (a 15.625 ms cycle).

Evaluation Pilot's Controllers

The cyclic stick and rudder pedals at the evaluation pilot's station are themselves simulations, based on the well known force feed-back principle. The forces applied by the pilot are fed to a dedicated, purpose built analogue computer which computes stick acceleration, velocity and position and uses these signals to position hydraulic actuators attached to the controllers. Both the main control linkage and a backlash element can be modelled on this system. The following parameters are variables in the analogue computation:

a. Backlash	**b. Main Linkage**
Extent	Inertia
Inertia	Viscous damping
Spring gradient	Spring gradient
Coulomb friction	Coulomb/static ratio
	Static Friction
	Break-out force
	Hard-stop limits
	Symmetry

For this study a simple second order stick was approximated within the capabilities of the analogue computation by setting the Extent of the Backlash model to zero and setting all the main linkage variables to zero with the exception of Inertia, Viscous damping, Spring gradient and Hard-stop limits. For cases requiring an isometric stick the Hard-stop limits were also set to zero

Aircraft Control Systems

Yaw and Collective. Yaw control was rate command, heading hold at the hover, blending into turn coordination as the IAS passed through 40 kt accelerating and returning to rate command heading hold at 35 kt decelerating. Collective control was direct drive, scaled one for one with the standard Bell 205 control.

Pitch and Roll. Both pitch and roll were rate command systems, identified by frequency sweep analysis as being:

$$\frac{\varphi}{\delta} = \left[\frac{53.7}{s\,(s\,+\,4.5)}\right] e^{-0.085\,s} \qquad (1)$$

and

$$\frac{\theta}{\delta} = \left[\frac{61}{s\,(s\,+\,3.2)}\right] e^{-0.110s} \qquad (2)$$

This resulted in values of bandwidth and Tau-p well within the ADS-33C Level 1 specification in both channels. To avoid excessive length, only the roll channel will be discussed in detail, it being understood that the pitch channel remains in the same relative relationship to the roll channel at all times. The HQR assignments refer to the entire aircraft, however, not simply to the roll axis.

As an addition to the basic inner loop, a first order filter was designed to bring the command response of the roll channel onto the Level1/Level 2 boundary, giving a transfer function :

$$\frac{\varphi}{\delta} = \left(\frac{8.66}{s\,+\,8.66}\right)\left[\frac{53.7}{s\,(s\,+\,4.5)}\right] e^{-0.085} \qquad (3)$$

As will be seen in later figures, this had the required effect and since only the command path was modified while the disturbance rejection characteristics of the plant remained unchanged, any changes in pilot opinion between the filtered and un-filtered cases can be attributed entirely to the change in the quality of the command control system and not to any change in the task content due to a difference in the aircraft's gust response.

Lateral Model Following Control System

To replicate the experiment described in Reference [7] as closely as possible, a model following control system was built in the lateral channel only, this was a relatively crude design based on the scheme shown in Figure (2).

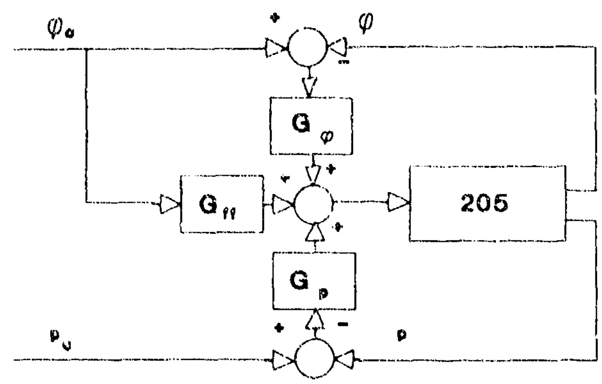

Figure 2: Model Following System

Assuming a very simple first order model of the Bell 205 A to be:

$$\frac{\varphi}{\delta} = \frac{L\delta}{s\,(s\,+\,L_p)} \qquad (4)$$

then G_{ff} is constructed by saying

$$\frac{\varphi}{\varphi_c} = 1 = G_{ff}\left[\frac{L\delta}{s^2 + L_p s}\right] \qquad (5)$$

from which

$$G_{ff} = \frac{(s^2 + L_p s)}{L\delta} = \frac{(\dot{p}_c + L_p p_c)}{L\delta} \qquad (6)$$

The gains G_φ and G_p were set empirically (based on existing knowledge of the Airborne Simulator) to 1.25 and 2.0 respectively. This system proved to have fairly good model following properties, identified by the Bode plot at Figure (3)

Driving this system with a command model (Eqn (1) without the delay element) raised no problems, the model rolling acceleration and roll rate were passed directly to the forward feed function but applying a sum of sine waves disturbing function proved difficult. The second derivative, ($-\,\omega^2$ *Sin* (ωt)) of the function reaches excessively high values for direct drive of the aircraft's actuators even

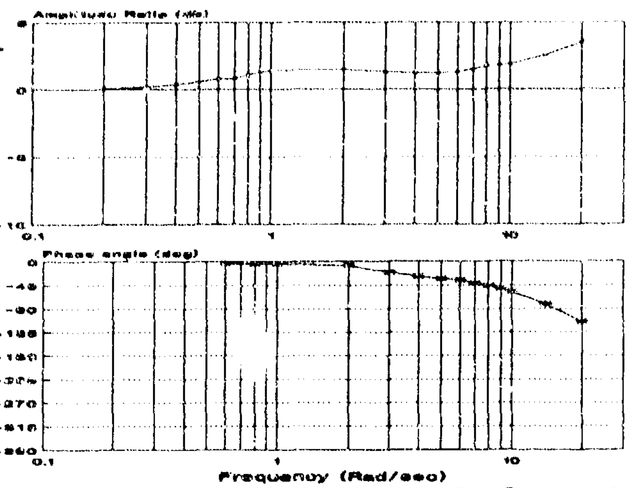

Figure 3: Model Following Performance Phi/Command

when scaled by the weighting array shown later. It was therefore necessary to scale this component by 0.25.

METHODOLOGY

Stick Design

While a conventional physical control system consists of a variety of both linear and non linear elements, there is no a priori reason why a controller for a modern helicopter should have similar characteristics or that they should share the same relative magnitudes. Since this first experiment was exploratory, it was decided to use a very simple second order model, the only non linear element *in the controller* being hard stops at full travel and these are reached only during slope landings. Stick models were built based on the classic mass, spring, damper system, the variables being inertia, spring gradient and damping coefficient. The mass, spring, damper system may be characterised by:

$$I\ddot{x} = -K_s x - K_v \dot{x} + F_a(t) \tag{7}$$

This is usually easier to use in transfer function form when it becomes:

$$\frac{x(s)}{F_a(s)} = \frac{1/I}{s^2 + (K_v/I)s + K_s/I} \tag{8}$$

Which may be written in the standard form:

$$\frac{x(s)}{F_a(s)} = \frac{1/I}{s^2 + 2\zeta\omega_n s + \omega_n^2} \tag{9}$$

by writing $\omega_n = \sqrt{(K_s/I)}$ and $\zeta = K_v/2\sqrt{(K_s I)}$.

When considering the relationship between the human operator and the controller he has to manipulate, it becomes clear that he has to adapt to both the static and the dynamic characteristics of that controller. In this sense, the static characteristics can be described simply by the force gradient and the displacement to controlled element relationship embodied in the system. However, to make all the possible elements variables would have made the experimental matrix too large and unwieldy for a single experiment. It was therefore decided to use a variety of controllers having different spring gradients, but a constant *applied force to swashplate angle gain under static conditions*. This was achieved by applying a gain to the stick position signal consisting of the product of a constant sensitivity gain Gs and the spring gradient value K_s, making the transfer function of Eqn (9) become:

$$\frac{\delta(s)}{F_a(s)} = G_s\left(\frac{\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2}\right) \tag{10}$$

which has constant zero frequency gain for all ω_n and ζ.

While this has the effect of not requiring additional sustained applied force on the part of the pilot as spring gradient was increased, it nevertheless reduced the magnitude of the biokinaesthetic feedback cues available to him. (Note that when using a Rate Command control system it is necessary to ensure that the pilot has full control over the swashplate angle at steady state to permit normal off-level operations). The limiting case here is the isometric stick, and an isometric model was used in the experiment.

Three spring gradients were used for the displacement controllers, 1.5, 3 and 9 lb/in respectively. These were chosen on the basis of experiences at the IAR during previous experiments using high gain feed-back control systems in the Airborne Simulator. It has been observed that the very light (0.5 to 1.0 lb/in) spring gradients typically used with unaugmented or lightly rate damped configurations were too prone to bio-inertial pick-up to be suitable for use with high gain feed-back systems. These gradients, read in conjunction with equation (10) above led to a selection of G_s to make the stick deflections for full swashplate deflection: ± 7.5 ,±3.75 and ± 1.25 in respectively, which may be compared to the ± 6.5 in found in the standard Bell 205A.

Three inertia settings were used in combination with these spring gradients to provide five basic cases (Labelled 0 to 4) of different undamped natural frequencies. For each Case the magnitude of the viscous damping coefficient was varied to alter the damping ratio of the stick. Three damping levels were used at each Case, underdamped (U), critically damped (C) and overdamped (S). Note that limitations in the stick computer prevented the realisation of an overdamped model in Case 3.

The immense complexity of the Airborne Simulator's stick model made the use of nominal values of the potentiometers inaccurate in setting model parameters, however, the settings used were repeatable to very close tolerances and the final stick models were documented by the frequency analysis of undamped free oscillation and hand excited frequency sweeps. The results of this documentation are shown in Table 1, while Figure (4) shows the results of the free oscillation tests and Figure (5) gives an example of case documentation frequency sweep analysis.

Isometric Cases. Output from the isometric stick was passed through a second order digital filter to provide a

Case	K_θ lbf/in	Mass slug	ω_n r/s	ζ	τ_e ms
Displacement Sticks					
0C	1.5	0.610	5.4	0.58	210
0S				1.02	377
0U				0.22	82
1C	3.0	0.424	9.2	0.75	162
1S				1.16	252
1U				0.34	74
2C	3.0	0.184	13.9	0.67	96
2S				1.21	146
2U				0.26	37
3C	9.0	0.336	17.9	0.71	79
3U				0.32	36
4C	9.0	0.157	26.2	0.82	63
4S				1.72	131
4U				0.37	28
Isometric Sticks					
S0			16	0.707	88
S1			12	0.707	118
S2			8	0.707	177
S4			4	0.707	353

Table 1 Summary of Stick Configurations

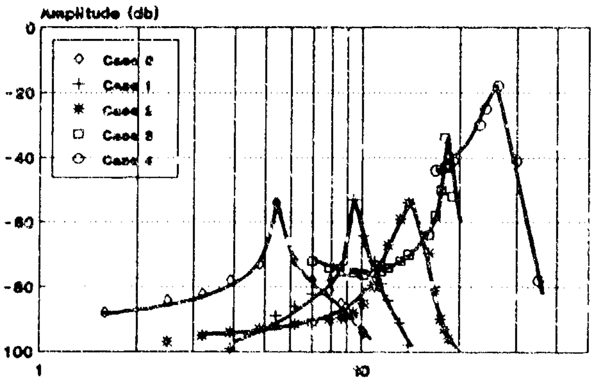

Figure 4: Free Oscillation Analysis, Lateral

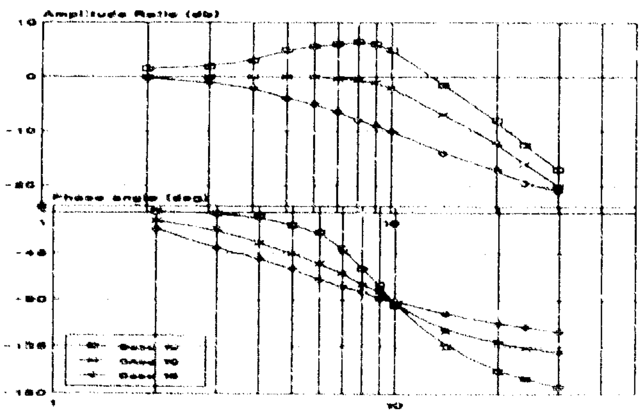

Figure 5: Sample Stick Documentation, Case 1

similar range of dynamic lags as were available from the displacement models. Four cases were used, labelled S0,S1,S2 and S4, having filter break-points set at 16,12,8 and 4 rad/sec respectively. These cases are also listed in the Table.

Total Aircraft Responses

Figures (6) and (7) show the experimental array of force response configurations on the map of bandwidth and Tau-p, the position of the displacement bandwidth is also shown for both the filtered and unfiltered cases. All cases were documented by the analysis of hand flown frequency sweeps and all were phase margin limited.

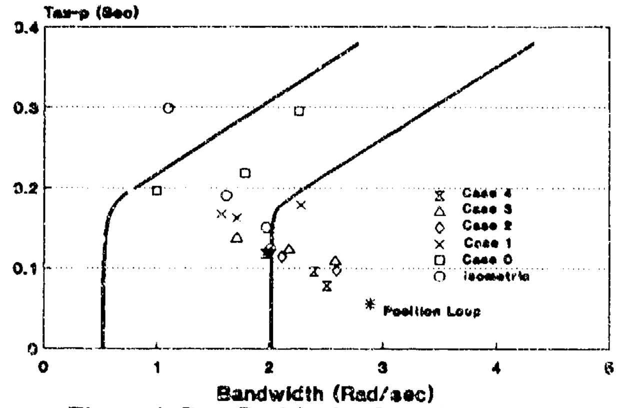

Figure 6: Case Positioning Map, Unfiltered

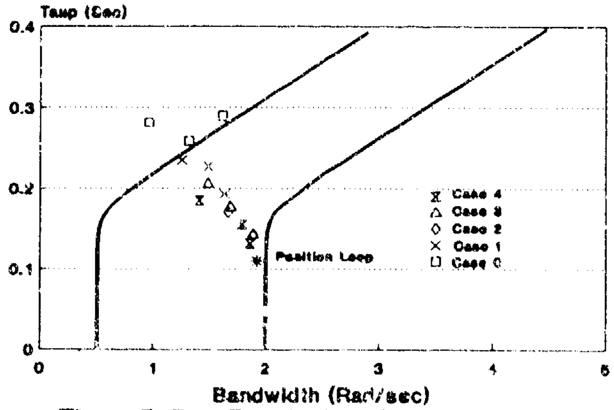

Figure 7: Case Positioning Map, With Filter

Tasking

General Hover Manoeuvring. Particularly at low speed or at the hover, helicopter piloting tasks can be considered to be of one of three types, high frequency stabilisation, gross single axis control with off axis stabilisation and simultaneous multi axis control and stabilisation. A set of tasks was used for this experiment to exercise these modes of operation, they comprised

Precision Hover	Precision Landing
Accelerate/Stop	Pirouette
Lateral Sidestep	Hesitation Turn
Off Level Landing/Take Off	

The details of how the pilots were asked to perform these tasks and the "Desired Performance" limits are shown in detail at Appendix A. Adequate performance was accepted as being that the task could be accomplished in a safe manner. The tasks were flown as a continuous exercise, taking between 350 and 700 seconds to accomplish and the pilots were asked to perform the sequence a minimum of two times before applying Cooper Harper[8] ratings (CHR) to the individual tasks. Repeats, of the entire sequence or of specific tasks, were at the evaluation pilot's discretion. The various stick models were presented in a pseudo random sequence and the pilots were unaware of the stick characteristics until shortly before the fly-by-wire system was engaged. Before commencing each set of evaluations the pilots were permitted a period of free manoeuvring to adjust to the new stick.

Lateral Tracking Task. A lateral regulation task, driven by the same attitude disturbing function as defined in Reference [7], was installed in the Airborne Simulator. The disturbing signal used the weighted sum of 9 discrete sine wave frequencies (Reproduced for completeness in Table 2).

The roll disturbing function was:

$$\varphi_d = 3.42 \sum_{i=1}^{9} a_i \, Sin(\omega_i t)$$

The values of ω_i and a_i were:

i	ω_i	a_i
1	0.47	1.0
2	0.70	1.0
3	1.16	1.0
4	1.86	0.5
5	3.49	0.2
6	6.98	0.05
7	11.17	0.025
8	13.96	0.015
9	18.62	0.010

Table 2: **Disturbing Function Details**

Since the control system for this experiment was of the rate command type it was not possible to drive the aircraft with the attitude signal itself, the derivative being used in its place. The task was flown at the hover and the pilots were required to maintain wings level during the disturbances. Following Reference [7], a pilot warm-up period was followed by several 27 second recorder runs.

Since this task was not particularly close in form to that described in Reference [7], a second version was implemented using the model following control system described earlier. In this case, the task was flown under simulated IMC and provision was made to permit the evaluation pilot to see either the true aircraft attitude or the model attitude. Since it is not possible to operate in a split axis system in the Airborne Simulator the evaluation pilot was obliged to fly the entire aircraft. However, his off axis tasks were made easier by making the pitch response attitude command and flying the exercise at 60 kt IAS well clear of the ground so that a fixed collective setting could be used.

Evaluation Pilots

The evaluation pilots in this study were two IAR staff pilots, both of whom are experienced, military trained test pilots. Pilot A has a total flight time of 9800 hours of which 1400 were in helicopters while Pilot B (total flight time 4450 hours) has recently been cross-trained to rotary winged aircraft and had only 125 hours in helicopters at the end of the experiment.

RESULTS

Piloting Techniques

The two pilots displayed quite different techniques in flying the Airborne Simulator in this exercise. Pilot A always took a full handed grip on the cyclic controller, with the rear of the hand-grip in contact with the thumb/index finger cusp, while Pilot B held the stick almost at the bottom of the grip using only the forward portion of his fingers. Pilot A made his control inputs with forearm and wrist, while Pilot B appeared to make small amplitude inputs with his fingers, reserving the larger muscle groups for large inputs. Pilot B frequently commented that he preferred the high gradient sticks (Cases 3 and 4) to the others because he preferred not to make large displacement inputs.

Handling Qualities Evaluations

General Hover Manoeuvring. The complete set of Cooper Harper ratings obtained during this experiment are given at Appendix B while figures (8) and (9) give an overall appreciation of the results as an average of CHR for all tasks for both the unfiltered displacement stick and the isometric controller. To avoid clutter, the results obtained from the two pilots are plotted separately.

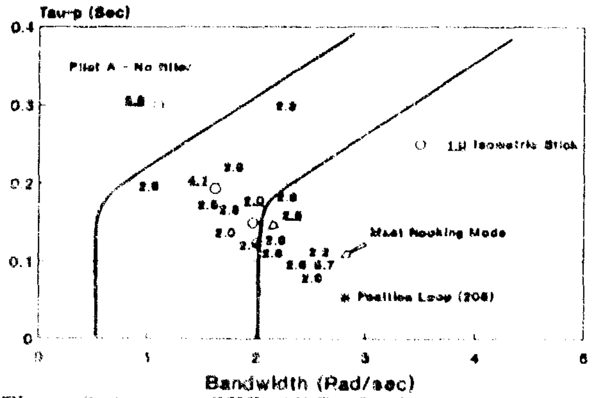

Figure 8: Average CHR, All Tasks, Pilot A, No Filter

Significant points of interest are the Level 2 ratings assigned well within the Level 1 boundary by both pilots.

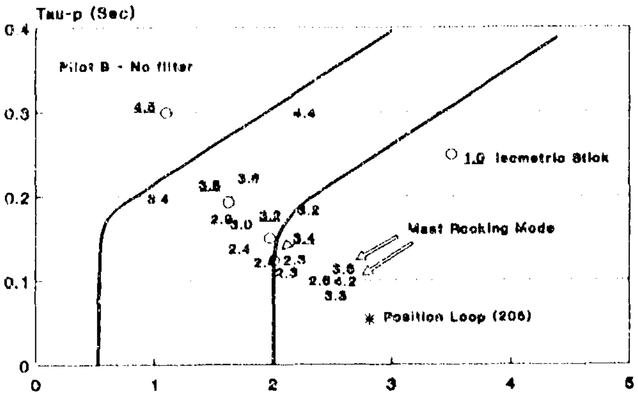

Figure 9: Average CHR, All Tasks, Pilot B, No Filter

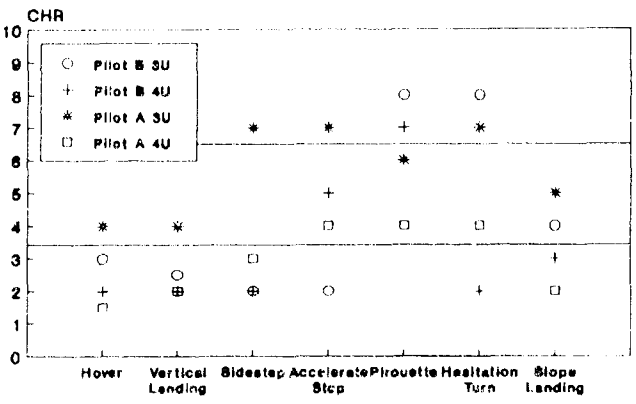

Figure 10: Cases 3U and 4U by Task

These occurred in the cases of the high frequency, underdamped sticks (Cases 3U and 4U) and were caused by the pilot/stick coupling with the Bell 205 mast rocking mode of oscillation. This type of coupling is not uncommon in this aircraft and results in a disturbing, occasionally divergent oscillation at about 15 rad/sec with the pilot sometimes having to abandon the controller to permit it to damp out. It is interesting to note, though, that if these average ratings are examined by their components (Figure 10) it can be seen that the sticks were well liked except in those tasks which induced the coupling. Generally the Level 3 ratings were assigned when the coupling was so severe that the pilot felt obliged to relinquish the cyclic controller for a short period to allow it to damp out before resuming the task. Pilot comments concerning the Level 2 assignments indicated that the CHR 4's were because the coupling was "nibbling at me" or "I feel that it is just about to bite", Level 2 ratings numerically higher than this were caused by the coupling taking place but damping out without the pilot having to abandon control, yet causing him to reduce his desired level of aggression.

Apart from the cases discussed above, the CHR assigned to the displacement sticks are dominantly Level 1, even when the force/attitude bandwidth and Tau-p are almost on the Level 2/Level 3 boundary. Note that in Figure 6 Pilot A assigns NO Level 2 ratings to any but the Cases mentioned above, Pilot B, however, produces two average CHR's at Level 2 (One 4.4, one a marginal 3.6), both for the very low frequency stick. His comments indicate that the 4.4 average was due to a sensation of the stick "fighting back" while the 3.6 (three CHR 3, four CHR 4) was because he disliked having to make physically large inputs to achieve the desired response. His evaluation of the third version of Case 0 was very similar to that of the other two, and equally marginal (four CHR 3, three CHR 4).

In-Line Filter. Placing a first order filter between the stick displacement signal and the input to the inner loop, which brought the command response characteristics of the plant onto the Level 1/Level 2 boundary of the Bandwidth/Tau-p criterion had the effect of degrading the handling qualities to a borderline Level 1/2 (as shown in Figure (11)). Since this is what would be predicted by the positioning of the plant it would appear that the quality of the command responses dominated the stick characteristics. Even when the force/attitude characteristics place the model well into the Level 3 area, one pilot still considers it to be Level 1, while the other places it just over the Level 1 boundary.

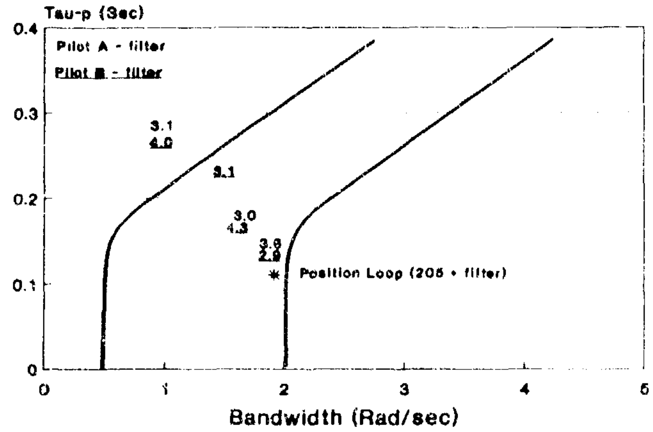

Figure 11: Average CHR, All Tasks, With Filter

Isometric Stick. As can be seen clearly in Figures (8) and (9) the Cooper Harper ratings for the isometric cases follow closely the anticipated values predicted by the bandwidth criterion. This confirms previous studies which have shown that delays downstream of the stick and for which the pilot can not compensate sub-consciously will affect the handling qualities of the vehicle.

Lateral Regulation Task. Figures (12) and (13) show the CHR and tracking performance results from the version of this task in which the aircraft was driven by the derivative of the attitude disturbance function, while Figures (14) and (15) refer to the version in which a model following control system was used and the pilot was under simulated IMC. For ease of comparison with other studies, these plots have Equivalent Time Delay (ETD), defined as ($\tau_e = 2\zeta/\omega_n$) on the horizontal axis. It should be noted that this is specifically the Equivalent Time Delay of the stick, not the entire aircraft.

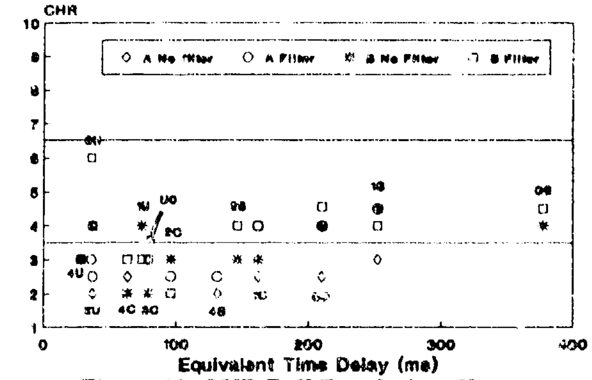

Figure 12: CHR Roll Regulation, Hover

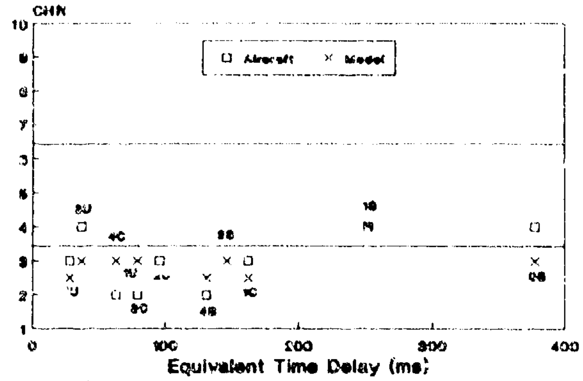

Figure 14: CHR, Roll Regulation, IMC

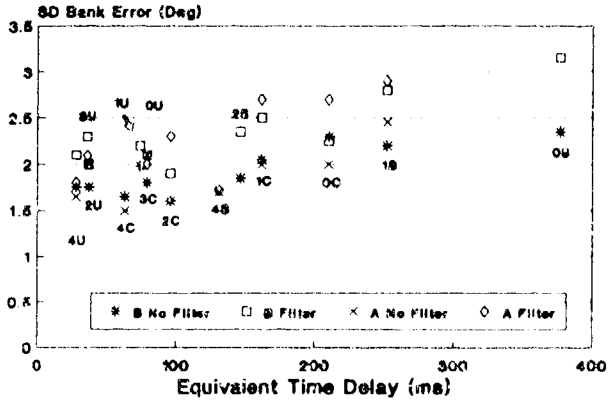

Figure 13: SD Bank Angle Error, Hover

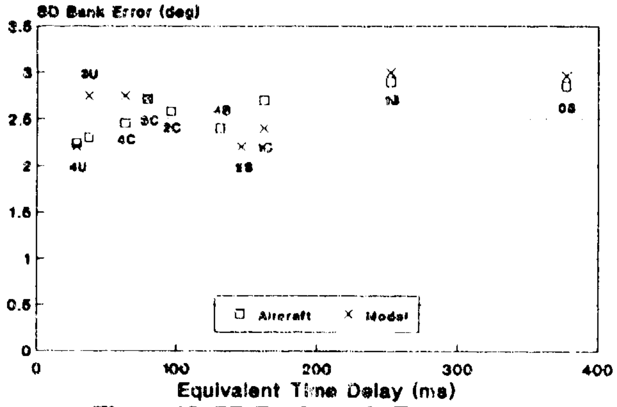

Figure 15: SD Bank Angle Error, IMC

DISCUSSION OF RESULTS

Introduction

One of the difficulties in trying to determine the effect of stick characteristics on the handling qualities of an aircraft is that one can not be certain as to which dominates, the stick or the quality or type of the vehicle's responses. It has been clearly shown (References [9],[2]) that changing the bandwidth or phase delay of the command response or changing the response type can have a dramatic effect on the handling qualities evaluations of the vehicle. Were this not so, the ADS-33C bandwidth criteria would not exist. Accepting this, the results of this study can only be taken to apply to an aircraft with a high quality RATE COMMAND control system. Not only is it uncertain, but from empirical knowledge unlikely, that the same results would obtain for, say, an ATTITUDE COMMAND system. The general evidence from this experiment is that the aircraft's characteristics dominate those of the stick, the lesson here is that research of this nature should only be undertaken if a plant of adequate quality is available.

General Hover Manoeuvring

Displacement Sticks. The CHR assigned for these tasks reveals two possible boundary conditions for centre mounted cyclic configurations. Both at high and low frequencies there is an indication that underdamped sticks should be avoided, albeit for different reasons. One of the evaluators appeared to find a boundary based on low undamped natural frequency.

The high frequency sticks with low damping ratios are susceptible to excessive bio-inertial feedback and possibly neuro-muscular resonance as described in Reference [6], while at low frequency (from pilot comment rather than CHR) the sensation of a bob-weighted stick is present and should be avoided. All underdamped second order systems will, of course, 'ring' after an abrupt input but at frequencies of 9 rad/sec or higher this ringing appears to be undetected by the pilots, while they were definitely aware of it with the 5.4 rad/sec stick. Although the coupling with the Bell 205 "Mast Rocking" oscillation noted during this experiment is aircraft type specific, it is likely that all helicopters will have excitable oscillatory modes at frequencies low enough to be triggered by pilot activity, voluntary or otherwise.

Pilot B, the naive subject also considered the lowest frequency stick to be, at best, marginally Level 1 and his opinion should probably dominate. One of the difficulties associated with handling qualities research using very experienced pilots as subjects is that they have probably already adapted, as a matter of past necessity, to quite unsatisfactory systems. That Pilot A in this study was not aware of any problem in having to move the cyclic stick through large deflections in routine tasks, while his less experienced colleague found this requirement quite irksome and commented frequently on it, is undoubtedly a manifestation of such adaptation.

Isometric Sticks. Comparing data from the displacement and isometric sticks, it becomes clear that the pilots were using the motion cues in the displacement controllers to very good, indeed dramatic, effect. One of the reasons for selecting such a high stick gradient as 9 lb/in was in an attempt to determine at what point the motion cues failed to provide the pilot with useful feed-back. In this the experiment failed, since the two Cases with high spring gradient, Cases 3 and 4 remained solidly Level 1, unless underdamped, even in the presence of a stick filter. It will be necessary to extend the range of spring gradients in future experiments to define such a limit.

Roll Regulation Task. While reading the literature of roll tracking or roll regulation experiments under simulated IMC, one finds two main techniques used in tasking the pilot. Either a display of 'attitude' is disturbed and the pilot is required to restore it to trim, or a symbol of some kind is perturbed and the pilot has to control attitude to restore it to a null. These can both be compensatory tracking tasks (depending on symbol drive) and in a fixed base simulator are probably identical as far as pilot behaviour is concerned. In the air, however, they are far from identical. If the aircraft is disturbed, the pilot receives vestibular and inertial cues two integrations in advance of the attitude disturbance, while if a display symbol is driven he has only his visual cues on which to rely. In compensating for aircraft external disturbances pilots typically react to the inertial cues with open loop pulse inputs, proportional in magnitude to the severity of the accelerations detected. Thus a corrective input is made before the aircraft has moved sufficiently to produce a significant attitude change. In the helicopter pilot in particular this is a highly trained reaction for without it maintaining a precise hover in turbulent conditions would be impossible. The implication of this is that great caution should be exercised in comparing data taken from fixed base simulation or 'null the symbol' flight experiments, with data from experiments of the type conducted here.

Roll Regulation at the Hover. The HQR data achieved in this task indicate no significant change from those assigned in the general hover manoeuvring phase for tasks with the same general frequency content. They are, however, somewhat at variance with the data from those tasks containing large discrete manoeuvres and did not discover those shortcomings in the stick models revealed by such tasks. In this task, the majority of cases of the unfiltered sticks are considered to be Level 1 by both evaluators, while again Pilot B shows his dislike for the very low frequency stick. That the filtered cases fall just over the Level 1 HQR boundary is again considered to be due to the command response having repositioned to the Level 1 boundary on the bandwidth criterion. This is strongly supported by the fact that the filtered cases drop quite early to Level 2 but are then assessed as an almost constant CHR 4 despite the equivalent time delay increasing from about 130 ms to 377 ms. This must suggest that the pilot is not aware of the increasing time delay in the stick and is not consciously compensating for it. Pilot performance, as measured by the standard deviation of bank angle error, degrades with increasing stick ETD from about 1.6 deg to 2.4 deg for the unfiltered sticks and from about 2.0 deg to 3.1 with the in-line filter. It is possibly significant that in all cases for which data were taken, the roll errors are less with the critically damped sticks than with the underdamped versions, another argument against using underdamped controllers.

Roll Regulation Under Simulated IMC. The HQR assigned in this task are remarkably similar to those achieved during the same task at the hover, despite a different task implementation and although the pilot was now limited to an instrument display for aircraft control. The majority of cases are still considered to be Level 1 but there are insufficient data to determine whether presenting the pilot with the actual aircraft state or the model state is of importance or not. Pilot performance in the IMC task is slightly poorer at the low ETD than when at the hover, but is much the same at high ETD. This is probably a function of the poorer cues available to him under IMC and that disturbance recognition was more significant when the stick lags were low.

General Comparison. A general comparison between both the CHR and pilot performance data acquired in this study and those published in Reference [7] indicates that more benign handling qualities and slightly better performances were achieved in the Airborne Simulator than in the CH-47. Taking the internal evidence of this study and the results of previous research from References [9] and [2] the reason for this is probably associated with the positioning of the basic plant on the bandwidth criterion map, as shown in Figure (16), rather than for any other reason. This being accepted, the results of the two studies are not altogether different, the trends being in much the same direction but somewhat displaced from one another in the expected direction.

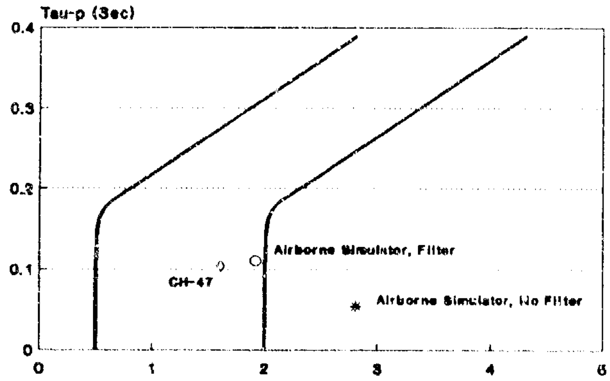

Figure 16: Basic Position Loops, 205 and CH-47

CONCLUSIONS

Although this was only an exploratory study, and although there is considerable in-depth analysis still to be done (in particular it is intended to subject all pilot control inputs to frequency analysis in an attempt to determine how the pilot compensates for differing stick dynamics), there is sufficient evidence to draw several major conclusions:

a. Under damped sticks (damping ratio of the order of 0.3) should be avoided, as should sticks with an undamped natural frequency less than about 9 rad/sec. A tentative boundary based on these premises is shown in Figure (17).

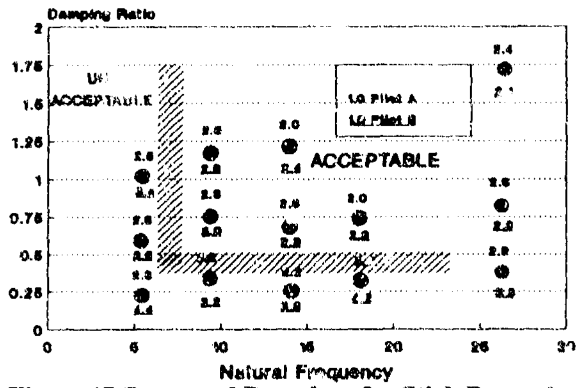

Figure 17:Suggested Boundary for Stick Dynamics

b. Spring gradients of at least 9.0 lb/in are thoroughly acceptable provided the maximum displacement does not required an unreasonable force.

c. The quality of the plant has a profound effect on the handling qualities of the vehicle and the variance in HQR assigned to an aircraft with responses close to the Level 1 boundary may be sufficient to mask changes due to the characteristics of the stick itself. The inner loop responses of the aircraft should be solidly Level 1 to permit meaningful research in this area.

FUTURE STUDIES

Future studies in this area are required to determine if the suggested low frequency boundary really exists, if there is an absolute limit on acceptable spring gradient and precisely where the low damping boundary should be drawn. As a further extension, the effect of changing the aircraft's response type should be investigated. It is the intention of the FRL to continue this line of research after first converting part of the stick simulation to digital computation to permit simpler and more accurate model realisation.

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APPENDIX A

GENERAL HOVER TASK DETAILS

THE TASK DESCRIPTIONS GIVEN HERE ARE REPRINTED FROM THE EVALUATION PILOT BRIEF USED IN THIS STUDY

PRECISION HOVER

Pick a traffic cone for reference and come to a five foot hover with it in view. Hover for 45 seconds attempting to maintain:

Height +/- 1 foot
Position +/- 1 foot X and Y

PRECISION LANDING

Using the same cone for reference, make a precision vertical landing adjacent to it. Aim for the same X/Y limits as for the hover and attempt to make a smooth, continuous descent to touchdown.

LATERAL UNMASK/MASK (SIDE-STEP)

Using the markers as demonstrated:

Establish 10 foot hover and hold for 10 seconds. Perform rapid lateral translation to second marker, establish hover and maintain for 5 seconds, rapid lateral translation back to start point and re-establish hover. Aim to achieve:

Height +/- 3 feet
Accelerate and decelerate without bank angle oscillation
Heading +/- 5 degrees
Fore/aft motion +/- half fuselage

ACCELERATE/STOP

Establish a 10 foot hover, accelerate to reach 35 kt groundspeed at the gate (45 kt if alternate course in use), and decelerate to end at a hover inside the Pirouette circle. Aim to achieve:

Height +/- 4 feet

No undesirable or uncommanded motions during recovery to hover.

PIROUETTE

At the marked circle, establish a 10 foot into wind hover. Commence yawing lateral translation to maintain fuselage over markers and nose pointing at centre marker. Re-establish hover at start point. Aim to maintain:

Height +/- 4 feet
Constant lateral velocity/ yaw rate

Re-establish hover without bank angle oscillations
Complete circle in 60 seconds

PDEAL TURN WITH HESITATIONS

From a 10 foot hover, perform a brisk 360 degree pedal turn, pausing every 90 degrees for 3 seconds. Aim for:

Height +/- 2.5 feet
Heading pauses +/- 5 degrees

OFF LEVEL LANDING

Inside the marked box, make normal approach to slope landing, when the uphill skid is in contact with the ground, maintain the constrained hover for 20 seconds before lowering the downhill skid. Aim for a steady and smooth lowering of the downhill skid with no yawing excursions once in contact.

During take-off, again maintain a constrained hover for 20 seconds before breaking clear of the ground.

APPENDIX B

DETAILED COOPER-HARPER RATINGS

CHR Pilot A - GENERAL HOVER MANOEUVRING

Case	Hover	Ldg	Side Step	Accel Stop	Pirou-ette	Hes Turn	Slope Ldg	Mean
0C	2	3	2	2.5	3	3	2.5	2.57
0U	2	2.5	2.5	2	2.5	2.5	2	2.29
0S	3	3	2.5	3	3	4	3	3.07
0S*	2	3	2.5	3	2.5	2	2.5	2.50
1C	2.5	3.0	2	2	2	2	2.5	2.29
1U	3	2.5	3	3	3	2.5	2.5	2.79
1S	3	3	2	2	2	2.5	3	2.50
1S*	4	4	4	4	3	2	3	3.43
2C	3	2.5	3	3	2	2.5	2.5	2.64
2U	2.5	3	3	3	2	2		2.21
2S	2	2	2	2	2	2	2	2.00
3C	2	2	2	2	2	2	2	2.00
3U	4	4	7	7	6	7	5	5.71
3C*	3	3	2.5	2.5	3	2.5	3	2.79
4C	2	2	3	2	3	3	3	2.57
4U	1.5	2	3	4	4	4	2	2.93
4S	2	2	2.5	2.5	2.5	2	3	2.36
4S*	3	2.5	2.5	1.5	3	2.5	2.5	2.50
0SF	3	3	4	4	2.5	2.5	3	3.14
2CF	2.5	2.5	4	4	2.5	2.5	3	3.00
3CF	3	4	4	3	4	3	4	3.57
S02	3	3	2	2	2.5	2.5	2.5	2.50
S12	2	2	2	2	2	2	2	2.00
S22	4	4.5	4	2.5	4	5	4.5	4.07
S42	6	7	6	2.5	6	6	7	5.79

CHR ROLL REGATION TASK

Case	CHR	Case	CHR	Case	CHR
0C	2.5	0U		0S	
1C	2.5	1U		1S	3
2C		2U	2	2S	
3C	3	3U			
4C	2.5	4U		4S	2
0CF	4	0UF		0SF	
1CF	4	1UF		1SF	4.5
2CF	2.5	2UF		2SF	
3CF	2.5	3UF	3		
4CF		4UF		4SF	2.5
S0	5	S1		S2	3
S4	5				

APPENDIX B

DETAILED COOPER-HARPER RATINGS

CHR Pilot B - GENERAL HOVER MANOEUVRING

Case	Hover	Ldg	Side Step	Accel Stop	Pirou-ette	Hes Turn	Slope Ldg	Mean
0C	3	3	4	4	4	3	4	3.57
0U	5	4	4.5	4	4	4	5	4.36
0S	4	4	5	4	4	4	4.5	4.21
1C	3	2.5	4	3	3	3	2.5	3.00
1U	3	2.5	4.5	3	3	2.5	4	3.21
1S	3	3	3	3	2.5	2.5	3	2.86
2C	2.5	2	3	2.5	2	2	2	2.29
2U	3	3	4	4	4	3	4	3.57
2S	2	2	3	2.5	2.5	2.5	2	2.36
3C	2.5	2	2	2.5	3	2	2	2.29
3U	3	2.5	2	2	8	8	4	4.21
4C	2.5	2.5	2	3	2.5	2	4	2.64
4U	2	2	2	5	7	2	3	3.29
4S	2	2	2.5	2.5	2.5	2	3	2.36
0SF	4	4	4	4	4.5	3	4.5	4.00
2CF	4	3	4	3	4	3	4	3.57
3CF	4	4	5	4	4	4	5	4.29
S02	3	3	4	4.5	3	2.5	4	3.43
S12	3	3	4	3	3	2.5	4	3.21
S22	4	4	4	4	4	3	4	3.86
S42	4.5	4	5	4	4	4	6	4.50

CHR ROLL REGATION TASK

Case	CHR	Case	CHR	Case	CHR
0C	4	0U		0S	4
1C	3	1U	4	1S	4.5
2C	3	2U	4	2S	3
3C	2	3U			
4C	2	4U	3	4S	3
0CF	4.5	0UF		0SF	4.5
1CF	4	1UF	3	1SF	4
2CF	2	2UF	4.0	2SF	4
3CF	3	3UF	6		
4CF	3	4UF		4SF	3
S0	7	S1	4	S2	3
S4	5				

AN INVESTIGATION INTO THE USE OF SIDE-ARM CONTROL FOR CIVIL ROTORCRAFT APPLICATIONS

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ABSTRACT

An evaluation of the handling qualities of civil rotorcraft incorporating force or displacement sensing side-arm controllers with varying levels of control integration was carried out on the NAE Bell 205 Airborne Simulator. Evaluators were certification pilots from the FAA and Transport Canada. The results indicate that integrated 4-axis side-arm control is a viable option for civil rotorcraft operations, even when used in conjunction with very low levels of stability and control augmentation.

Introduction

The advent of fly-by-wire technology and its adaptability to integrated multi-axis side-arm control will have far-reaching effects on the design and operational utility of rotorcraft. Some of these effects are highly visible such as on physical constraints in cockpit design, pilot view and comfort and crashworthiness. Other effects, such as those on the handling qualities, in terms of pilot workload and performance, can only be defined by acquiring in-flight data.

Background

The application of multi-axis side-arm control for rotorcraft operations has been investigated by the Flight Research Laboratory (FRL) of the National Aeronautical Establishment (NAE) since 1979 (Ref. 1 to 5). These past activities have been aimed primarily at military rotorcraft operations addressing, in large part, military rotorcraft handling qualities specifications. Although certain phases of military operations resemble civil use of rotorcraft, requirement specifications and certification procedures differ.

The Flight Research Laboratory has been performing research on civil helicopter handling qualities in cooperation with the U.S. Federal Aviation Administration (FAA) under MOA AIA-CA-31. This report deals with one of the latest experiments performed under this agreement.

Scope of the Program

This experiment was designed to address the following issues:

a) Is multi-axis integrated side-arm control a viable option for civil rotorcraft operations?

b) How is pilot workload and performance affected by the use of this mode of control versus the use of conventional controls while performing tasks representative of civil operations.

c) Are there any special civil certification issues which must be addressed for deflection-sensing and force-sensing integrated side-arm controls?

THE AIRBORNE SIMULATOR

Experiments were carried out using the NAE Airborne Simulator, an extensively modified Bell 205A-1 with special fly-by-wire capabilities that have evolved over the last seventeen years (Figure 1). The standard hydraulically boosted mechanical control actuators incorporate servo-valves that can be positioned either mechanically from the left (safety pilot) seat or electrically from the right (evaluation pilot) seat full authority fly-by-wire station. Fly-by-wire inputs are generated by a set of motion sensors and a computing system consisting of two LSI 11/73 and one Falcon microprocessor and D/A and A/D converters. Inputs to this system come from electrical controllers which may be either a conventional stick, pedals and collective combination with a programmable force-feel system or, alternatively, a 4-axis isometric force or deflection side-arm controllers or any viable combination of these systems.

Other modifications to the NAE Airborne Simulator have been made to increase the *simulation envelope* of the facility. To quicken the control response of the teetering rotor system, the standard Bell 205 stabilizer bar was removed; and to provide an additional pitch axis control, the longitudinal cyclic-to-elevator link was replaced with an electro-hydraulic actuator, although, for this program, the elevator remained fixed in the neutral position. Reference 6 provides a full description of the NAE Airborne Simulator.

Aircraft Configuration

The use of a side-arm controller in a rotorcraft implies that some level of *fly-by-wire* technology is present in the aircraft, if only to allow the electrical signals of the controller to be passed to the control system. On the other hand, any rotorcraft with a side-arm controller could also be highly advanced to the point of almost totally automated flight. While both extremes raise interesting research and certification issues, it was decided early in the experiment development process that the rotorcraft dynamics to be used in the evaluation should be representative of the most probable configuration which would first appear on the civil market.

Although it is not the only successful civil rotorcraft on the market, the Sikorsky S-76 is representative of most rotorcraft currently in production and clearly is a standard in terms of stability augmentation and IFR capability. With this in mind, the decision was made to configure the NAE Bell 205 Airborne Simulator to possess dynamic characteristics which were similar to the S-76 with stability augmentation system (SAS) engaged. Unlike the standard S-76 SAS, which decreases with speed and reverts to a constant level of damping at speeds below 40 knots, the airborne simulator rate damping matched the S-76 levels at high speeds but continued in a linear reduction all the way to the hover. Interaxis control coupling between all axes were reduced to a very low level by the use of simple control cross feeds to the respective control axes. This characteristic is also similar to a fully augmented S-76. The hover rate damping derivatives of the Airborne Simulator, as used in this experiment, were 3.0 and 4.2 sec^{-1} for roll and pitch axes respectively. Bode plots of the aircraft control response in terms of attitude per unit of control input are included as Figures 2 and 3. These *units* of control input are directly related to the controller sensitivity values given in Table 1. The implementation of control filtering and integral trim on each of the controllers is documented in Figure 4 while Figure 5 shows the pitch and roll control system architecture.

The yaw axis of the Bell 205 was configured as a rate command / heading hold system which blended to a sideslip command / turn coordination system at 35 knots (Figure 6). The vertical axis was a standard collective system with the sensitivity and heave damping of a standard Bell 205.

Controllers

For this experiment, two side-arm control configurations were flown and compared with conventional controls comprising a cyclic stick, tail rotor pedals and collective lever.

The side-arm control configurations were:

a) a 4-axis force controller with compliance in pitch and roll axes (Figure 7)
b) a 4-axis deflection controller (Figure 8).

Table 1 summarizes the characteristics of the three control configurations. **It must be noted that the 4-axis displacement controller evaluated in this experiment possessed physical breakout/gradient characteristics which were not optimized.** The same controller was evaluated in a prior experiment (Ref. 3) with nearly optimum characteristics which are also described in Table 1. In addition to the three major systems, various *integration levels* of side-arm control were also examined for each side-arm controller. These integration levels, as shown in Figure 9, were 4 + 0 (fully integrated), 3 + 1c (collective separate), 3 + 1p (pedals separate) and 2 + 1 + 1 (fully distributed).

EXPERIMENTAL PROCEDURE

Evaluation pilots typically assessed either one or two controller configurations on a given flight. To ensure that each evaluator was consistent in his performance of the evaluation tasks, the safety pilot demonstrated all tasks using the conventional controls at his station on the first flight. From that point on, evaluators assigned handling qualities ratings (HQRs) using the Cooper-Harper handling qualities rating scale (Ref. 7), and filled out a questionnaire (Figure 10) for each control configuration as it was encountered. Post flight debriefings gave the project engineers the opportunity to clarify the written comments of the pilot and to discuss, in more depth, the pilot's reasoning behind his assessments. Table 2 gives the sequence of evaluations for each evaluator. This order was designed to determine whether the sequence of evaluations (force or displacement first) would alter pilot assessments.

A total of 47.1 flight hours were flown by four evaluators (12 hours each). On completion of all evaluations, each evaluator filled out a general questionnaire (Figure 11).

Tasks

The evaluators were required to perform the tasks shown pictorially in Figure 12 two or three times for each configuration and to provide evaluations for the following tasks:

❑ *Precision Hover*

The evaluator was asked to maintain a precision hover with respect to a traffic cone viewed through side window markings (longitudinal and lateral position approximately ± 3 feet). Height was to be maintained at 5 ± 2 feet and heading to ± 5 degrees of nominal.

❑ *Precision Landing*

A landing was performed with the view of the traffic cone maintained in the side window markings (position accuracy approximately ± 1.5 feet). Vertical descent rate was required to be continuous to touchdown with no perceptible longitudinal or lateral drift.

❑ *Sidestep*

A sideward hover-taxi manoeuvre was required across a circle of 200 feet in diameter. Height was to be maintained at 10 ± 3 feet, heading at ± 10 degrees from nominal, and the manoeuvre was to be completed in 15 seconds or less.

❑ *Hover with Divided Attention*

The evaluator was required to change radio frequency while maintaining a hover position of ± 10 feet horizontally and a height between 2 feet and 15 feet above ground.

❑ *Pirouette*

The aircraft was manoeuvred around a marked circle of 200 feet in diameter with the nose pointed towards the centre of the circle at all times. Tracking tolerancies were ± 10 feet from the circle circumference with height maintained at 10 feet ± 5 feet and heading was to be controlled within ± 10 degrees of the circle center-point. Lateral velocity was to be controlled smoothly, allowing completion of one circuit in a maximum of 45 seconds.

❑ *Figure Eight*

The evaluator was asked to track, in forward flight, a marked figure eight pattern composed of two, 200 foot diameter circles. Height was to be maintained at 10 ± 5 feet, allowable lateral tracking tolerances were ± 10 feet from the marked track and the manoeuvre was to be completed in less than 50 seconds.

❑ *Quick Stop*

From a hover position, the aircraft was accelerated to 35 knots groundspeed and then rapidly decelerated to a stop in a total distance of approximately 600 feet as referenced by ground markers. Heading was to be maintained at ± 10 degrees and the maximum allowable height was 25 feet.

❑ *Slope Landing*

A landing on a four-degree slope was performed with aircraft heading perpendicular to the slope. The manoeuvre was to be performed with precise control of the downslope skid and with no perceptible drift on touchdown.

❑ *Obstacle Clearance Takeoff and Steep Approach*

From a hover, with maximum engine power, an obstacle clearance takeoff was performed into a tight circuit with a steep approach to a hover.

❑ *Entry into Autorotation*

While in cruise, with the evaluator pilot in control, the safety pilot reduced the throttle to idle to simulate a rapid engine failure. The evaluator then selected a suitable field for landing and performed left and right 90 degree turns while controlling airspeed and rotor speed to within the Bell 205 specified limits. Because throttle control was not available to the evaluator, the safety pilot took control for the recovery. Laboratory policy does not allow practice in full-on autorotation landings in the airborne simulator.

❑ *Instrument Approach*

The evaluators were provided with a precision tracking task in the form of azimuth, elevation and airspeed information representing an MLS approach at a 6 degree elevation angle. A flight director display was used to track the localizer and the glideslope at 60 knots, and then decelerate on a profile based on distance from a simulated touchdown point (approximately 1.3 ft/sec^2) to 20 knots. (See Reference 8 for a more complete description of the basic approach and flight director system).

Evaluators

Four experienced helicopter certification test pilots performed the evaluations, three from the FAA and one from Transport Canada. A summary of their relevant experience is tabulated in Table 3.

ENVIRONMENTAL CONDITIONS

Relevant atmospheric conditions during the program varied from calm winds in smooth conditions to winds gusting from 15 to 20 knots with moderate turbulence. The last *fly off* sequence of three configurations – conventional, force (4 + 0) and deflection (4 + 0), were flown in rapid succession to ensure common wind conditions for each pilot's evaluation.

PILOT RATINGS

The results of the pilot ratings for each manoeuvre are plotted in Figures 13 to 23.

Hover

Pilots were able to perform this task to acceptable accuracy with all controller configurations. Figure 13 indicates that pilots preferred all of the force sensing controller configurations, except the (3 + 1)p configuration, even over the conventional configuration.

Reducing the level of integration of the force controller offered no apparent advantages. The deflection controller configurations were the least acceptable ones for this task, with some improvement in handling qualities available by reducing the integration level to the fully distributed case (2 + 1 + 1).

Landing

Figure 14 indicates that three configurations of the force controller were preferred in this manoeuvre with the deflection controller configurations least preferred. With all configurations, this task was performed to satisfactory performance levels. Reducing the integration level of either of the hand controllers did not provide significant workload relief.

Sidestep

In this manoeuvre (Figure 15), conventional controls and the force controller configurations were preferred, with very slight preference given to the reduced integration level configurations of the force controller.

Divided Attention Hover

Figure 16 indicates a marked preference for the force controller configurations. The deflection controller configurations were rated at least as good as the conventional controls.

Pirouette

The fully integrated force controller was preferred for this task (Figure 17), even over configurations where the integration level was reduced with this controller. On the other hand, with the deflection controller, although rated poorest, some benefit is apparent in reducing the integration level.

Figure Eight

Figure 18 indicates that conventional controls and the fully integrated force controller were rated best for this manoeuvre. Reducing the level of integration on the force controller appeared to degrade the handling qualities slightly. Again, the deflection controller was rated the poorest with some benefit provided when integration level was reduced.

Quick Stop

This manoeuvre was the only one in which the conventional controls were preferred over all other configurations (Figure 19) However, the force controller was rated only slightly poorer with no apparent benefits provided by reducing integration level. The deflection controller was rated much poorer (bordering on unacceptable) but significant improvements were apparent when the integration level was reduced.

Slope Landing

In this task (Figure 20), the force controller configurations were preferred again with no benefit provided by reduced integration level. The handling qualities with the deflection controller were significantly degraded with obvious improvements when the integration level was reduced. The 2 + 1 + 1 configuration with this controller was rated the same as with conventional controls.

Obstacle Clearance Takeoff and Steep Approach

The force controller with the lowest level of integration was rated best for this task (Figure 21). However, conventional controls and the force controller with higher levels of integration were rated only slightly poorer. With the deflection controller, marked improvements were apparent at reduced integration levels , to the point that the 2 + 1 + 1 configuration was almost as good as with the force controller.

IFR Decelerating Approach

Results of pilot ratings for the IFR tracking task are shown in Figure 22. Two evaluators judged the force controller to be better than conventional controls – one rated both the same and one rated the force controller one rating poorer, but felt that the force controller reduced pilot workload and was optimized with the flight director control laws. The deflection controller was rated poorest by all evaluators, primarily due to poor breakout/gradient force characteristics.

Autorotation Entry

Fully integrated side-arm controllers were rated poorest for autorotation (Figure 23). The dominant complaint was a lack of collective position feedback cue on initial collective application. Thereafter, the force controller characteristics were adequate in providing reasonable control of rotor rpm, a factor lacking in the deflection controller because of poor breakout/gradient force characteristics.

Learning Trends

In order to highlight learning trends, pilot ratings of the first and last exposure to a particular configuration are shown in Figure 24 for the conventional controls, fully integrated force control and fully integrated displacement control configurations. The reader is reminded (Table 2) that two pilots experienced all integration levels of the force-sensing controller before being introduced to the deflection-sensing controller. The reverse is true for the other two pilots. No noticeable differences in final assessments could be attributed to these different evaluation sequences. Also, these investigations were not necessarily performed in the same atmospheric conditions for each evaluator. The data in Figure 24 shows that the displacement controller configurations displayed the largest learning curve effect with a typical 1 HQR improvement for most tasks over the training length of the experiment. The ratings for the quickstop, however, show no improvement for this controller, suggesting either that much more training was necessary or that the characteristics of the controller combined with that task were especially unsuitable. (The latter was confirmed by pilot comments).

The 4 + 0 force controller learning curves are in general shallow and similar to the conventional controller trends. This similarity, and pilot comments regarding learning curve effects, suggests that pilots adapted to the 4 + 0 force controller was easily adapted to for most tasks.

The data in Figure 24 for pirouette and figure 8 tasks should be highlighted. These two tasks involve considerable multi-axis control which has been cited as a possible limitation for sidearm controllers. The fact that both controllers demonstrated steep learning curves for exposures on the order of a few hours and that the force controller final ratings were as good as conventional controls, dispells this reservation regarding side-arm controllers. It also points out that adquate training is necessary for proper evaluation of these devices.

PILOT COMMENTS

In general, all evaluators felt that the basic aircraft characteristics represented typical helicopter handling qualities. However, most evaluators suggested the fixed horizontal stabilizer resulted in extreme pitch attitudes when the aircraft tail was turned into wind.

Conventional Controls

Pilots cited some deficiencies in the conventional control configuration. Two of the evaluators had difficulty in yaw axis control and stabilization. It is felt that this difficulty stemmed from two factors, non-optimum pedal force characteristics coupled with a yaw axis system which had dynamics significantly different from a conventional unaugmented helicopter yaw axis. This interaction caused the two evaluators to have problems obtaining smooth and consistent control of the yaw axis. A typical comment was "jerky" or "steppy" in yaw. While the other two evaluators did not highlight this deficiency, possibly because they adapted to the system more quickly, these pilots did miss the lack of a force trim release system on the conventional cyclic and disliked the higher than "normal" cyclic stick forces that they experienced. Despite these deficiencies, all four evaluators rated the configuration as certifiable and, as indicated above, *typical*.

Force Sensing Side-Arm Controller

Evaluators were impressed, even on first exposure to this control system, with the ease at which they could perform stabilization tasks with this controller. The integral trim system allowed precise modulation of the aircraft controls and alleviated the requirement for the pilot to continually concern himself with aircraft trim, even in rapidly changing wind/heading conditions. The learning curve was assessed as steep for all configurations using this controller and, with the exception of three evaluations of marginal certifiability due to yaw axis/wind difficulties in the pirouette manoeuvre, all configurations incorporating the force sensing controller were assessed as certifiable. Deficiencies cited for the force sensing side-arm controller were as follows:

1) In some manoeuvring tasks, the quickstop, the obstacle clearance takeoff and steep approach and the autorotation – evaluators would appreciate better control position feedback cues, especially in the collective axis.

2) Some evaluators initially complained of inter-axis control coupling on early exposures to the controller; however, these complaints were not received during later evaluations, suggesting that this could be a learning curve related effect.

3) For all levels of controller integration, comments regarding the slope landing task, which was rated as a marginal Level 1 handling qualities manoeuvre, highlighted the need for better indications of the rotor tip-path-plane. Improved control position indicators could possibly meet this need.

Deflection-Sensing Side-Arm Controller

This control configuration was rated the poorest of all configurations for all the tasks. No significant benefits were perceived from the small controller deflections that provided a level of control position feedback to the evaluator, or perhaps any such benefits were masked by other deficiencies. The dominant deficiency appeared to be poor breakout/gradient force characteristics of the controller. It is worthy of note that this same controller was rated much better in previous work at the NAE (Ref. 3) where cyclic pitch breakout force was 26% less and pitch gradient 77% greater, and where lateral cyclic breakout force was 24% less and lateral gradient was 126% greater. With the poor breakout/gradient characteristics, reducing the level of controller integration (number of axes) on the controller resulted in significant benefits in improved workload. This effect was not as noticeable, however, on the force sensing controller which had nearly optimum force characteristics.

In addition to the poor physical characteristics of the displacement controller, which were cited by all four evaluators, any deficiencies described for the force sensing controller were usually repeated for this controller as well.

Reduced Integration Levels

Pilot comments directly related to the integration level of the side-arm controller displayed a number of tendencies:

1) As described above, for a controller with poor physical characteristics, any reduction in integration level improved the vehicle handling qualities.

2) The (3 + 1)c configuration provided only a slight improvement in vehicle handling qualities, even at the earliest stages of the pilot learning curve on side-arm controllers.

3) At least two of the evaluators consistently preferred yaw axis control on the side-arm controller rather than the (3 + 1)p configuration. Generally, if a single axis split is required, the consensus was that collective should be the separated control.

CONCLUSION

The following conclusions can be drawn from this experiment:

a) The use of integrated 4-axis side-arm control is a viable option for civil rotorcraft operations, even when used with very low levels of stability and control augmentation such as represented in this experiment.

b) Pilot workload level and performance for configurations with the force sensing 4 + 0 controller was as good or better than with conventional controls for most tasks and, with the provision of improved control position information to the pilot, this type of control has the potential for further improvement in handling qualities.

c) The breakout/gradient force characteristics and sensitivities of side-arm controllers may dominate aircraft handling qualities. A systematic evaluation of a range of these characteristics for all representative tasks is required to establish satisfactory boundaries for both force-sensing and deflection-sensing controllers. This would provide much needed guidance to manufacturers of such systems.

d) A number of certification issues were suggested by the evaluators. Most of these would be addressed in the incorporation of fly-by-wire technology such as:

- fault/failure analysis to ensure redundancy
- provision for monitoring coupled systems
- testing for electro-magnetic interference

Some issues directly relevant to integrated side-arm control are:

- definition of acceptable characteristics as in c) above
- definition of acceptable aircraft dynamic stability in relation to integrated side-arm control
- establishing pilot/co-pilot control priority in dual pilot operations
- the enhancement of control position or tip path plane cues to the pilot

Overall, the force sensing 4 + 0 controller was preferred for most manoeuvres over the conventional control configuration

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TABLE 1: Controller Charcteristics

	Breakout	Gradient	Travel (+/-)	Sensitivity	Integral
Conventional					
Pitch	0.5	1.0	6.0	.46	0
Roll	0.25	1.0	6.5	.31	0
Yaw	7.0 (lb)	15.0 (lb/in)	4.5	.53 (unit/in)	0
Collective	adjustable friction	0.0	5.35 (in)	.29 (unit/in)	0
Force Side-Arm					
Pitch	0.3	15	0.5	.27	.125
Roll	0.3	15	0.5	.27 (units/lb)	.125
Yaw	0.75	∞	0.0	.09 (unit/in-lb)	0
Collective	0.075 (lb)	∞	0.0 (in)	.03 (unit/lb)	1.90
Displacement Side-Arm					
Pitch	2.3	0.9	15°	.26	0.5
Roll	1.3	0.10	17°	.12	0.05
Yaw	1.9 (in-lb)	0.17 (in-lb/deg)	12°	.22 (unit/deg)	0.0
Collective	0.7 (lb)	2.2 (lb/in)	.5 (in)	see note below	0.50
Displacement Side-Arm (Ref 3)					
Pitch	1.7	.16			
Roll	0.95	.23			
Yaw	1.9 (in-lb)	.13 (in-lb/deg)			

Note: The displacement side-arm controller incorporated a non-linear sensitivity in the vertical axis where units = .4 x^3 + 1x and x = 2 * controller displacement (in).

TABLE 2: Controller Configuration Sequences

- 4 + 0 = 4-axis side-arm
- (3 + 1)c = 3-axis side-arm, conventional collective
- (3 + 1)p = 3-axis side-arm conventional pedals
- (2 + 1 + 1) = pitch roll side-arm, conventional pedals & collective

EVALUATOR

A		B		C		D	
Conventional		Conventional		Conventional		Conventional	
Force	(4 + 0)	Deflection	(4 + 0)	Deflection	(4 + 0)	Force	(4 + 0)
"	(3 + 1)c	"	(4 + 0)	"	(4 + 0)	"	(4 + 0)
"	(3 + 1)p	"	(3 + 1)c	"	(3 + 1)c	"	(3 + 1)c
"	(2 + 1 + 1)	"	(3 + 1)p	"	(3 + 1)p	"	(3 + 1)p
Deflection	(4 + 0)	"	(2 + 1 + 1)	"	(2 + 1 + 1)	"	(2 + 1 + 1)
"	(4 + 0)	Force	(4 + 0)	Force	(4 + 0)	Deflection	(4 + 0)
"	(2 + 1 + 1)	"	(3 + 1)p	"	(3 + 1)c	"	(3 + 1)c
"	(3 + 1)p	"	(3 + 1)c	"	(3 + 1)p	"	(3 + 1)p
"	(3 + 1)c	"	(2 + 1 + 1)	"	(2 + 1 + 1)	"	(2 + 1 + 1)
"	(4 + 0)	"	(4 + 0)	"	(4 + 0)	"	(4 + 0)
Force	(4 + 0)	Deflection	(4 + 0)	Conventional		"	(4 + 0)
Conventional		Conventional		Deflection	(4 + 0)	Force	(4 + 0)
						Conventional	

TABLE 3: Evaluator Relevant Flying Experience

Pilot	Total Time (hours)	Total Helicopter (hours)	Total Side-Arm (hours)
A	5800	3000	20 Research
B	4100	2400	400 Cobra
C	3500	3000	5 Cobra
D	9200	5700	5 Cobra

FIG. 1: THE NAE BELL 205 AIRBORNE SIMULATOR

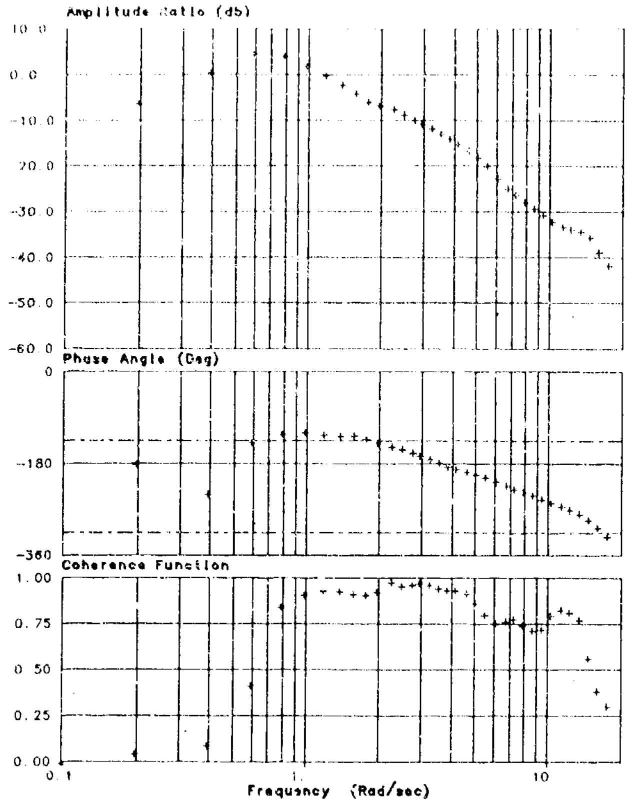


FIG. 3: ROLL AXIS BODE PLOT, δ_ϕ TO ϕ(RAD) (HOVER FLIGHT CONDITION)

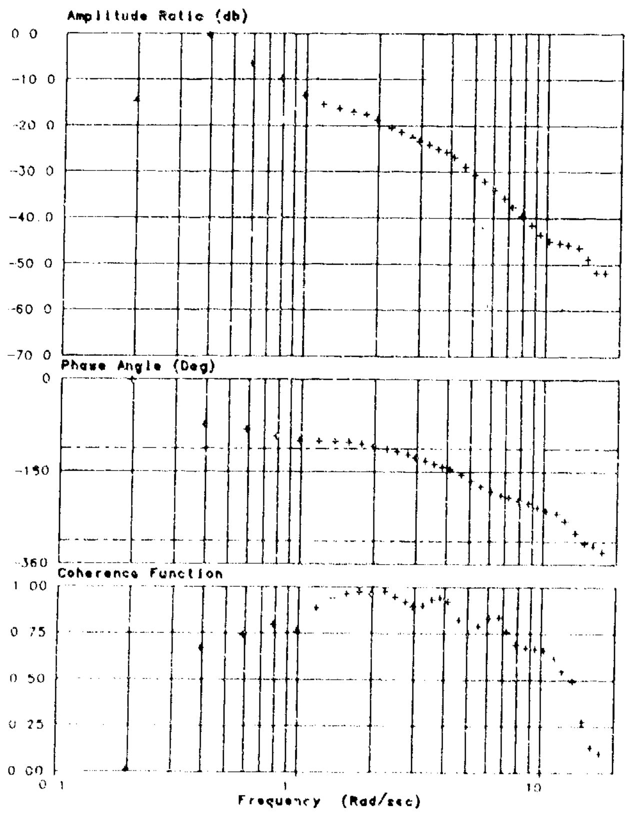


FIG. 2: PITCH AXIS BODE PLOT, δ_θ TO θ(RAD) (HOVER FLIGHT CONDITION)

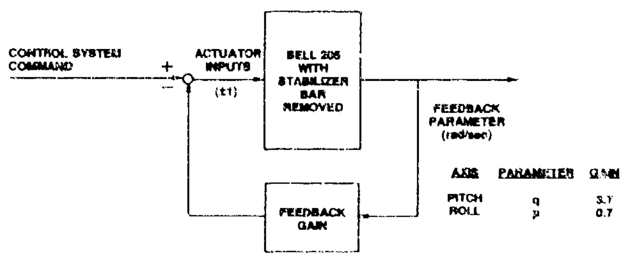
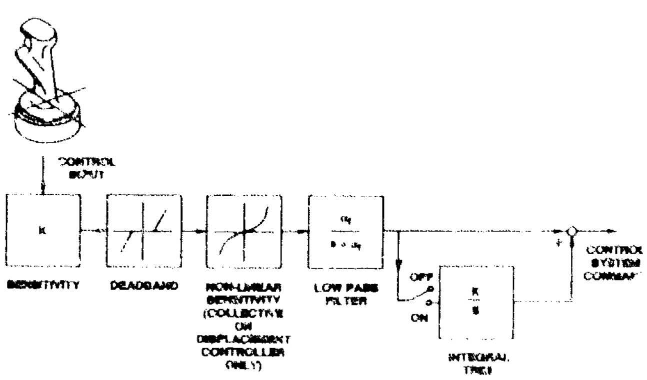


FIG. 5: PITCH AND ROLL CONTROL SYSTEM ARCHITECTURES



NOTE: SEE TABLE 1 FOR APPROPRIATE VALUES

FIG. 4: CONTROL INPUT CONDITIONING

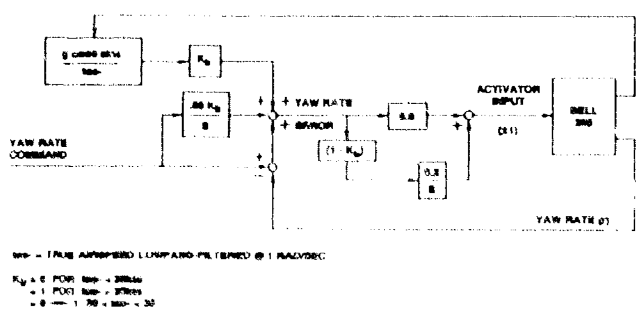


FIG. 6: YAW AXIS CONTROL SYSTEM

FIG. 7: FORCE SENSING SIDEARM CONTROLLER

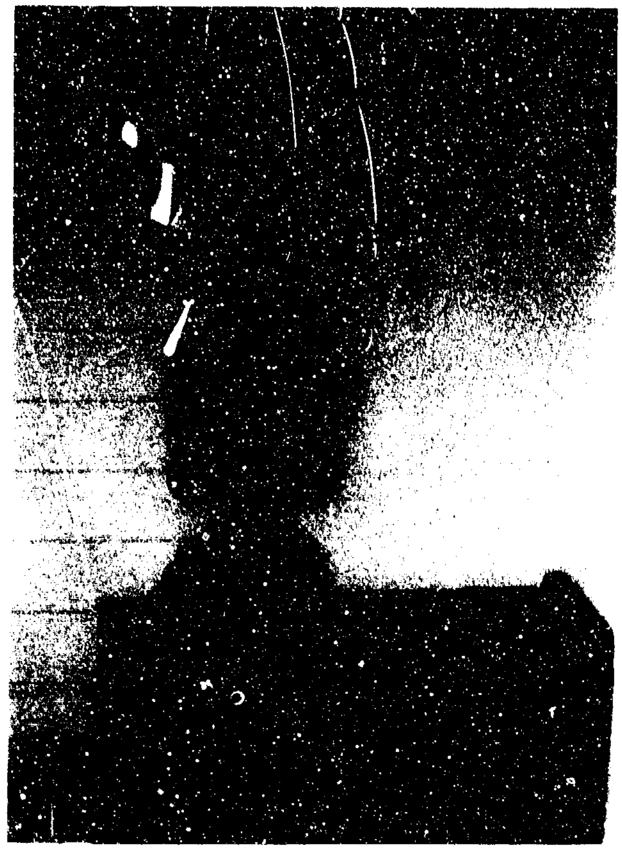
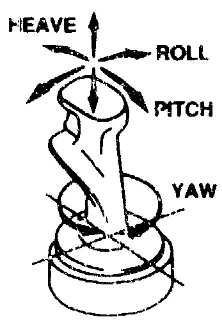
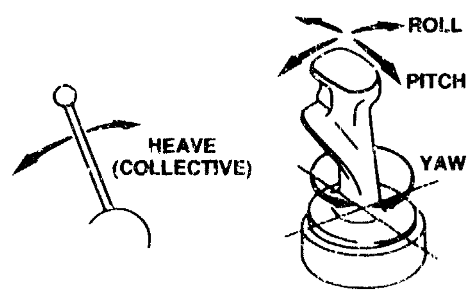


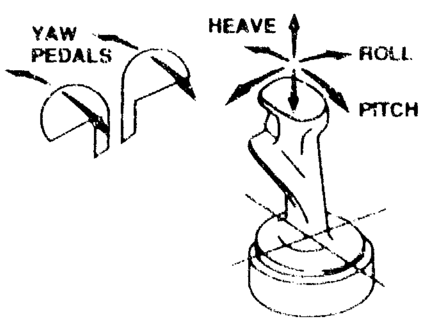
FIG. 8: DISPLACEMENT SENSING SIDEARM CONTROLLER



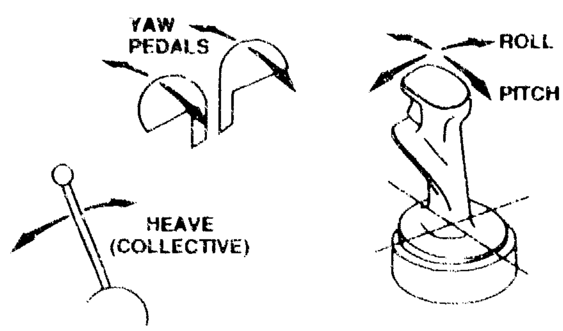
4 + 0 - FULLY INTEGRATED



(3 + 1)c - SEPARATE COLLECTIVE



(3 + 1)p - SEPARATE PEDALS



(2 + 1 + 1) - FULLY DISTRIBUTED

FIG. 9: CONTROLLER INTEGRATION LEVELS

Sidestick Applications in Civil Rotorcraft

Evaluation Pilot ______ Safety Pilot
Flight Number 09- File Numbers ______ Date ______
HSI / CAH / CONV (4+0) / (3+1)c / (3+1)p / (2+1+1)
Cooper Harper Handling Qualities Ratings and Comments

Precision Hover

Vertical Landing

Sidestep

Divided Attention Hover

Pirouette

Figure 8

Quickstop

Slope Landing

Obstacle Clearance Takeoff/Steep Approach

** Based on the handling qualities demonstrated during this flight, would you issue certification for this configuration (yes, marginal or no)? If your response is marginal or no, which manoeuvres drove you to this decision and what should be improved on the configuration to allow its certification?

What part did the current wind and turbulence conditions play in your evaluation?

FIG. 10: EVALUATOR QUESTIONNAIRE

Name: ______

GENERAL COMMENTS ON FAA/NAE

SIDEARM CONTROLLER EXPERIMENTS

1 Please provide general comments on each control configuration which you have flown. In particular, an assessment of certifiability, control deficiencies and whether you feel that modifications are possible to correct these deficiencies or improve the controller.

2 Tasks were selected for this program with two aims in mind, one was to represent typical manoeuvres in addressing certification standards, but others were included in an attempt to highlight possible deficiencies when integrating control functions on one controller. Please comment on the adequacy of the selected manoeuvres in achieving these two aims.

3 What major advantages do you feel could be gained with the use of sidearm controllers in civil rotorcraft?

4 Based on the best controller configuration which you experienced, what improvements would you suggest?

5 If you were presented a vehicle for certification which incorporated a sidearm controller (not necessarily one of the configurations presented in this experiment) based on your current experience on sidearm controllers, what major issues would you concentrate on during the flight tests?

6 Any additional comments? (Please use the back of this page)

FIG. 11: GENERAL QUESTIONNAIRE

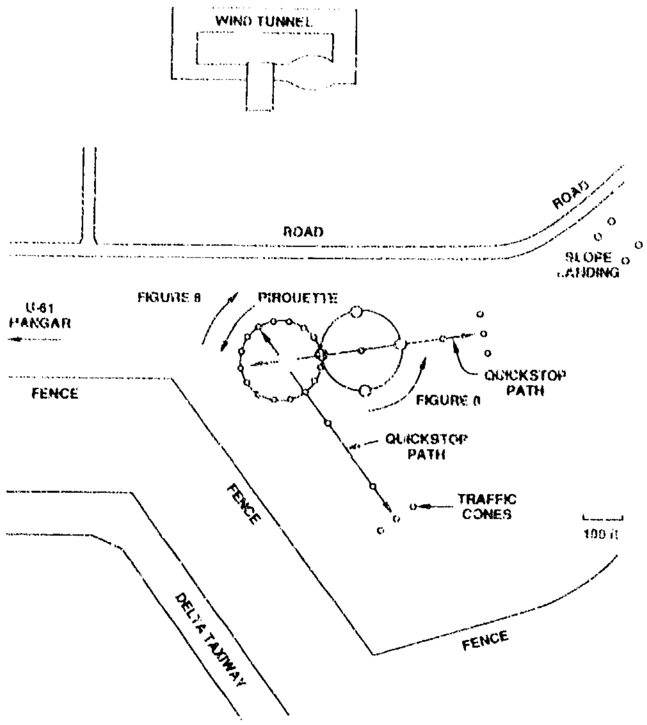


FIG. 12: EVALUATION COUPSE

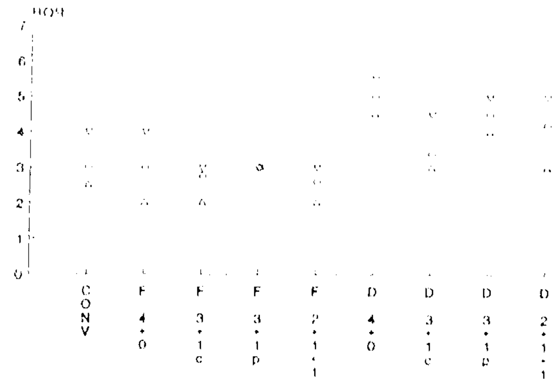


FIG. 15: SIDESTEP RATINGS

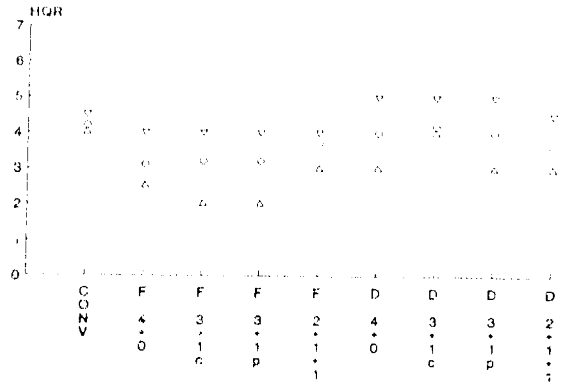


FIG. 16 DIVIDED ATTENTION HOVER RATINGS

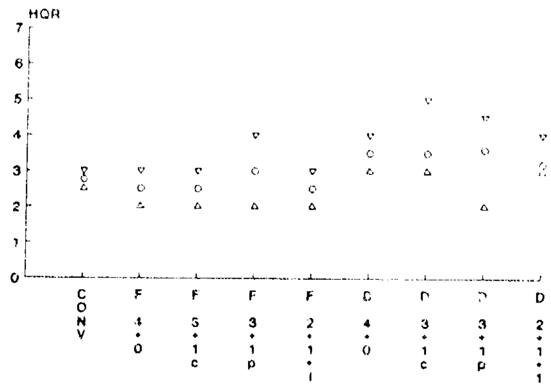


FIG. 13: HOVER RATINGS

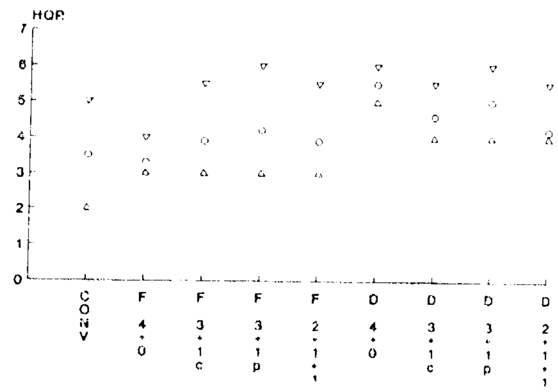


FIG. 17: PIROUETTE RATINGS

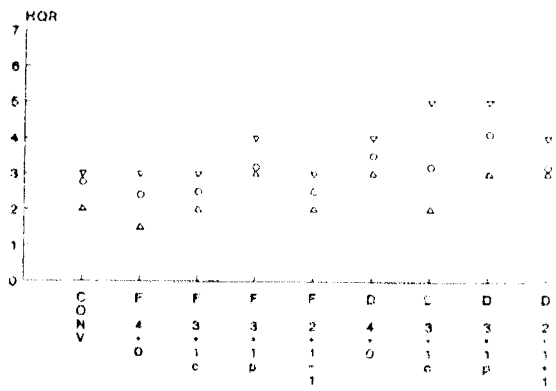


FIG. 14: LANDING RATINGS

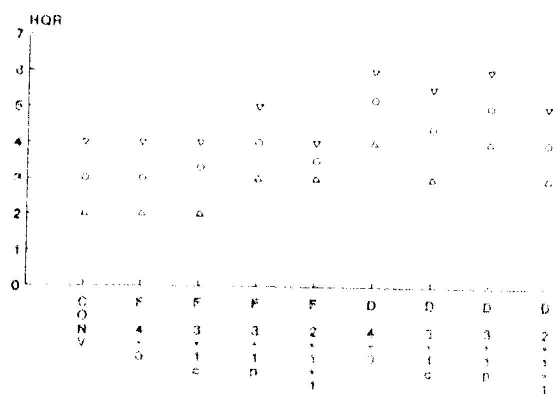


FIG. 18: FIGURE EIGHT RATINGS

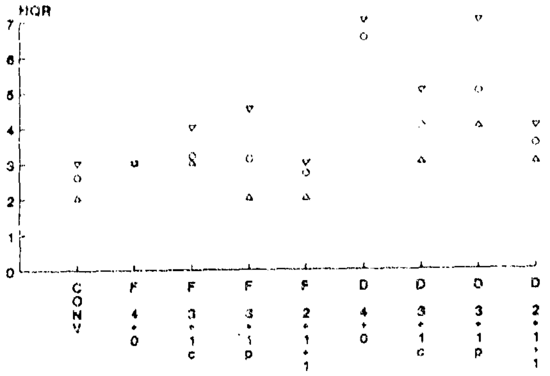


FIG. 19: QUICKSTOP RATINGS

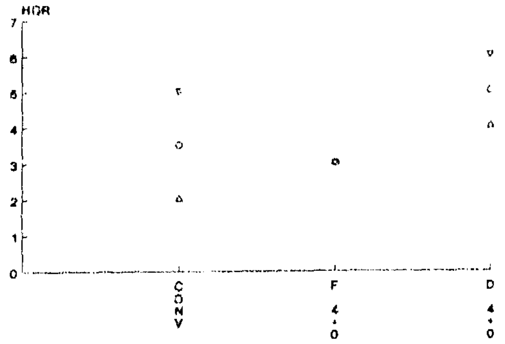


FIG. 22: IFR DECELERATING APPROACH RATINGS

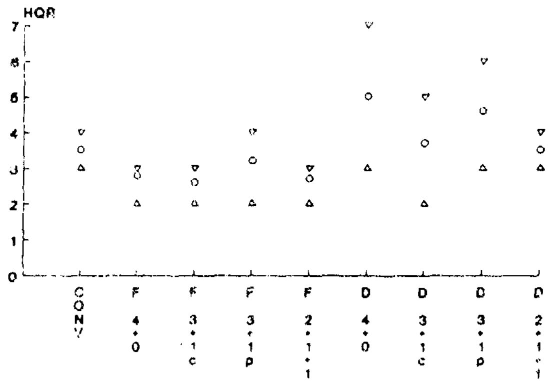


FIG. 20: SLOPE LANDING RATINGS

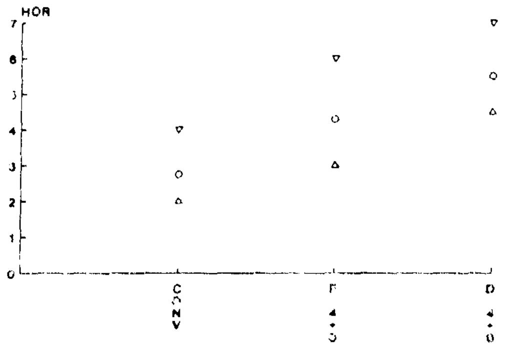


FIG. 23: AUTOROTATION RATINGS

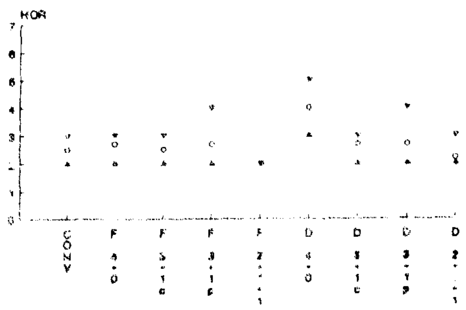


FIG. 21: OBSTACLE CLEARANCE TAKEOFF AND STEEP APPROACH RATINGS

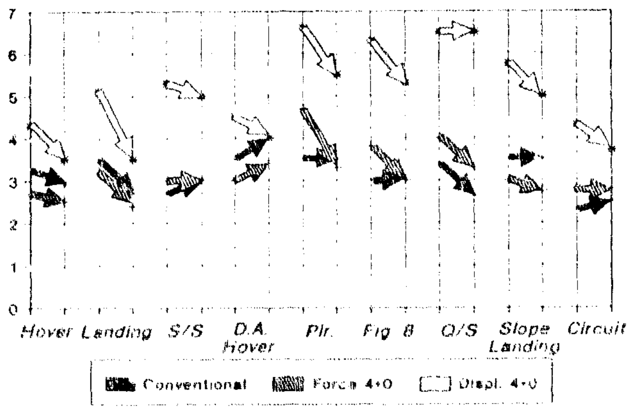


FIG. 24: LEARNING TRENDS ON INTEGRATED SIDEARM CONTROLLERS

DETERMINATION OF DECISION-HEIGHT WINDOWS FOR DECELERATING IMC APPROACHES IN HELICOPTERS

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1 SUMMARY

This program was conducted to define the basic limitations of the pilot plus rotorcraft in making the transition from a very low decision height (DH) to a steady hover over the helipad. The term "decision-height window" is defined herein as the limits of glideslope/localizer tracking errors, and groundspeed variations, that can exist at breakout to allow a safe visual transition to hover. The dimensions of the decision-height window can have a significant impact on the required rotorcraft handling qualities, and for setting autopilot coupler and flight director performance standards for decelerating instrument approaches in rotorcraft.

There have been several FAA and NASA experiments conducted to investigate tracking accuracy for helicopter instrument, and visual approaches, e.g., see Refs. 1 and 2. However, this work is the first to consider the required tracking and speed tolerances (i.e. "window") for decelerating instrument approaches to a very low decision height (50 ft above ground level (AGL)).

There are a wide variety of factors that must be considered in the determination of the dimensions of a decision-height window, e.g.,

- Rotorcraft flight dynamics and handling qualities limits.
- Limitations associated with the human pilot.
- Rotorcraft field-of-view
- Availability of airspace.
- Available real estate for the helipad, and approach lighting.
- Rotorcraft performance for a missed-approach

This paper is concerned with the first two factors noted above. It is intended that the results of this study will be superimposed on the other considerations for the determination of a decision-height window for a given set of conditions consisting of rotorcraft performance limits, MLS configuration, and helipad geometry.

This work was done in the context of an exploratory study to determine what factors are important, and to obtain a general idea of the order of magnitude of the dimensions of the decision-height window. Further testing should be conducted to determine the effects of different rotor configurations (the test aircraft was a Bell 205 with a teetering rotor), helipad geometry and lighting, rotorcraft field-of-view, glideslope angle, and a more detailed look at the effects of winds. The testing focused primarily on the longitudinal axis, and further work to determine the maximum allowable lateral offsets should be accomplished.

The program was conducted jointly by the Federal Aviation Administration (FAA) and the Canadian National Research Council (NRC) under a memorandum of agreement between those two agencies. The test aircraft was the NRC variable stability Bell 205A, and all testing was conducted at the NRC facility in Ottawa Canada. Hoh Aeronautics Inc. (HAI) provided technical assistance in this program under an FAA subcontract. A more detailed description of the program may be found in technical reports published by the FAA and the NRC (Refs 3 and 4 respectively).

FUNDAMENTAL CONSIDERATIONS

Approach Geometry

The nature of the approach geometry for very low decision-heights is such that glideslope errors can result in significant differences in the range to the hover point from breakout, as illustrated in Figure 1. Here it can be seen that a high approach results in a steeper visual segment (defined by γ_f in Figure 1). This can be alleviated by increasing the hover altitude, but possibly at the expense of loosing the helipad under the nose, or risking reentry into cloud and a missed approach. In this program, the nominal glideslope angle (γ_o) was 9^o, the decision height was set at 50 ft, and the hover height was 10 ft, resulting in a nominal final segment (γ_f) of 7.2^o.[1] If the pilot is below glideslope at breakout, the final segment (γ_f) will be more shallow than the reference 7.2^o. Steep approaches are more critical from the standpoint of flight dynamics, and shallow approaches are critical in terms of helipad sighting at breakout, or collisions with obstructions. This program was focused primarily on the flight dynamics and handling qualities issue, and hence concentrated on the problem of being high and/or fast

Relationships Between Decision Height Window and Helicopter Performance Limits.

Helicopter performance limits that can have an effect on the transition from decision height to hover are:

1. Insufficient acceleration capability without entering a region of unacceptable handling qualities or autorotation.

2. The restricted height-velocity envelope for single engine rotorcraft.

3. Settling with power, or vortex ring state.

4. Degraded handling due to low airspeed resulting in transition in and out of translational lift.

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[1] All of the glideslope angles in this study are negative. The sign of these angles has been dropped in the text and figures as a matter of convenience. Also, reference to greater or steeper values of glideslope angle are intended to refer to more negative values.

These are illustrated in Figure 2, on a generic decision-height window. The shape of the upper boundary of this window indicates a tradeoff between groundspeed and glideslope error. This is a result of the first of the above factors, which infers a limit on the ability of the rotorcraft to dissipate energy. The total energy is the sum of potential energy (Δh), and kinetic energy (V^2), so that a tradeoff exists between groundspeed, and glideslope error at breakout.[2]

Height-velocity problems could occur for a breakout at low airspeed and a high-on-glideslope condition (i.e. large value of γ_f). This condition would be most likely if there were a tailwind at breakout. Even if the rotorcraft is not in the height-velocity envelope, a close proximity to this condition may be limiting from a piloting standpoint.

The potential for steep approach angles dictates that the vortex-ring state issue be addressed, as illustrated in Figure 2. The sketch in Figure 2 is taken from the full-scale rotor data in Ref. 5, which indicates that ring-vortex encounters require a flight path angle of 30^o or greater.[3] Actually, it is the angle-of-attack of the rotor that is fundamentally limiting, and the interpretation in terms of flight path angle assumes a nearly level pitch attitude, and that the flight path angle is with respect to the airmass (i.e. the "aerodynamic" flight path angle). It follows that the most critical condition would be a breakout high on the glideslope, and in a tailwind. The relationship between the aerodynamic and inertial flight path angles is:

$$\tan\gamma_{aero} = \tan\gamma_{inertial}\left(1 + \frac{V_{wind}}{V_{airspeed}\cos\gamma_{aero}}\right)$$

For $\gamma \leq 20^o$,

$$\gamma_{aero} \approx \gamma_{inertial}\left(1 + \frac{V_{wind}}{V_{airspeed}}\right)$$

where $\gamma_{Inertial}$ would be equal to γ_f to make the pad from the decision point, and a positive wind is a tailwind. Practically speaking, it is unlikely that an approach would be conducted in tailwinds strong enough to increase the aerodynamic flight path angle to values large enough to enter the vortex ring state for glideslopes of 12^o or less. However, for steeper glideslopes, it may become a limiting factor.

The maximum pitch attitude illustrated in Figure 2 is related to performance, in that the only way to convert low power to deceleration along the flight path is through pitch attitude. Hence, the peak pitch attitude is a measure of the power, or energy deficiency, that can be converted into deceleration without changing flight path. Because of the steepness of the constant power curves at very low airspeeds (backside of the power-required curve), simply holding collective constant while decelerating results in a rapid need for increasing pitch attitude as speed approaches zero, to hold flight path angle constant (e.g., like in a quickstop maneuver). A fast condition at breakout can result in a need for significant deceleration all the way to hover. In such cases, the pilots in the present experiment were observed to delay adding collective until the very end,

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[2] Groundspeed is used because it is, by definition, zero at hover. Therefore, any groundspeed that exists at breakout must be dissipated at the hover point or an overshoot will occur.

[3] The edge of the vortex ring region involves some recirculation of air through the rotor plane. This shows up as increased vibration, but with little accompanying loss of performance. Proceeding deeper into the vortex-ring region involves recirculation of air over a significant portion of the rotor (black region of Figure 2), and settling with power.

and as a result, large peak pitch attitudes occurred just prior to hover. The magnitude of the peak pitch attitude is directly related to the minimum power that the pilot is willing to use in the final deceleration to hover.

The left side of the decision-height window is defined by the minimum airspeed that can be comfortably flown during the transition from IMC to VMC. Many helicopters do not posses good flying qualities in the transition region between forward flight and hover (region of effective translational lift). For example the test aircraft (a Bell 205A) tended to "buck and gallop" when operating in this region, which was considered as unacceptable by the pilots (HQR = 7). The low airspeed limit may also be set by the minimum airspeed for which IFR flight is approved. Once a minimum acceptable airspeed is established, the left boundary becomes a function of the tailwind (groundspeed = airspeed + wind) between the decision point and hover.

The lower (bottom) boundary of the DH window does not depend on the rotorcraft flight dynamics, since in the limit, the rotorcraft can fly level or even climb slightly to reach the hover point. Therefore, as noted in Figure 2, the lower limit is set by obstruction clearance and visibility constraints.

Returning to the upper boundary, it would be useful to characterize the total energy dissipation required to transition from the decision point to hover in terms of rotorcraft performance data. In that context, it is convenient to borrow a concept developed to define the vulnerability of powered lift STOL aircraft to wind shear, called the effective flight path angle, or γ_{eff} (e.g., see Ref . 6). The effective flight path angle, as it is applied to the present problem, is defined as follows.

$$a_x = \dot{V}_I + g\sin\gamma_a$$

$$g\sin\gamma_{eff} = a_x$$

$$\dot{V}_I = \frac{V_{DH}^2}{2R}$$

Where: R = Range to Hover at DH
V_{DH} = Groundspeed at DH

$$\sin\gamma_{eff} = \frac{V_{DH}^2}{2gR} + \sin\gamma_a$$

Note that the V^2 term is indicative of the kinetic energy required to decelerate from the decision point to hover, and that γ_a is the angle between the horizon and the airspeed vector required to reach the pad from the decision point. The complete expression for the effective flight path angle, including the effect of winds, is derived in Refs 3 and 4, and is given as follows.

$$\sin\gamma_{eff} = \frac{V_{DH}^2}{2g\sqrt{R^2 + (h_{DH} - h_{HOV})^2}} + \frac{V_{DH}\sin\gamma_f}{\sqrt{V_{DH}^2 + V_w^2 - 2V_{DH}V_w\cos\gamma_f}}$$

Where V_w is the wind velocity; positive as a tailwind.

From the geometry in Figure 1, the range from the decision point to hover over the pad is a function of the nominal glideslope angle, γ_O, and the glideslope error, d_ϵ as follows:

$$R = \frac{(h_{DH} - d_\epsilon)}{\tan\gamma_o}$$

Finally, for glideslope angles less than about 20^O, the slant range is approximately equal to the horizontal distance ($R_S \approx R$, $\cos\gamma_f \approx 1$, and $\sin\gamma_f \approx \gamma_f$). For zero wind, this yields the following approximation.

$$\frac{\gamma_{eff}}{\gamma_o} \approx \frac{1}{(h_{DH} - d_\epsilon)} \left(\frac{V^2_{DH}}{2g} + h_{DH} - h_{HOV}\right)$$

For DH = 50 ft. and a 10 ft hover:

$$\frac{\gamma_{eff}}{\gamma_o} \approx \frac{1}{(50 - d_{50})} \left(\frac{V^2_{50}}{2g} + 40\right)$$

This approximation was found to be accurate within a few percent for all of the cases tested in this flight experiment. The extrapolations to larger flight path angles discussed later in the paper utilized the more exact expression for effective flight path angle. As noted above, the $V^2/2g$ term may be thought of as proportional to the kinetic energy, ($h_{DH} - h_{HOV}$) as proportional to the potential energy, and γ_{eff}/γ_O as a measure of the total energy that must be dissipated. This concept allows one to lump the deceleration requirement and geometric flight path angle into a single "effective" flight path angle. The advantage of this is that γ_{eff} can be plotted directly on the rotorcraft γ vs. V performance curves as shown in Figure 3. These example γ vs. V curves are for the XV-15 tilt rotor aircraft in the helicopter mode. In the example shown in Figure 3, the nominal glideslope angle is 12^O, the groundspeed at the 50 ft decision height is 20 kts, and the rotorcraft is assumed to reach the decision height with a 25 ft glideslope error. The component of γ_{eff} due to the geometric flight path angle from the decision point to a 10 ft hover (γ_f in Figure 1) is 19.2^O; 12^O due to the nominal glideslope, and 7.2^O due to being 25 ft high. The component of the effective flight path angle due to the need to decelerate from 20 kts to zero groundspeed is 8.5^O (the V^2 term in the definition of γ_{eff}). This results in a total effective flight path angle of 27.7^O.

The effective flight path angle will be the basis for mapping the rotorcraft performance capability on to the "decision height coordinates", defined here as glideslope error vs. groundspeed at DH (d_ϵ vs. V_{DH}), and for extrapolating the results of this program to other rotorcraft and glideslope angles.

EXPERIMENTAL SCENARIO AND RESULTS OF INITIAL EXPLORATORY RUNS

The NRC variable stability Bell 205A was configured to have handling qualities similar to a current rotorcraft with a limited authority stability augmentation system (e.g. the Sikorsky S-76). A conventional cyclic stick and collective were provided as cockpit controllers. The initial phase of the testing was to determine the γ-V characteristics of the Bell 205A to allow estimates of the decision height window based on the effective flight path angle. The resulting γ - V curves are shown later in Figures 9 and 10. It should be noted that the horizontal component of the airspeed vector is plotted as the x coordinate in these figures to allow a direct comparison with groundspeed at DH (i.e. so airspeed = groundspeed in zero wind).

The nominal glideslope angle in the experiment was 9^{0}, and some runs were made with a 6^{0} glideslope. Initial testing indicated that glideslope angles of over 9^{0} resulted in excessive down collective requirements for glideslope corrections in the Bell 205.

Each run was initiated in level flight on the localizer, or just prior to turn-on to the final approach course. The evaluation pilot flew the precision approach to a 50 ft decision height in simulated instrument meteorological conditions (IMC), and completed the approach to a hover over the helipad visually (below 50 ft AGL). IMC was simulated by means of electronically fogged goggles, which were wired to the radar altimeter, and which automatically cleared at 50 ft thereby simulating breakout. The goggles were cleared on every approach so that the evaluation pilot did not have to make the missed-approach decision at DH.

The deceleration profile was taken from the Ref. 7 and 8 work, and consisted of a constant attitude deceleration of about .045 g. from 60 kts groundspeed to a nominal 20 kts groundspeed at DH. The radar altitude box on the EADI, and the digits within the box flashed at 10 ft above decision height, and remained flashing while below this height.

Nearly all of the runs were made using the manual flight director for the IMC portion of the approach. A few coupled runs were made, and these indicated that the transition was slightly more difficult because of the additional workload of disengaging the coupler, and transitioning into the control loop. There was some concern that the details of the mechanization of the autopilot cutoff could have a significant impact on the results, so it was decided to proceed with manual approaches for this initial exploratory program.

It was determined that the pad markings investigated in this program did not have a significant effect on the transition from DH to hover. The baseline pad consisted of eight orange traffic cones outlining a square which measured 100 feet on each side. A second pad was tested which consisted of the same basic markings plus lead-in cones which depicted two large arrowheads along the final 350 feet of the final approach course. These lead-in cones were of little or no value because they were under the nose almost immediately after breakout. essentially all of the runs were made to the pad without lead-in cones. In actual practice, such lead-in guidance would be useful to maximize the lower boundary of the DH window by extending the range of the "helipad environment". It was hypothesized by one pilot that lead-in strobes surrounding the pad, and pointing the way to the center of the pad, would be useful to assist in finding it immediately upon breakout, especially with a lateral offset.

The majority of the runs were made to obtain estimates of the size and shape of the critical upper boundary of the decision-height window using a 9^{0} glideslope angle. The conditions at DH were systematically varied by injecting errors into the flight director control laws. The raw data displayed to the pilot was not affected so the pilot was aware of the errors at DH, as he or she would be in actual operations. This provides a valuable cue since the pilot "knows where to look" for the pad during the first few critical seconds after breakout

Systematic variations of localizer errors were beyond the scope of the program. However, the lateral flight director was degraded from the Ref. 7/8 study, and this resulted in random localizer errors at breakout which ranged from zero to about 60 ft. These lateral errors were not deemed to be a significant problem by any of the pilots.

Data was collected for four pilots. Two pilots were test pilots from the NRC, one is an FAA certification pilot from the Southwest region, and the other is the author of this paper. All are commercially rated helicopter pilots with significant experience in handling qualities flight testing, and the use of the Cooper Harper handling qualities rating (HQR) scale.

ANALYSIS OF RESULTS

Correlation of Pilot Rating Data and Commentary

The detailed pilot rating results are presented in a spreadsheet in Ref. 3 along with a summary of the pilot commentary for each run. The data for the 9° glideslope on the DH window coordinates (d_{ϵ} vs V_{DH}) in Figure 4.

The data fairings in Figure 4 (which define the upper boundaries of several example DH windows to be discussed below) resulted from the following interpretations of the pilot rating results.

- All regions of the DH window coordinates containing points with HQR = 7 have been defined as "unacceptable", even if other data points in that region have been rated HQR = 4, and 5. The rationale is that these 7 rating(s) resulted from a slight delay or initial misuse of the controls in initiating the proper action from a very marginal initial condition (see pilot commentary in Reference 3 or 4). If the region is so critical that an overshoot can result from a slight delay or mistake, it should be disallowed for normal commercial operations. This interpretation of the data causes the curves to drop off steeply at speeds above 25 knots due to two points with HQR = 7 at 32 and 36 kts groundspeed. These two data-points cause a number of cases with HQR = 4 to fall in the unacceptable region. This is discussed in more detail below.

- Regions with a large number of 6s, some 5s and an occasional 4 have been defined as unacceptable. This is based on the rationale that HQR = 6 is indicative of a requirement for extensive pilot compensation just to make the pad without an overshoot (i.e. "adequate performance"), and/or very objectionable but tolerable deficiencies. A region where such ratings are abundant seems excessively risky for commercial operations to very low instrument minimums. This interpretation causes the left side of the upper boundary to be flatter than it would be if only HQRs of 7 were considered as unacceptable (see Figure 4).

- The "marginal region" contains mostly points which were assigned HQRs between 4 and 5. That is, "deficiencies warrant improvement" and adequate performance requires "moderate" to "considerable" pilot compensation on the Ref. 9 HQR scale.

- The "desirable region" consists of mostly ratings of 3 or better, i.e., "satisfactory without improvement" on the HQR scale. A few points with HQR = 4 in this region all contain pilot comments related to vibration; a problem unique to the Bell 205A and not related to the DH window.

The desirable region in Figure 4 is proposed as a design goal for approach couplers and flight directors intended to achieve a decision-height of 50 ft following a decelerating approach in IMC conditions. The suggested demonstration would be similar to that used for fixed wing approach couplers for Category IIIa autoland systems.[4]

The outer limit of the marginal region is proposed as the missed approach boundary, and would be full scale deflection on the raw data for the final segment of the approach.

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[4] Current fixed-wing category IIIa couplers are usually designed and certified within the guidelines of Advisory Circular 20-57A which specifies touchdown performance limits which must be achieved.

The above rationale is based, in part, on the pilot commentary for ratings which fall along and outside the faired boundaries. These are summarized on Figure 5. This figure includes the faired lines from Figure 4, and several shaded ellipses to indicate general regions of pilot commentary. The right-most ellipse indicates that the upper limit on groundspeed at DH is defined by excessive pitch attitudes and high collective activity (essentially a quickstop maneuver). Moving up and to the left, the commentary still indicates that the pitch attitude is marginally high, collective activity is excessive, and that pad visibility is becoming a problem. The left-most ellipse indicates a region where the pilots were concerned with rotorcraft performance limits associated with steep approaches, and that pad visibility has become a significant problem. It should be noted that the glare-shield on the Bell 205 is reasonably low, and that the visibility over the nose of some more modern helicopters is somewhat worse. The urgency along the upper boundary is generally classified as moderate; not a good situation for a transition from IMC to a VMC hover, at altitudes below 50 ft AGL.

The faired boundaries from Figure 4 are presented without pilot ratings or commentary in Figure 6. The shaded region on the right side of Figure 6 contains primarily HQRs of 4 to 4.5 with two notable exceptions - a pair of 7s. The upper limit of this region is defined by a line of constant γ_{eff}. In fact, if it were not for the two pilot ratings of 7, the line of $\gamma_{eff} = 20^o$ would nicely separate the HQR 6 and 7 ratings from the HQR 4 and 5 ratings. Hence, it can be argued that an effective flight path angle of 20^o represents the limit of the energy dissipation capability of the Bell 205, <u>as long as the pilot responds immediately, and with the correct control inputs at DH</u>. However, if the pilot encountered some delay (such as in making a transition from fully coupled to manual flight, or lacking good speed and sink-rate cues at DH), an overshoot of the pad, and/or excessive pitch attitudes tended to result. A margin or pad is required from the $\gamma_{eff} = 20^o$ line for groundspeeds above about 25 kts to eliminate the shaded region in Figure 6. A review of the time histories of runs along this more restricted boundary indicate that it results in a maximum pitch attitude of about 14^o. For the two runs in the shaded region where the pilots started late and assigned ratings of 7, a maximum pitch attitude of 21^o was observed. In fact, this is approximately the maximum pitch attitude observed for all runs in the unacceptable region for all of the pilots.

The peak pitch attitude used during the final portion of the deceleration to hover is closely correlated to the minimum power (collective position) used by the pilot. It is therefore not surprising that the steep upper boundaries which correlate the pilot rating data in Figure 4 are well fitted by lines of constant torque (power). This is illustrated in Figure 7 where it can be seen that the data fairings (thick lines) follow lines of constant torque after departing lines of constant γ_{eff}. The usefulness of this observation will be discussed in more detail in the following section.

DEVELOPMENT OF EXAMPLE DECISION-HEIGHT WINDOWS

The physical significance of γ_{eff} is that it represents an equivalent task that is more easily understood and interpreted than the complex deceleration to hover from breakout. The "equivalent task" is to fly at a constant flight path angle ($\gamma = \gamma_{eff}$) and at the speed at DH. For example, if the helicopter arrives at the decision-height 35 ft above a nominal 9^o glideslope, and at a groundspeed of 25 kts, γ_{eff}/γ_o is calculated to be 3. Therefore, $\gamma_{eff} = 27^o$ (3×9^o), and the task of decelerating to hover is equivalent to flying at a constant speed of 25 kts and a flight path angle of 27^o. Plotting this on the γ - V curves for the Bell 205, (e.g. see Figure 8) indicates that it would require a very low amount of torque (approximately 7 psi), and hence is near a basic rotorcraft deceleration limit.

While the γ - V curves represent the theoretical performance capability of the rotorcraft, it is well known that there are other restrictions which must be superimposed on these curves. Some restrictions are performance related (e.g., vortex ring state, height-velocity curve, and vibration), and other restrictions tend to be centered about degraded handling qualities. The results discussed above indicate that below about 25 kts groundspeed, the limit consists of a line of constant effective flight path angle, γ_{eff}. Above 25 kts, the boundaries departed from a constant γ_{eff} by bending down sharply and thereby limiting the maximum groundspeed (e.g., see Figures 4 or 6). Interestingly these departures from constant γ_{eff} are well approximated by lines of constant torque or power. This is illustrated in Figure 7, where lines of constant torque, and lines of constant aerodynamic flight path angle have been mapped from the γ - V coordinates onto the DH window grid of glideslope error vs. groundspeed. Here, it can be seen that the data fairings in Figures 4 and 6 can be well approximated by lines of constant effective flight path angle and lines of constant torque as follows.

Nominal missed approach boundary	Extended Missed Approach Boundary
$V < 25$ kts -- $\gamma_{eff} \leq 20°$	$V < 35$ kts -- $\gamma_{eff} \leq 20°$
$V \geq 25$ kts -- Torque ≥ 10 psi	$V \geq 35$ kts -- Torque ≥ 5 psi

The above missed approach boundary definitions are superimposed on the Bell 205A γ - V curves in Figure 8a. Any point in the noted "nominal safe region" of this plot represents a flight path angle-airspeed combination that is flyable with good handling qualities. Once a safe γ - V region is defined, its boundaries can be mapped onto coordinates of d_ϵ vs. V_{DH} (via the γ_{eff}/γ_o equation by solving for d_ϵ) to obtain a decision-height window. An example of such mapping is illustrated in Figure 8b for the Bell 205, and a 9° glideslope angle.

A less conservative "extended boundary" for the missed approach window is defined if the shaded region on the right side of Figure 6 is ignored. This extended boundary is defined by $\gamma_{eff} = 20^{\circ}$, and by a line of constant torque of 5 psi above 35 kts in Figure 8a. These limits are mapped onto the d_ϵ vs. V_{DH} coordinates in Figure 8b. A comparison of the extended missed-approach window with the "nominal" window in Figure 8b indicates that the primary difference is that significantly higher airspeeds are allowed. The rationale for the less conservative extended window would be based on significantly improved cues due to helipad markings and lighting, and possibly a larger landing area or overrun. The helipad in this experiment was 100 ft on a side as compared to the pad used in the FAA tests at Atlantic City (Ref. 1) which was 150 ft on a side

As noted above, the extended DH window results from inclusion of most of the shaded region in Figure 6, and hence the two HQR = 7 ratings at V_{DH} = 32 and 36 kts. As discussed earlier, these ratings both represent cases where the pilots noted some confusion between breakout and initiating action to decelerate to hover (see pilot comments in Refs. 3 or 4 for cases 5 and 78). All other ratings in this region are HQR = 4 to 4.5. It could be argued that with better cuing or a larger landing area, any delays or initial misapplication of controls would be eliminated. The physical significance of eliminating the margin afforded by the "extended region" in Figure 6 can be illustrated on the γ - V curves in Figure 8a. The upper torque limit (10 psi) represents the nominal missed-approach window, and the lower torque limit (5 psi) represents the extended missed-approach window. Consider now the following example. If the pilot arrives at DH at point A (Figure 8a), and experiences some delay in initiating the proper action at breakout (due to poor cues, etc.), it will be necessary to operate in the "extended" region (e.g., point B) to stop at the pad. That is, since the deceleration was started late, a lower torque is required to make it. The pilot rating data indicate that this is not a problem since most HQRs in the "extended region" (i.e., the shaded region of Figure 6) are 4. If, on the other hand, the extended missed approach window is used, and the pilot arrives at DH at point B and encounters the same delay (like the two 7s at 32 and 36 kts), the possibility of "making it" is very poor since he will be operating in a region of solid 7s, i.e., point C.[5] This is potentially hazardous since reaching the decision-height point with the landing area in sight, by definition, means that an abort is no longer part of the pilots mental scenario. The overshoot would result in an accident unless there is some "overrun area"

[5] Point C cannot be mapped on to the DH window coordinates because the effective decision height is actually lower than 50 ft as a result of the pilot delay. However, operating at point C requires the deceleration characteristics that were found to be unacceptable in the experiments (i.e. torque < 5 psi and $\gamma_{eff} > 20^{\circ}$).

i.e., the larger helipad discussed above as rationale for using the extended missed approach DH window.

The point of considering the nominal vs. extended decision height window is not very interesting for the Bell 205, since it is unlikely that anyone would consider using such an old machine for IMC decelerations to a 50 ft DH. However, it does illustrate that, in general, the missed approach DH window should be based on a margin from the γ -V limits where the helicopter is not comfortable to fly. A more quantitative and generalized definition of such limits will require additional study. Such work is best accomplished in ground-based simulation where the generic characteristics of the rotorcraft can be systematically varied, and operation in marginal regions of the γ - V curves can be safely tested. The DH window for approach coupler and/or flight director requirements should obviously be based on regions of the γ - V curves where the performance and flying qualities are in the desirable range.

Effect of Configuration and Glideslope Angle

The ratio γ_{eff}/γ_0 has been shown to be a function of the glideslope error, decision height, hover height, and groundspeed at DH. This relationship is plotted in Figure 9 which gives an indication of the variations in window size as a function of γ_{eff}/γ_0. These generic curves indicate that the DH window for missed approach reduces to dimensions which may be unachievable (e.g., glideslope errors of less than 20 ft at 20 kts) when the ratio of the limiting γ_a (and hence γ_{eff}) of the helicopter, to the glideslope angle is equal to or less than 2 (i.e., $\gamma_{eff}/\gamma_0 \leq 2$). For the Bell 205 this rule of thumb would limit the glideslope angle to $20^o/2 = 10^o$ which is consistent with the experimental results. That is, initial testing indicated that glideslope angles much greater than 9^o were not practical in the Bell 205. Conversely, increasing γ_{eff}/γ_0 increases the size of the window. A direct comparison of the Bell 205 missed-approach window dimensions for glideslope angles of 6^o, 9^o, and 12^o is shown in Figure 10. Here it can be seen that the window for a 12 degree glideslope would require very stringent, but not unreasonable approach coupler performance. It is doubtful that such performance is obtainable with a flight director. Reducing the glideslope angle to 6^o increases the missed-approach DH window dimensions to the point where the flight director of Refs 7 and 8 would be adequate.

In summary, to obtain DH window with dimensions that are consistent with approach coupler and flight director performance capability:

- The aerodynamic flight path angle that can be flown without degradations in handling or excessive vibrations at low speeds, should be at least twice the glideslope angle.

- The minimum power that is practically usable must not result in an overly restrictive right boundary.

- The approach coupler or flight director laws must provide very tight tracking in the presence of all expected winds, wind-shears, and gusts.

Work needs to be done to determine the flight path angle capability of modern rotor systems. The results of the Ref. 7/8 program would indicate that fully coupled approaches will be necessary for steep flight path angles unless the flight director deficiencies noted therein can be resolved. A review of 3-cue flight director control laws and tracking performance is given in Ref. 10.

Effect of Helipad Dimensions

The down-range dimension of the helipad will clearly have a significant impact on the size of the longitudinal decision-height window. This effect may be estimated by assuming that the target hover point is 50 feet from the far end of the pad (i.e., superimpose the pad used in the present experiment on the far end of the pad with increased dimensions). This will result in a decreased value of γ_0, and hence increased

γ_{eff}/γ_O. The increased dimensions of the decision-height window can then be estimated from the γ_{eff}/γ_O equation as plotted in Figure 10.

SUMMARY OF RESULTS AND CONCLUSIONS

A methodology has been developed to determine certain critical decision-height window dimensions based on pilot/rotorcraft factors and limitations. Specifically, the method consists of determining the regions of the γ - V performance curves where the rotorcraft flying characteristics are acceptable, and mapping these regions onto coordinates that define the DH window (i.e., d_ϵ vs V_{DH}). This mapping is accomplished via the "effective flight path angle" parameter or γ_{eff}.

Insights and initial estimates of the DH window dimensions have been obtained from an exploratory flight test program with the Canadian NRC variable stability Bell 205. These results are summarized below.

- The upper and right boundaries of the DH window are based on helicopter performance limits.
- The left boundary of the DH window is based on rotorcraft handling at very low airspeeds.
- The bottom boundary of the DH window is based on obstruction avoidance, and pad visibility. This boundary is not affected by rotorcraft performance or handling qualities.
- The right boundary of the DH window is based on the minimum usable torque (power), and related maximum acceptable pitch attitude during deceleration. Some margin is required to account for pilot delay in initiating the deceleration after breakout.
- The upper boundary of the DH window is based on the maximum aerodynamic flight path angle that can be flown at low airspeeds. It requires some margin for pilot delay or control misapplication in conditions of poor visual cuing, but is less critical in this regard than the right boundary.
- The dimensions of the DH window are directly proportional to the ratio of the maximum usable aerodynamic flight path angle (i.e., γ_{eff}) to the glideslope angle (γ_O). Values of $\gamma_{eff}/\gamma_O \leq 2$ result in DH window dimensions that would be difficult to achieve consistently with existing flight hardware in a variety of wind and windshear conditions.

The DH window dimensions obtained from considerations of pilot/rotorcraft factors and limitations should be superimposed on other operational considerations such as field-of-view, airspace availability, available real estate for helipad and approach lighting, obstructions, noise, etc. The ultimate decision on setting the window dimensions should be based on flight testing on a case-by-case basis. The results of this study, and any future studies, should provide a basis for estimating reasonable window dimensions prior to such testing.

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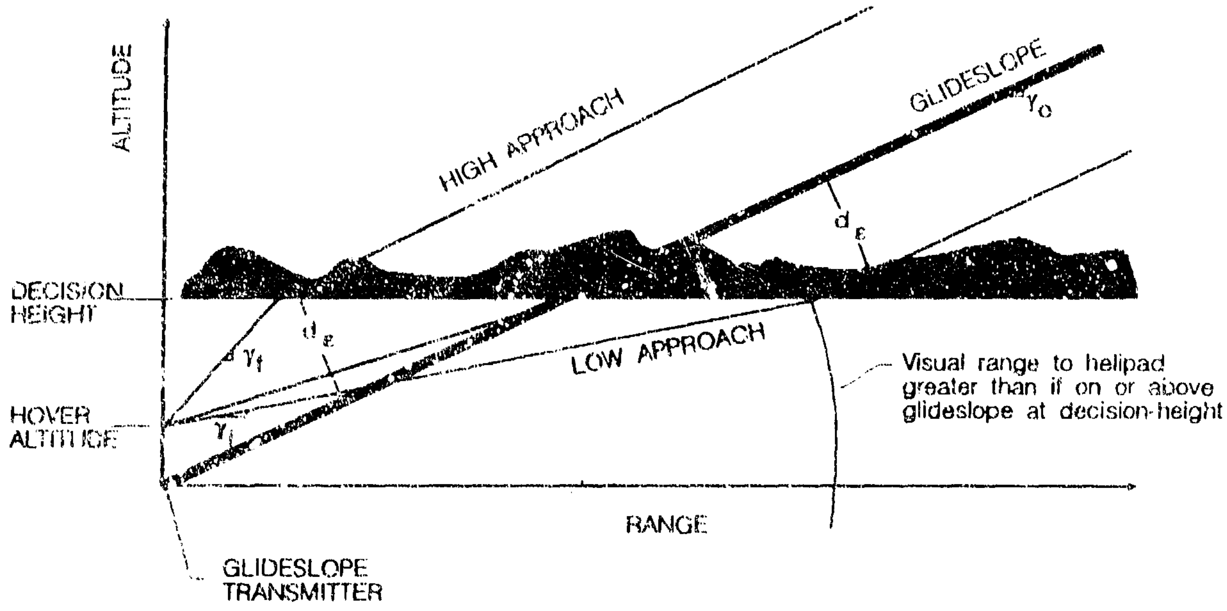

Figure 1. Approach Geometry Between Decision-Height and Hover

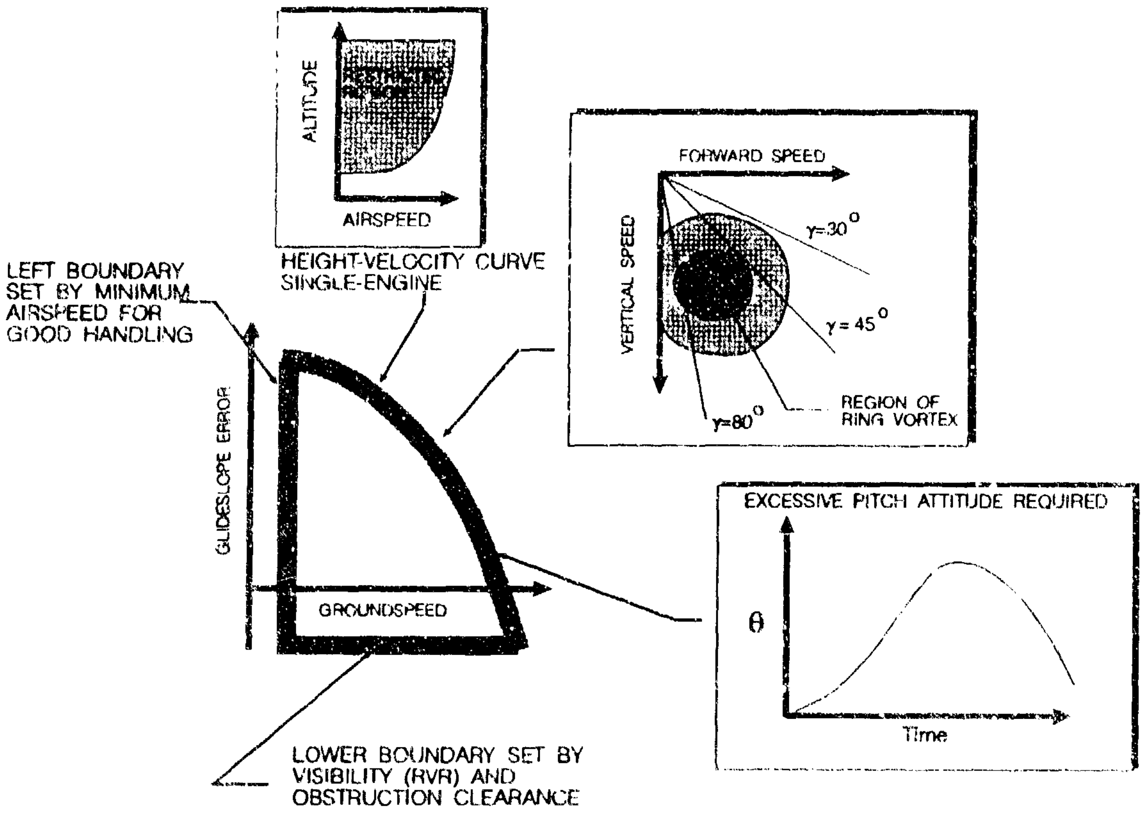

Figure 2. Basic Rotorcraft Limitations That May Affect Decision-Height Boundaries

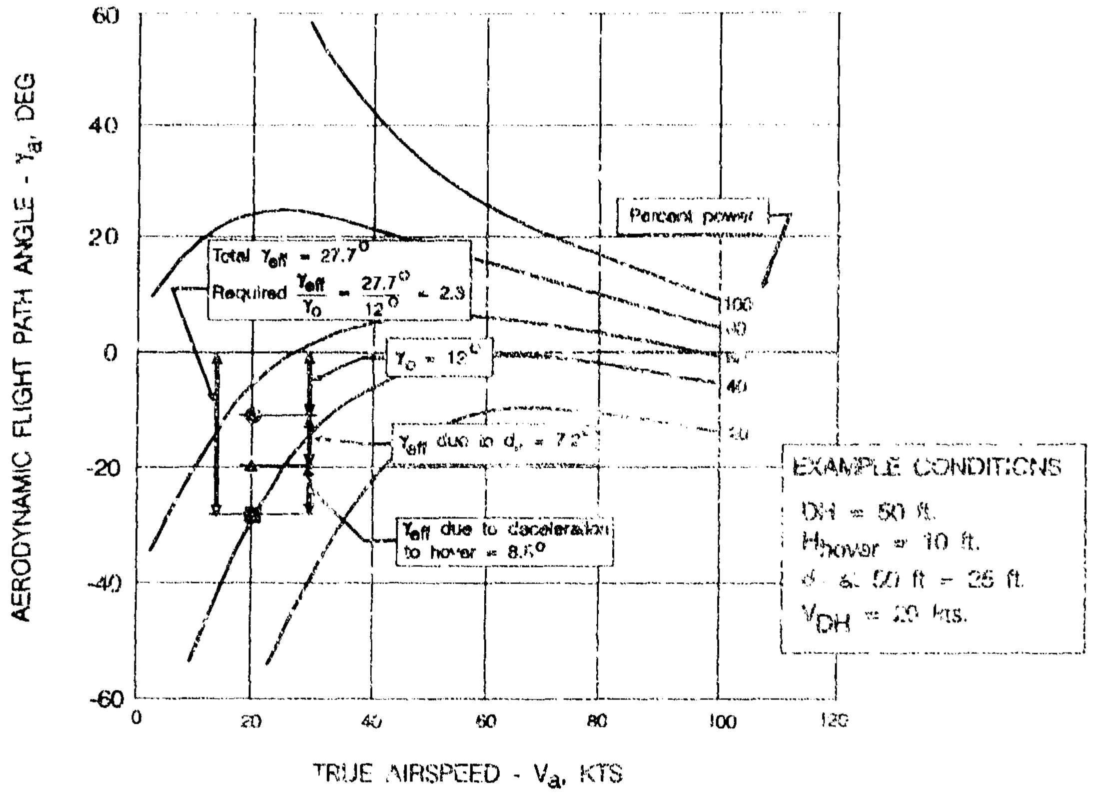

Figure 3. γ - V Curves for XV-15 Tilt Rotor Aircraft With Example Showing Components of γ_{eff}

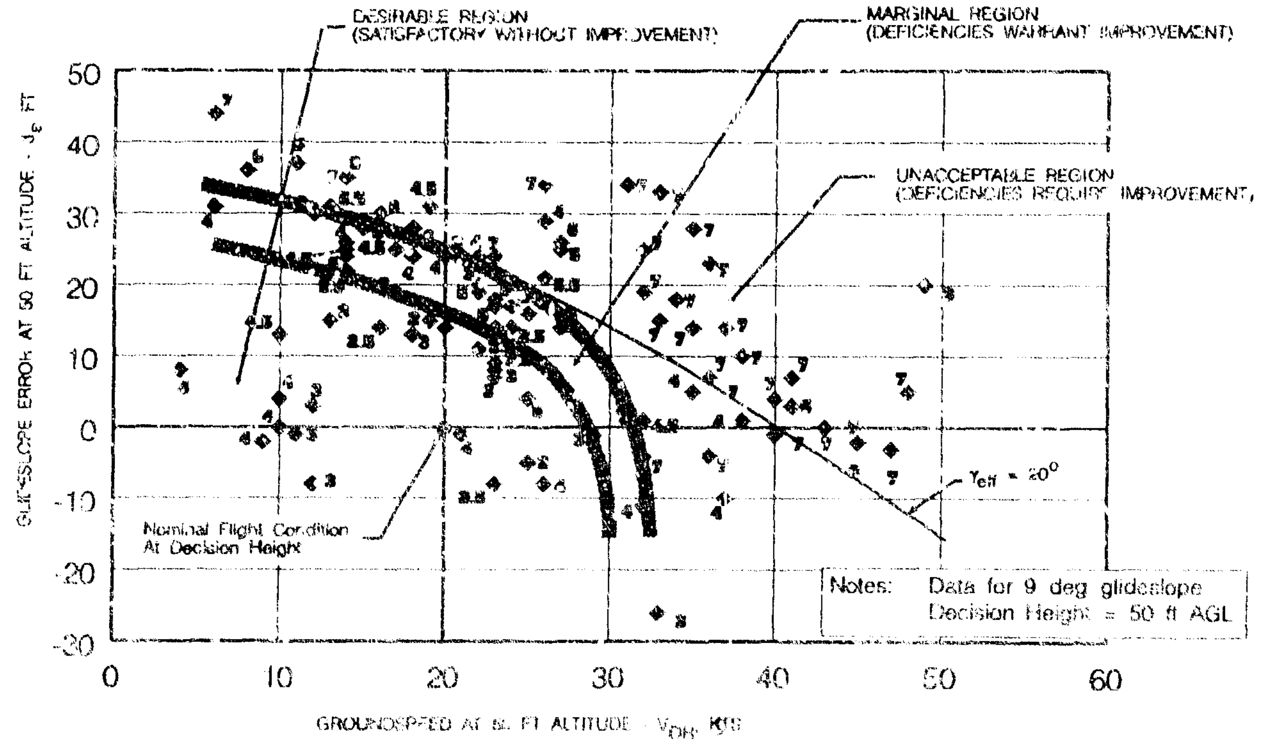

Figure 4. Pilot Ratings as a Function of Glideslope Error and Groundspeed at Decision Height (γ_o = 9°)

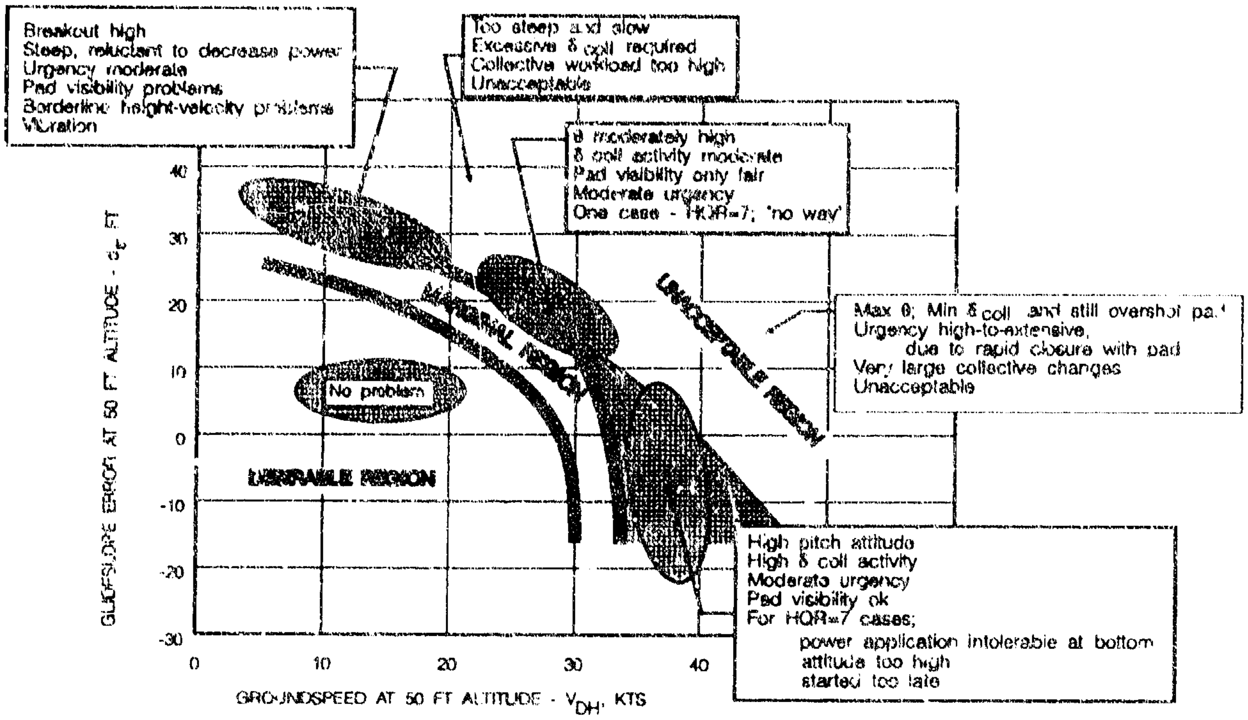

Figure 5. Summary of Pilot Commentary

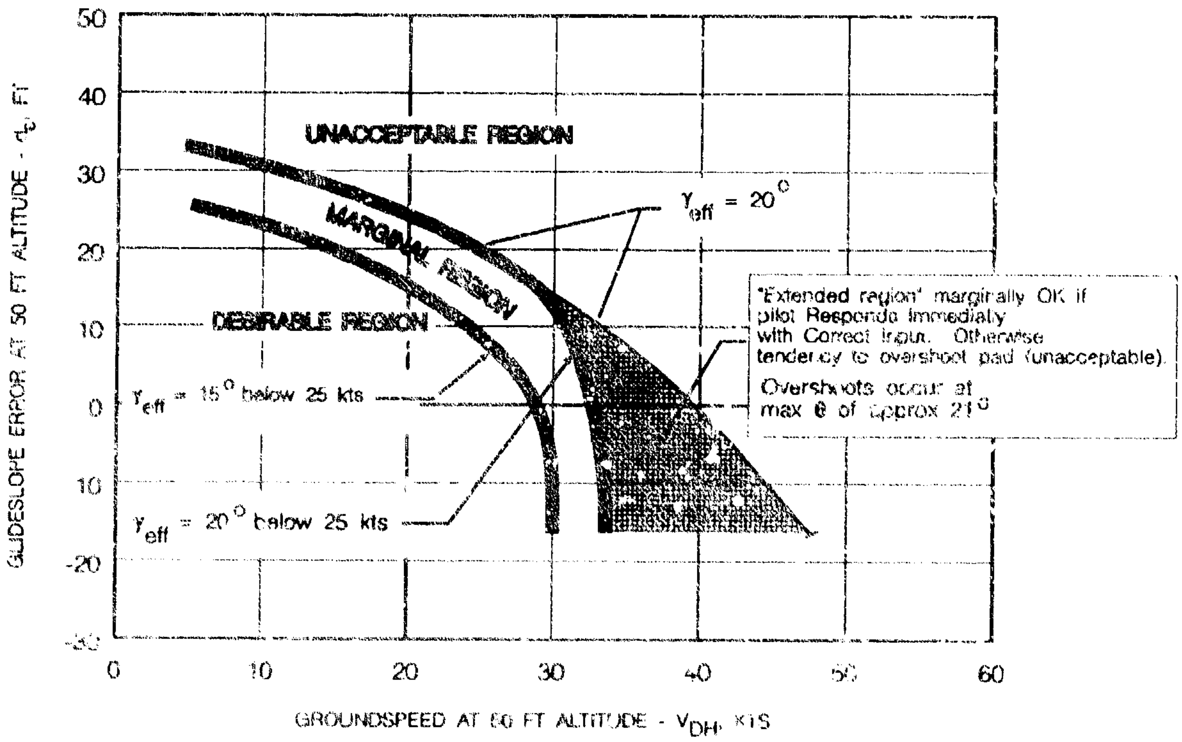

Figure 6. Extended Region for Decision Height Window

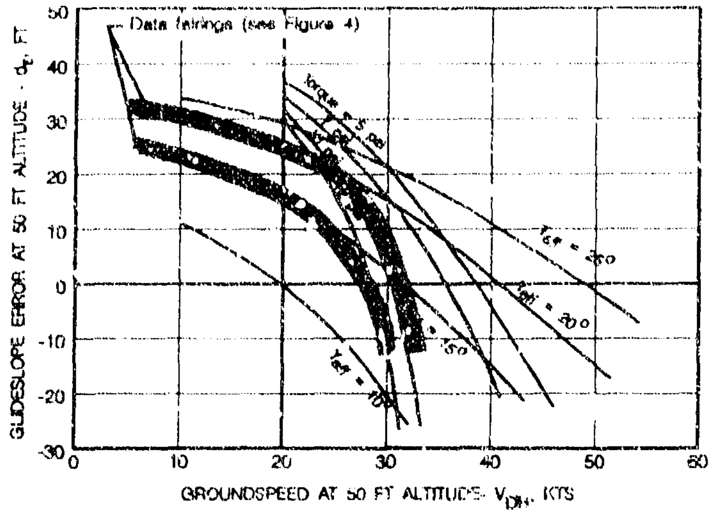

Figure 7. Lines of Constant γ_{eff} and Constant Torque Plotted on Decision Height Window Coordinates ($\gamma_o = 9°$)

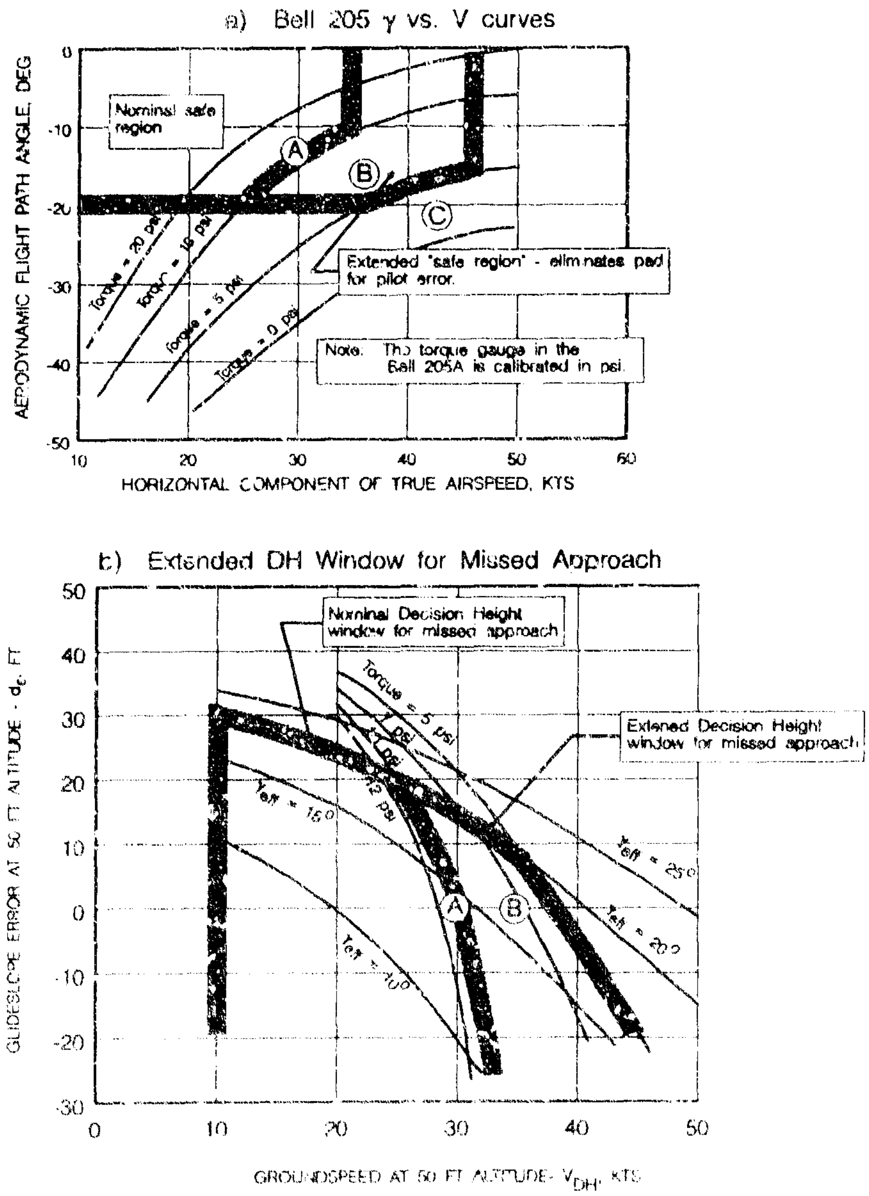

Figure 8. Helicopter Performance Limits Mapped onto d_ϵ vs. V_{DH} to Obtain Nominal and Extended DH Windows for Missed Approach ($\gamma_o = 9°$)

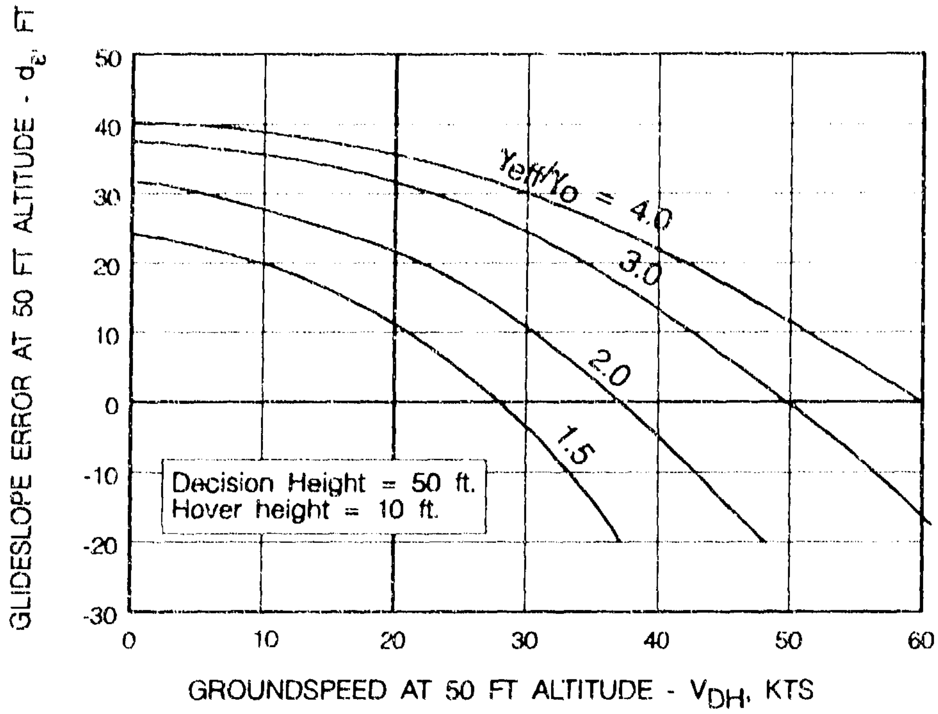

Figure 9 Effect of Systematic Variation of γ_{eff}/γ_o on the Decision-Height Window

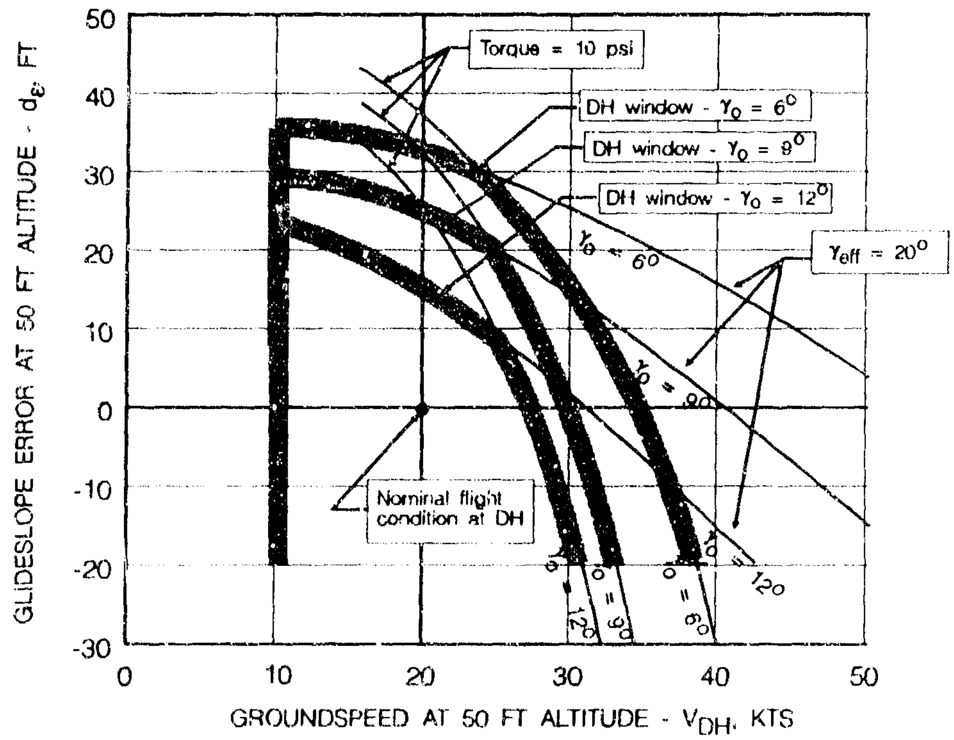

Figure 10. Effect of Glideslope Angle on Bell 205A Missed-Approach Decision-Height Windows

EVALUATION DES LOIS DE PILOTAGE EVOLUEES SUR DAUPHIN N

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1 - INTRODUCTION

Le concept de commandes de vol électriques s'insère dans le cadre du contrôle actif généralisé sur hélicoptère dont le pôle d'intérêt principal se situe dans l'amélioration des qualités de vol des machines actuelles et la diminution de la charge de pilotage en mission. Ces deux thèmes constituent les axes principaux de recherche du développement exploratoire des commandes de vol électriques du DAUPHIN 6001, développement mettant en oeuvre sur hélicoptère des technologies éprouvées à ce jour sur avion.

La philosophie de développement des C.D.V.E. sur hélicoptère s'éloigne quelque peu de celle poursuivie pour les avions dont le but principal était d'augmenter les performances de leurs machines en terme de maniabilité et de manoeuvrabilité. L'augmentation de ces performances se traduisait généralement par une dégradation de la stabilité dynamique de l'avion qui se trouvait alors compensée artificiellement par les C.D.V.E. L'hélicoptère étant naturellement instable et fortement couplé en axes, la démarche poursuivie est à fortiori différente et consiste dans un premier temps à rétablir des qualités de vol acceptables pour l'hélicoptère et à réduire dans un deuxième temps la charge de travail du pilote en proposant à la fois un pilotage plus simple de l'hélicoptère (par objectifs) et une aide dans la surveillance des limitations du domaine de vol, notamment dans des manoeuvres agressives.

Les systèmes actuels (pilotes automatiques) dédiés à ce jour à l'amélioration du pilotage n'avaient pour vocation que de se substituer au pilote pour maintenir l'hélicoptère sur des trajectoires pré-déterminées. Leur architecture n'était pas prévue initialement pour inclure le pilote dans la boucle d'asservissement si ce n'est au travers d'une fonction S.A.S. permettant de garantir une stabilité apparente minimale sur action pilotée. Néanmoins, les performances de ces systèmes sont limitées à ce jour par des contraintes de sécurité, imposant des autorités de fonctionnement trop insuffisantes pour garantir un pilotage performant en manoeuvres rapides.

Disposant d'un meilleur niveau de sécurité, les commandes de vol électriques autorisent des autorités de commande plus importantes et permettant ainsi d'atteindre des niveaux de qualités de vol largement supérieurs, se concrétisant par exemple par un découplage complet des axes de commande en manoeuvre. Par ailleurs, ce type d'architecture de commandes de vol facilite l'introduction de commandes miniaturisées idéales pour l'optimisation ergonomique des cockpits futurs et bien adaptées à la surveillance passive des limitations du domaine de vol (surveillance exigeant de préférence une action sensitive au niveau des manches pilote plutôt qu'une alarme visuelle ou sonore à l'intérieur du cockpit).

Ces considérations justifient en grande partie l'intérêt que l'on doit porter à ce type de commandes de vol notamment pour les hélicoptères de combat futurs.

2 - OBJECTIFS DE L'EXPERIMENTATION DES COMMANDES DE VOL ELECTRIQUES SUR DAUPHIN 6001

Les deux objectifs principaux des commandes de vol électriques se rapportent essentiellement à l'allégement de la charge de travail du pilote en mission et à l'amélioration des qualités de vol de l'hélicoptère.

Elles se proposent :

1 De transformer l'hélicoptère en un appareil stable sur l'ensemble du domaine de vol, y compris sur manoeuvre agressive.

2 De garantir à tout moment un découplage des axes de commande de l'hélicoptère afin de simplifier le pilotage de l'appareil dans son enveloppe opérationnelle.

3 D'augmenter la maniabilité de l'hélicoptère si nécessaire, afin de diminuer les actions de commande exigées aujourd'hui sur les machines à rotor souple.

4 De simplifier l'apprentissage du pilotage de l'hélicoptère en réalisant une adaptation parfaite des objectifs de pilotage aux contraintes opérationnelles de chaque mission (pilotage par objectif).

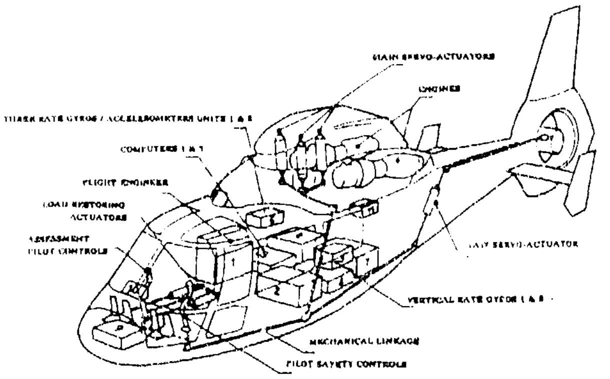
5 De gérer automatiquement les limitations du domaine de vol que le pilote est tenu de surveiller aujourd'hui en mission au travers des indications visuelles et sonores fournies et des consignes stipulées dans son manuel de vol.

L'ensemble de ces considérations constituent les axes de recherche du développement exploratoire des commandes de vol électriques du DAUPHIN 6001. Ces études ont donné lieu à ce jour à des essais en simulation pour définir les lois de pilotage idéales permettant d'atteindre les niveaux de qualités de vol exigés et à la réalisation du démonstrateur volant (DAUPHIN 6001) pour évaluer en vol les performances d'un tel système.

Ce document exposera la méthodologie suivie dans la réalisation des lois de pilotage et détaillera l'architecture du système C.D.V.E. retenue pour ce démonstrateur. Les résultats des premières évaluations seront présentés à la fin de ce document mettant en évidence les performances des lois de pilotage étudiées dans le cadre de cette expérimentation.

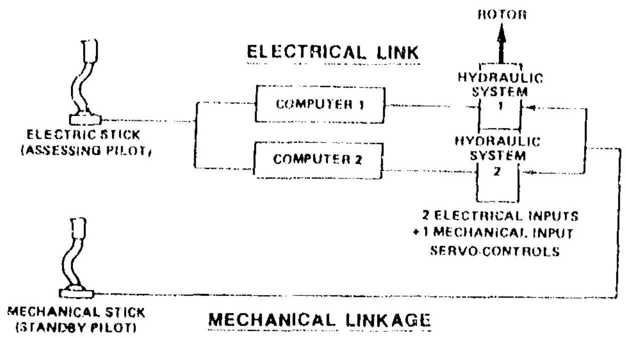
3 - PRESENTATION DE L'ARCHITECTURE DU SYSTEME

L'architecture du système retenue pour le DAUPHIN 6001 est une architecture duplex électrique avec secours mécanique afin de respecter le niveau de sécurité exigé sur ce type de démonstrateur volant. Cette architecture est présentée ci après où sont représentés tous les composants intervenants dans le système C.D.V.E..



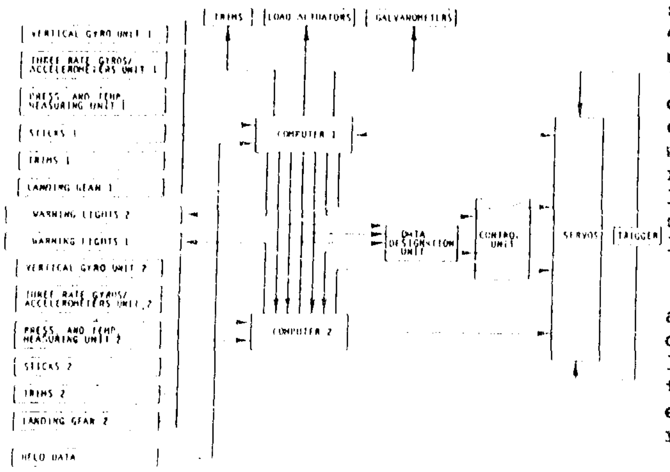
Le secours mécanique est réalisé en place gauche par la timonerie classique de l'hélicoptère que l'on a conservée sur ce poste de commande. Cette contrainte de réalisation a donc nécessité de promouvoir des servocommandes à deux entrées électriques et à une entrée mécanique en lieu et place des servocommandes classiques utilisées à ce jour sur un DAUPHIN série. Pour des problèmes d'encombrement la servocommande arrière a été conservée, la servocommande C.D.V.E. commandant le rotor arrière se trouvant en série avec cette dernière. Le passage en mode secours est assuré à tout instant par une recopie des positions mécaniques équivalentes sur les manches co-pilote, garanti par le lien mécanique entre la timonerie de secours et les tiroirs de commande des servocommandes C.D.V.E. Le retour en mode mécanique peut s'effectuer manuellement soit par une action volontaire du co-pilote au travers de ses boutons de débrayage prévus à cet effet (situés sur ses manches cyclique et collectif), soit par surpassement d'effort du co-pilote exercé sur ces commandes, soit par la manette de déconnexion du système C.D.V.E. située à portée des deux pilotes sur la console centrale. Ce retour en mode mécanique peut également s'effectuer automatiquement sur détection de panne du système C.D.V.E. à partir de la surveillance des paramètres de fonctionnement du système complet introduite tant au niveau des calculateurs embarqués qu'au niveau des servocommandes C.D.V.E.

Les ordres de commandes électriques sont réalisés par deux calculateurs embarqués se surveillant mutuellement. Cette surveillance est réalisée par des échanges d'informations entre ces deux calculateurs pour vérifier la cohérence des données qu'ils reçoivent et des données qu'ils transmettent aux organes de commande. Ces données sont relatives aux informations captées émanant des divers senseurs du système C.D.V.E. (positions de manche, capteurs d'état du mouvement de l'hélicoptère, recopies des positions des servocommandes) digérées en interne par les lois de pilotage des calculateurs. Ces lois élaborent les ordres de pilotage à transmettre aux étages d'entrée des servocommandes en vue de réaliser des déplacements servocommandes compatibles des objectifs de pilotage désirés. Ces ordres sont consolidés en sortie des calculateurs avant transmission aux organes de commande de chaque servocommande.



Les ordres transmis par les calculateurs sont duplex et attaquent les deux étages d'entrée de chaque servocommande. Ces ordres sont surveillés en entrée de chaque servocommande pour vérifier la cohérence des informations émanant de chaque calculateur. Cette surveillance est réalisée par un système électronique disposé en interne de chaque servocommande. Les étages d'entrée sont chargés de réaliser l'asservissement des commandes des deux distributeurs venant déplacer les deux corps de la servocommande.
Les capteurs utilisés dans le système C.D.V.E. sont donc dupliqués, chaque bloc de capteurs venant informer son calculateur associé. Les performances des capteurs utilisés dans cette expérimentation sont tout à fait conventionnelles et font appel à des données de type gyrométriques, gyroscopiques, accélérométriques et barométriques.

L'engagement du mode électrique s'effectue à partir d'un Boîtier de commande (B.D.C.) sur lequel sont disposés les boutons de sélection des divers modes de fonctionnement du système C.D.V.E. Notamment les fonctions "test prévol" permettant de vérifier au sol le bon fonctionnement du système et le "mode synchronisation" sont engagés directement à partir de ce Boîtier de commande. Ce dernier mode est un point de passage obligatoire avant tout engagement du mode électrique de manière à ne pas générer des à-coups sur les servocommandes. Il consiste à synchroniser toute la chaîne de commande de vol électrique (manches électriques, calculateurs) à partir des positions mécaniques imposées avant la phase d'engagement par le pilote de sécurité au travers de la timonerie mécanique de secours. La vérification de la synchronisation des commandes de vol électriques s'effectue à partir d'un voyant disposé directement sur le Boitier de commande autorisant ainsi le passage au mode électrique. Le levier d'engagement du mode électrique est auto-maintenu, une fois l'engagement effectué ; l'auto-maintien étant dégagé après toute détection de panne du système ou toute action volontaire des navigants , dégagement se traduisant par un retour obligatoire en mode mécanique. Pour faciliter l'expérimentation, ce Boitier de commande est doté d'un sélecteur de lois de pilotage permettant de tester au cours du même vol plusieurs type de lois sur les mêmes manceuvres et dans les mêmes conditions de vol.



L'ingénieur navigant dispose d'un poste de désignation de données (P.D.D.) lui permettant d'une part d'injecter dans le système des stimuli calibrés pré-programmés, d'autre part de modifier des gains de lois de pilotage si nécessaire, et de contrôler l'état du système électrique à chaque instant. Ce Boitier permet donc en cas de panne de connaitre l'origine de la panne et d'y remédier si possible.

Les performances des servocommandes ont été augmentées vis à vis de celles montées à ce jour sur les DAUPHIN de série. Leur bande passante se situe à 12 Hz et leur vitesse maximale de déplacement s'élève à 120mm/s permettant un plein débattement en 0,5 s.

Les calculateurs embarqués sont programmés dans des langages différents (PASCAL et LTR) diminuant ainsi les sources d'erreur dans la programmation des logiciels embarqués. Une trame ARINC permet l'échange des informations nécessaires entre les deux calculateurs.

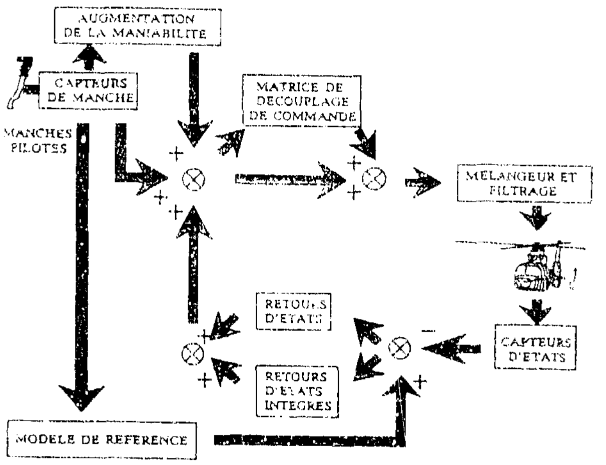
L'ensemble de ces équipements constitue l'architecture du système C.D.V.E. retenu dans le cadre du développement exploratoire du DAUPHIN 6001. Les intervenants dans ce programme ont été SFENA pour les calculateurs embarqués, Boitier de commande et poste de désignation de données, SAMM pour les servocommandes et les vérins de sensation artificielle permettant de gérer les efforts au niveau des manches pilote, et le C.E.V. pour participer à l'évaluation en vol.

4 - ARCHITECTURE DES LOIS DE PILOTAGE

L'architecture des lois de pilotage évaluées à ce jour dans le cadre de ce développement exploratoire est basée sur une technique de suivi de modèle de référence de type implicite. Cette architecture bien adaptée à la réalisation d'un pilotage par objectif permet en outre de garantir une bonne robustesse aux lois de commande ainsi générées.

Le principe consiste à asservir dynamiquement l'hélicoptère à un modèle de référence représentant la dynamique idéale désirée sur action de manche. Ce principe d'asservissement offre une certaine souplesse dans la sélection des objectifs de pilotage désirés, variables selon la mission demandée, sans remettre en cause l'architecture globale des lois de commande. La précision entre le comportement de l'hélicoptère et celui du modèle de référence est garanti par des retours intégraux disposés en interne des lois de pilotage, permettant par ailleurs de gérer les modes long-terme de l'hélicoptère.

L'architecture de ces lois fait donc apparaitre plusieurs blocs fonctionnels dont les rôles consistent à garantir à l'hélicoptère de bonnes performances en terme de découplage d'axes, de stabilité et de maniabilité. Celle-ci peut se représenter sous la forme :



On retrouve sur cette architecture deux blocs de retours d'états (directs et intégrés) dont le rôle est de garantir une bonne stabilité dynamique à l'hélicoptère sur l'ensemble du domaine de vol et de promouvoir des découplages entre axes satisfaisants. Les retours intégrés permettent par ailleurs d'asservir à moyen/long terme l'hélicoptère au modèle de référence en vue de réaliser les objectifs de pilotage exigés par le pilote au travers de ses déplacements de manche. Le bloc "découplage de commande" n'est là que pour faciliter la tâche des lois de pilotage dans le découplage des axes de l'hélicoptère sur actions pilotées.

Le bloc "augmentation de maniabilité" permet de modifier temporairement la consigne instantanée de pilotage en vue d'accroitre si nécessaire la maniabilité apparente de l'hélicoptère.

Le bloc "mélangeur et filtrage" représente le mixage des commandes en sorties des calculateurs en vue de commander les étages d'entrée des quatre servocommandes du DAUPHIN 6001. Le filtrage figurant dans cette architecture correspond à la suppression des fréquences propres de vibration de l'hélicoptère que l'on retrouve directement sur les informations captées nécessaires à l'élaboration des lois de commande. La réinjection de ces fréquences de vibration sur les servocommandes est tout à fait inutile voire même dangereuse en cas de résonance. Le déphasage équivalent induit par l'introduction de ce filtrage peut se ramener à un retard pur équivalent de 50 ms à 2 Hz, c'est à dire dans les fréquences de pilotage traitées par les lois de commande.

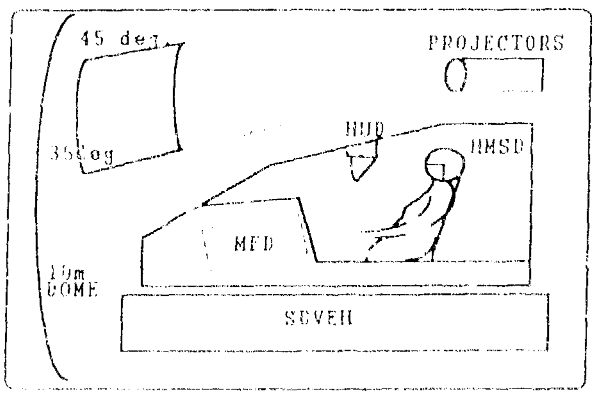
5. RESULTATS DES ESSAIS EN SIMULATION

L'objectif de ces essais était de valider le plus en amont possible les concepts et la définition précise des lois de pilotage des commandes de vol électriques proposées pour l'expérimentation sur le DAUPHIN 6001.

Ainsi, après l intégration des logiciels "temps réel" des CDVE sur le simulateur du Centre d'Essai en Vol à Istres SDVEH (Simulateur de vol d'études pour hélicoptères), une phase d'essai de simulation pilotée a permis de tester dans le plus large spectre de tâche possible (IMC, NOE, transitions) la validité des concepts proposés.

5.1. Présentation des moyens d'essais

L'étude a été menée sur une cabine fixe de simulation. Les commandes collectif et palonnier étaient reliées à des dispositifs électro-hydrauliques programmables permettant de faire varier les paramètres tels que seuil, raideur, frottements sec, visqueux. La cabine, équipée d'une instrumentation classique étaient installée à l'intérieur d'une sphère où était projeté sur un champ de 60° de diagonal, un paysage généré à partir d'une maquette de terrain. Le pilote avait également à sa disposition un viseur tête haute et un écran tête basse sur lesquels étaient projetés une symbologie de pilotage (voir planche suivante).



Le modèle de base de l'hélicoptère était le modèle S80 développé par le CEV avec une base de données du type hélicoptère DAUPHIN.

Dans cette étude, il est apparu intéressant de retenir des scénarios de vol précis qui étaient répétés par le pilote pour chaque configuration. Une configuration de vol étant caractérisée par différents types: de loi de pilotage, de minimanche, de visualisation (Viseur tête haute, écran tête basse, instrument de bord), de stabilisation machine.

Dès lors, le pilote prononçait, par configuration, un jugement sur les qualités de vol de l'hélicoptère. Les éléments de cette évaluation étant les commentaires du pilote ainsi que l'attribution d'une note (type échelle de Cooper-Harper).

De plus, les essais étaient enregistrés sur une bande magnétique pour être rejoués par la suite en temps différé à l'Aérospatiale afin de mettre en évidence tous les problèmes rencontrés pendant la simulation "temps réel".

Plusieurs lois de pilotage ont été proposées en évaluation. La planche suivante montre les principes ou plutôt les objectifs de pilotage proposés par axe de pilotage.

Tous les objectifs pouvaient être "mixés" entre eux et fournir ainsi de nombreuses configurations de loi de pilotage.

CONTROL LAWS IN TEST

LONGITUDINAL CONTROL STICK
velocity, pitch attitude, pitch rate

LATERAL CONTROL STICK
bank attitude, roll rate

COLLECTIVE CONTROL STICK
vertical speed, load factor, power

FOOT PEDALS
yaw rate in low speed, side-slip in cruise speed

5.2. Présentation des résultats d'essais

Les objectifs de pilotage retenus dans un premier temps pour notre évaluation en vol sont restés classiques et se rapprochent du comportementnaturel instantané d'un hélicoptère. Ce choix semble se justifier par le besoin de pouvoir passer d'une machine conventionnelle à ce nouveau type de machine sans trop de discontinuité dans la philosophie de pilotage. Cette approche industrielle justifie ce choix initial des objectifs de pilotage en début d'expérimentation, pilotage qui englobe l'ensemble des missions que remplit aujourd'hui un hélicoptère, sans prétendre afficher des qualités de vol optimales sur chacune d'elle.

Le pilotage proposé et réalisé par le modèle de référence consiste donc à:

- piloter la vitesse angulaire de tangage sur la commande cyclique longitudinal,
- piloter la vitesse angulaire de roulis sur la commande cyclique latérale en garantissant le virage coordonné en croisière,
- piloter directement le pas général sur le manche collectif,
- piloter la vitesse angulaire de lacet en basses vitesses et le facteur de charge latéral en croisière sur le palonnier.

Ce pilotage se rapproche effectivement du comportement naturel d'un hélicoptère à court terme, tout en offrant en plus une stabilité et un découplage des axes bien plus performants.

Il faut cependant noter que la loi de pilotage retenue ne permet pas d'effectuer des manoeuvres telles que le poser et le décollage sur pente ou la roulage.
D'une manière générale il manquait au pilote dans ces configurations le retour d'information des positions de gouvernes.

6. RESULTATS DES ESSAIS EN VOL

D'une façon très générale, ces essais en vol ont consisté à étudier la faisabilité et l'intérêt des CDVE sur hélicoptère. Dans ce contexte, les essais en vol comportaient trois phases d'essais correspondant aux principaux objectifs recherchés pour cette évaluation:

1. la mise au point d'une loi directe, représentant un transfert unité entre les manches de pilotage et les gouvernes. Cette loi a permis de valider le système complet des commandes de vol électriques au niveau fonctionnel avant toute expérimentation de lois évolues,
2. l'évaluation de la loi de pilotage évoluée, déterminée durant la phase d'essais sur simulateur. Les performances de cette loi de pilotage pouvant être comparées vis à vis d'un pilotage conventionnel avec la loi directe,
3. l'évaluation du pilotage de l'hélicoptère par l'intermédiaire d'un minimanche dans la configuration de pilotage testée sur simulateur, c'est à dire 2+1+1.

Durant les phases 2 et 3, on s'est intéressé principalement à relever l'apport des commandes de vol électriques sur hélicoptères en terme:
- d'amélioration des qualités de vol,
- de diminution de la charge de travail du pilote.

6.1. Présentation des moyens d'essais

Ces essais ont été l'occasion pour les essais en vol de se familiariser avec les techniques modernes dans le domaine du contrôle automatique généralisé du vol. Ceci concernait bien sûr l'installation de mesure embarquée mais également les méthodes d'analyse des qualités de vol. Ces dernières ont permis d'expérimenter les moyens d'essais, le vocabulaire et les concepts nouveaux qui ont été développés par les évaluateurs américains et qui sont en train de devenir le moyen universellement reconnu pour tenter de quantifier les qualités de vol.

Dans ce contexte, ont été pratiqués, entre autres, l'usage de l'échelle de COOPER HARPER pour noter la possibilité d'accomplir une tâche qui peut se résumer à un élément de mission ainsi que des essais spécifiques permettant au Bureau d'Etudes de positionner le DAUPHIN vis à vis des propositions de nouvelles normes QDV.

6.2. Présentation des résultats d'essais

L'avancement actuel du programme permet de fournir les résultats relatifs aux phases d'essai en vol n° 1 et 2

Sept pilotes ont à ce jour effectué environ 30 heures de vol qui ont permis d'évaluer la loi de pilotage proposée pour l'expérimentation.

6.2.1. Analyse des qualités de vol

L'avis général des pilotes se résume par un comportement de la loi en termes de découplages d'axes et de niveau de stabilité atteint, visibles notamment en zone de turbulences. La mise au point de la loi de pilotage n'a suscité que très peu d'heures de vol, mettant ainsi en évidence le niveau de robustesse d'une telle architecture de loi.
L'apprentissage de cette loi s'est avérée être instantanée, ne nécessitant que très peu d'heures de vol d'accoutumance, justifiant ainsi le choix des objectifs de pilotage retenus pour cette première expérimentation.
Les qualités de vol apportés par la loi sont conformes aux résultats de simulation et correspondent au cahier des charges fixés initialement dans notre expérimentation sur les niveaux de stabilité et de découplager à atteindre.

La maniabilité en roulis a semblé à priori trop importante à grandes vitesses. Cette sur-maniabilité se traduisant par un pompage piloté rencontré dans la tenue d'une inclinaison donnée en turbulences, est due en réalité au seuil trop important existant sur la commande de manche de l'axe considéré. Ceseuil est relatif à une mauvaise connaissance de la position neutre du manche roulis due aux bruits de mesure sur cet axe et au jeu mécanique du manche.

Cet inconvenient de la chaîne de commande cyclique sera résolu par l'introduction d'une commande de type minimanche sur cet axe, commande attendue à ce jour par les pilotes qui ont participé à cette évaluation car bien adaptée à ce type de loi de pilotage expérimenté.

Dans l'ensemble, la qualité des découplages d'axe et le niveau de stabilité de la machine notamment dans les fortes turbulences ont semblé tout à fait satisfaisants. Les transitions entre la croisière et les basses vitesses n'ont pas soulevé de problèmes spécifiques dans les deux sens de traversée.

Les résultats de l'analyse quantitative des qualités de vol sont fournis en annexe.

Compte tenu de ces résultats et en se référant aux critères QDV proposés par la proposition de nouvelles normes ADS.33.C, nous constatons que le DAUPHIN muni de cette loi de pilotage évoluée se place à la frontière du niveau 1 sur les trois axes. L'objectif initial n'ayant pas été d'optimiser cette maniabilité, ces résultats sont satisfaisants dans l'ensemble. Il restera à améliorer ultérieurement la bande passante sur les trois axes afin de disposer de plus de degrés de liberté dans le réglage de cette maniabilité.

6.2.2. Le traitement des limitations

La necessité d'une gestion appropriée des limitations du domaine de vol, déjà apparue en simulation, s'est clairement imposée lors de l'évaluation en vol de la loi de pilotage évoluée.
Leurs implications sur la sécurité du vol, la charge de travail du pilote et l'aspect conformité aux règlements en vigueur ont conduit à considérer le traitement des limitations, plus généralement, comme un point crucial et incontournable pour le développement futur des CDVE sur hélicoptères.

En effet, il faut souligner l'aspect insidieux que peut revêtir la perte de contrôle à bord d'un aéronef équipé d'un système CDVE performant et possédant une pleine autorité sur toutes les commandes. Le pilote qui sera habitué en vol "normal" à un certain confort de pilotage (absence de couplage, maniabilité optimale,...) ne saura pas forcément quelle attitude adoptée (sur quelles commandes agir ?) à l'approche d'une situation qui pourrait devenir catastrophique et vers laquelle il est entraîné irrémédiablement par l'objectif de pilotage qu'il a assigné au système.

Le traitement des limitations associèes aux CDVE doit conduire certes à l'étude d'une solution technique mais également à une réflexion des services certificateurs sur un cadre réglementaire nouveau.
Certaines recommandations des services officiels français telles que les deux principes de base suivants:
- le pilote doit avoir à sa disposition une "restitution" de la marge de commande de ses gouvernes à l'approche de leurs butées,
- cette marge doit être suffisamment dimensionnée pour que le pilote soit en mesure de contrer une pertubation qui interviendrait en limite de marge,

ont conduit à des études spécifiques menées par l'Aérospatiale dans le cadre des essais en vol sur le DAUPHIN CDVE.

Les essais à venir sur le DAUPHIN permettront l'application d'autres principes concernant entre autres les limitations du domaine de vol (vitesse air, facteur de charge) et les limitations moteurs (PMC, PMD, PMU, PIU).

7. CONCLUSION

Les commandes de vol électriques, prometteuses par les degrés de liberté qu'elles offrent, ouvrent une voie intéressante dans la réalisation de qualités de vol optimales pour nos hélicoptères futurs. Le développement exploratoire mené dans le cadre du DAUPHIN 6001 a été dédié à la recherche de nouveaux concepts de pilotage en vue de faciliter la tâche du pilote dans la maîtrise de son appareil. Cette approche, pour laquelle le caractère industriel a prévalu dans un premier temps, s'est concrétisée par des résultats assez satisfaisants sur le plan de la robustesse des lois de commande ainsi générées (mise au point rapide). L'harchitecture duplex du système CDVE retenue pour cette expérimentation est loin d'être représentative de celle d'un appareil de type NH 90. Néanmoins cette expérimentation est riche d'enseignement sur les difficultés que l'on peut rencontrer dans la réalisation de ce nouveau type de commandes de vol.

Cette première expérimentation assez prometteuse permet l'évaluation d'un nouveau type de commande plus ergonomique et mieux adapté aux contraintes de place des cockpits d'hélicoptère. Ce nouveau type de manche sera expérimenté dans les tous prochains jours sur le DAUPHIN 6001 afin de juger les performances de pilotage obtenues à partir de ces commandes toutes intégrées.

Il reste à approfondir certains concepts comme la gestion des limitations du domaine de vol qui représentent aujourd'hui une charge de travail relativement importante pour le pilote et qui peuvent limiter parfois l'agressivité de ses manoeuvres. L'évaluation de concepts de pilotage plus traditionnels sera également réalisée dans le cadre de ce développement exploratoire afin de respecter la logique de développement des lois du NH 90; celle-ci consistant à ne considérer dans un premier temps que des concepts de pilotage classiques et familiers pour les pilotes avant de franchir le pas dans un deuxième temps vers des concepts plus futuristes comme ceux évalués à ce jour dans le cadre de ce développement exploratoire.

ANNEXE

ANALYSE QUANTITATIVE DES QUALITÉS DE VOL

Analyse fréquentielle des Q.D.V.

L'objet de cette étude est de déterminer les fonctions de transfert équivalentes entre manche pilote et vitesses angulaires générées sur action de manche. L'analyse de ces dernières permet de connaître les performances des lois de pilotage en terme de temps de retard et bandes passantes équivalentes, afin de se positionner vis à vis des propositions de nouvelles normes Q.D.V. proposées à ce jour dans l'ADS.33.C.

Pour cette détermination, des sollicitations de type sinusoïdal à fréquences variables réalisées manuellement par le pilote permettent d'effectuer une analyse fréquentielle cohérente des réponses temporelles de l'hélicoptère ainsi générées, et d'en déduire consécutivement l'allure des fonctions de transfert désirées. Ces fonctions de transfert sont présentées en annexe (figures n°1 à 5) et permettent d'en déduire les performances de la loi de pilotage en terme de bande passante équivalente et de temps de retard d'injection de la commande sur les trois axes (roulis, tangage, lacet). L'analyse de l'axe collectif n'a pas été faite dans la mesure où cet axe reste conventionnel par rapport à une commande mécanique.

L'analyse spectrale des signaux enregistrés sur les trois axes permet de déterminer les coefficients des diverses fonctions de transfert recherchées mises sous la forme canonique suivante:

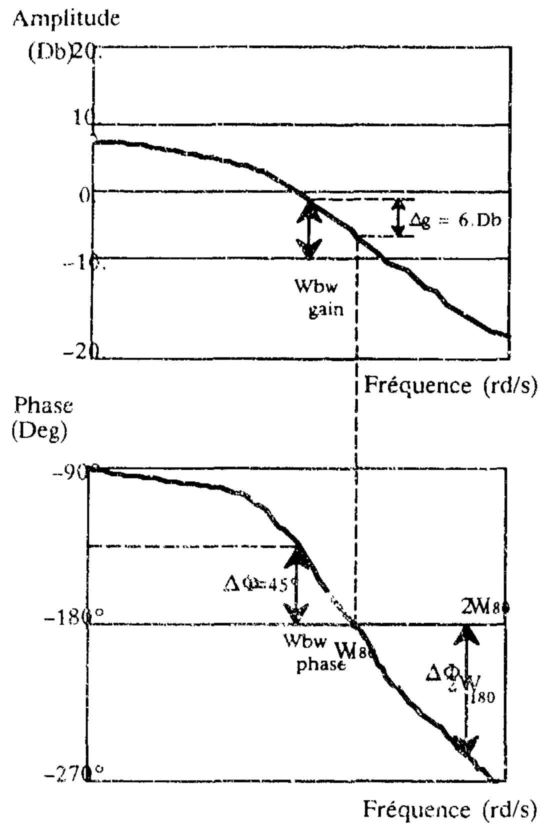
$$\boxed{\frac{\text{Assiette}}{\text{manche}} = \frac{\exp(-\tau.s).w_n^2}{s.(w_n^2 + 2\rho\ w_n .s + s^2)}}$$

A partir de ces fonctions de transfert et de leur représentation sous forme de Bode, il est possible de connaître le retard pur équivalent τ ainsi que la bande passante équivalente Wbw de l'hélicoptère sur chaque axe.

Le retard τ est obtenu directement par la formule :

$$\boxed{\tau = \frac{\Delta\Phi_{2W_{180}}}{57.3\ (2W_{180})}}$$

La pulsation W_{180} correspond à la pulsation de coupure relative à un déphasage de 180° sur le diagramme de Bode et $\Delta\Phi_{2W180}$ correspond à l'écart de déphasage entre celui obtenu à la pulsation double de W_{180} et 180°.



Dans notre cas de pilotage par objectifs (vitesses angulaires), la bande passante de chaque axe Wbw est donnée par le minimum de Wbw phase et de Wbw gain obtenu pour chaque fonction de transfert.

Les résultats obtenus pour les différents axes sur la loi par objectif sont présentés en annexe (figures n°1 à 5). Ceux-ci correspondent aux fonctions de transfert des differents axes identifiées pour deux points de vol relatifs à Vi < 40.kts et Vi = 80.kts. Sont présentés ci-dessous les valeurs numériques obtenues pour le retard pur équivalent de chaque axe ainsi que la bande passante associée.

Fonction de transfert pour Vi < 40. Kts

Axe de roulis

Le retard équivalent de cet axe est égal à

$\boxed{\tau = 0.16 \text{ sec}}$

La bande passante Wbw de cet axe est égale à

$\boxed{\text{Wbw} = 2.6 \text{ rd/s}}$

Axe de tangage

Le retard équivalent de cet axe est égal à

$\boxed{\tau = 0.17 \text{ sec.}}$

La bande passante Wbw de cet axe est égale à

$\boxed{\text{Wbw} = 1.9 \text{ rd/s}}$

Axe de lacet

Le retard équivalent de cet axe est égal à

$\boxed{\tau = 0.15 \text{ sec}}$

La bande passante Wbw de cet axe est égale à

$\boxed{Wbw = 1.9 \text{ rd/s}}$

Fonction de transfert pour Vi = 80. Kts

Axe de roulis

Le retard équivalent de cet axe est égal à

$\boxed{\tau = 0.15 \text{ sec}}$

La bande passante Wbw de cet axe est égale à

$\boxed{Wbw = 1.9 \text{ rd/s}}$

Axe de tangage

Le retard équivalent de cet axe est égal à

$\boxed{\tau = 0.19 \text{ sec}}$

La bande passante Wbw de cet axe est égale à

$\boxed{Wbw = 2.6 \text{ rd/s}}$

Axe de lacet

Le retard équivalent de cet axe est égal à

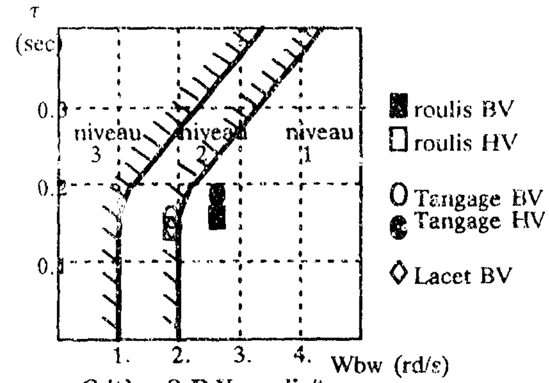
$\boxed{\tau = 0.14 \text{ sec}}$

La bande passante Wbw de cet axe est égale à

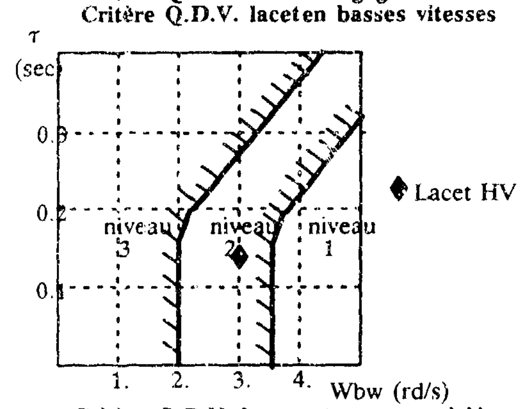
$\boxed{Wbw = 2.9 \text{ rd/s}}$

Compte tenu de ces résultats et en se référant aux critères Q.D.V. proposés par la proposition de nouvelles normes ADS.33.C, nous constatons que le DAUPHIN 6001 muni de cette loi de pilotage évoluée se place à la frontière du niveau 1 sur les trois axes. L'objectif initial n'ayant pas été d'optimiser cette maniabilité, ces résultats sont satisfaisants dans l'ensemble. Il restera à améliorer ultérieurement la bande passante sur les trois axes afin de disposer de plus de degrés de liberté dans le réglage de cette maniabilité.

Ces résultats sont synthétisés par les diagrammes exposés ci-après représentant les critères retenus pour notre étude :



Critère Q.D.V. roulis/tangage
Critère Q.D.V. laceten basses vitesses



Critère Q.D.V. lacet retenu en croisiére

L'analyse de la stabilité de l'hélicoptère en boucle fermée nous permet de représenter les modes obtenus en boucle fermée avec la loi de pilotage par objectif (voir figures n°8 et n°9). On remarquera le niveau de robustesse de la loi de commande lié à la concentration des modes dans le cône d'amortissement relatif à $\rho = 0.6$ sur l'ensemble du domaine de vol (Les lieux d'EVANS présentés en annexe sont relatifs à des balayages en vitesses partant du stationnaire et allant jusqu'à la V.N.E. du DAUPHIN 6001). Pour comparaison sont représentés également les modes propres de l'hélicoptère obtenus en loi directe (voir figures n°6 et n°7) : On constatera pour cela la divergence des modes phugoïdes visible en boucle ouverte et leur convergence obtenue en boucle fermée).

Analyse temporelle des Q.D.V.

Analyse des découplages d'axe

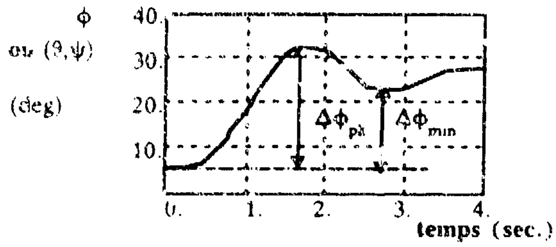
Cette analyse a consisté à vérifier la qualité des découplages entre axes suite à des sollicitations effectuées sur chacun d'eux. Les résultats obtenus en vol sont présentés en annexe (figures n°10 à 13), affichant les diverses sollicitations effectuées sur les différents axes et les évolutions consécutives de l'hélicoptère et ceci pour plusieurs cas de vol.

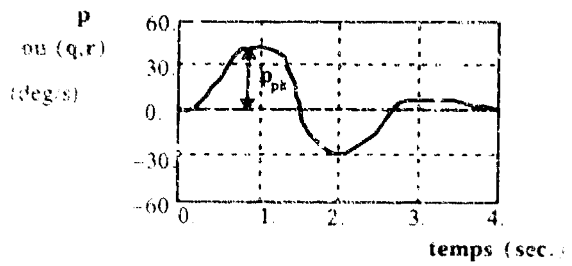
On constatera le bon comportement des découplages entre axes sur l'ensemble du domaine de vol notamment à moyen/long terme, le court terme n'étant pas totalement maitrisé par non-disposition d'informations suffisantes au niveau des lois de pilotage. Pour améliorer ce dernier, disposer des valeurs dérivées des vitesses angulaires serait un moyen suffisant pour contrer les évolutions instantanées non maitrisées à ce jour, informations se ramenant en réalité à connaître les angles de battement du rotor.

Etude de la maniabilité

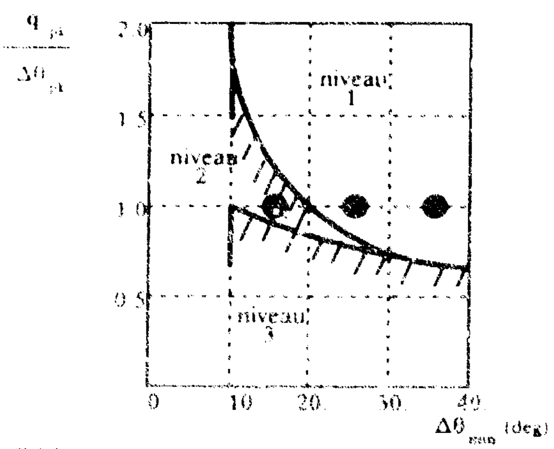
Le deuxième aspect de l'analyse temporelle a consisté à regarder les pics en vitesses angulaires obtenus sur prise d'assiette. Cette analyse a été faite sur les trois axes roulis/tangage/lacet. Les résultats obtenus sur ces trois axes sont présentés sur les schémas ci-dessous conformes à la proposition de norme A.D.S.33.C. Les stimuli injectés sur chacun des axes ont été raisonnables dans cette première expérimentation sans vouloir tenter d'atteindre des variations d'assiette trop rapides vues du pilote. Ces premiers résultats montrent que le niveau 2 de la norme ADS.33C est atteint. Il restera à approfondir cette maniabilité par des actions de manche plus rapides en temps et plus importantes en amplitude (Les résultats visualisés dans les diagrammes de maniabilité exposés ci-après correspondent à des actions de manche de 10% à 30% sur 1 sec.)

En prenant pour définition $\Delta\theta_{pk}$, $\Delta\phi_{pk}$, $\Delta\psi_{pk}$ et q_{pk}, p_{pk}, r_{pk} :

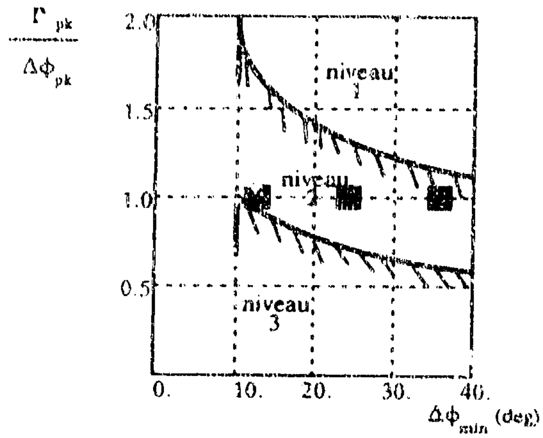




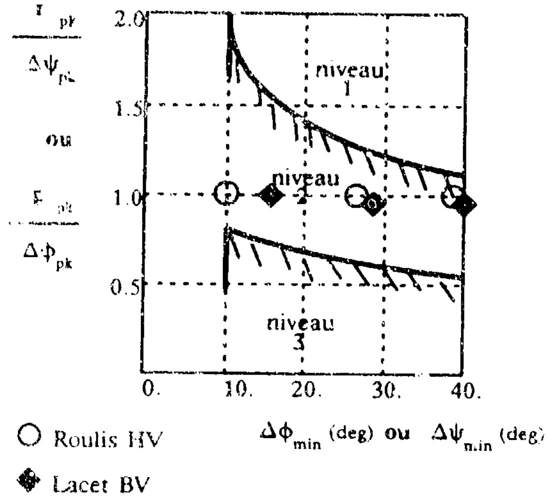
Les résultats obtenus sur les différents axes ont été les suivants :



Critère de maniabilité en tangage (basses vitesses)



Critère de maniabilité en roulis (basses vitesses)



Critère de maniabilité en lacet (b.v.) et roulis (h.v.)

FIGURES

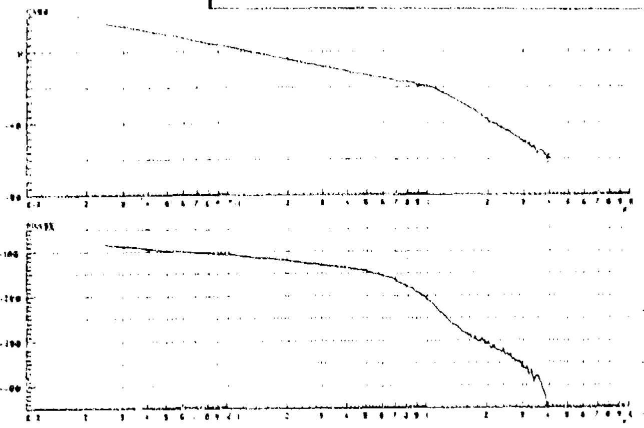


Figure 1 Pitch Transfer θ/θ_{sm} (vt = 80 kts)

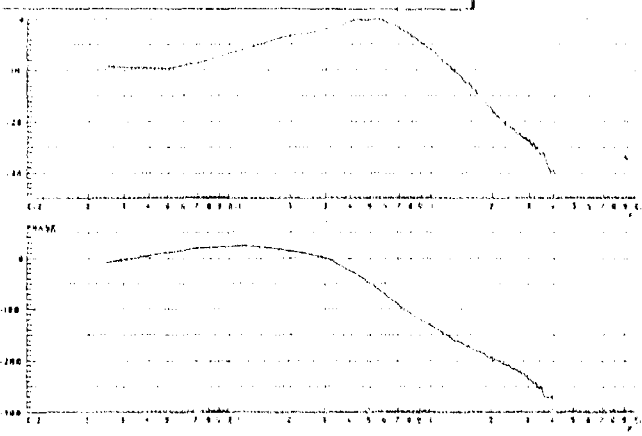


Figure 5 Yaw Transfer r/θ_{rm} (vt = 80 kts)

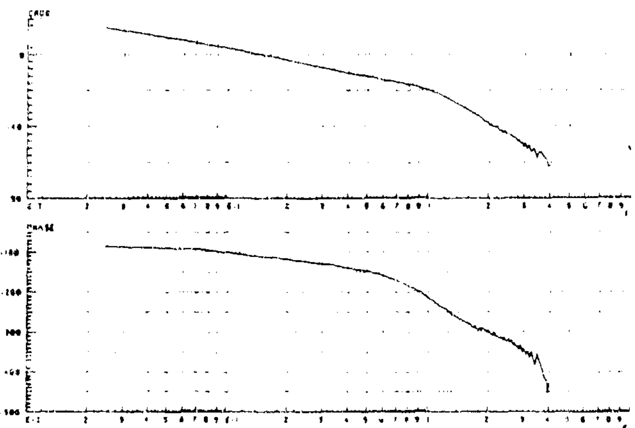


Figure 2 Pitch Transfer θ/θ_{sm} (Hover)

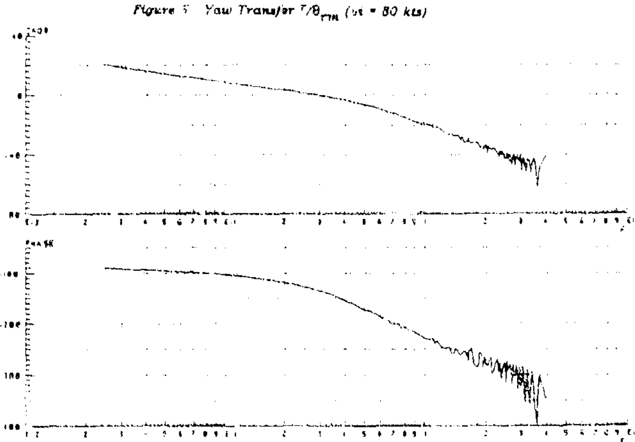


Figure 6 Yaw Transfer ψ/θ_{rm} (Hover)

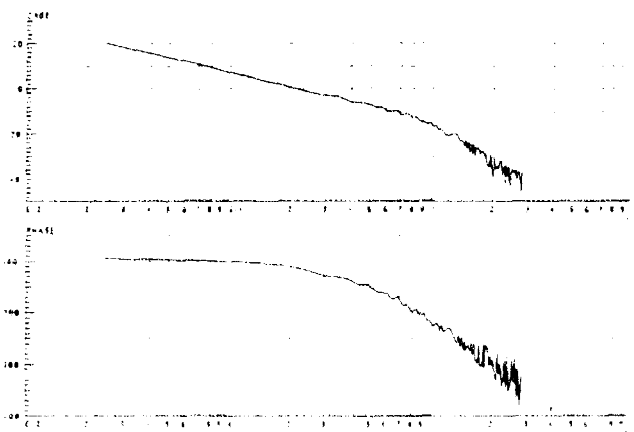


Figure 3 Roll Transfer φ/θ_{1m} (vt = 80 kts)

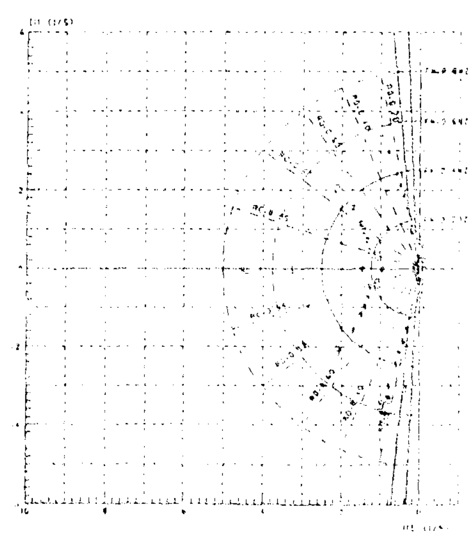


Figure 7 . Open Loop Stability (50 kts < u < 175 kts)

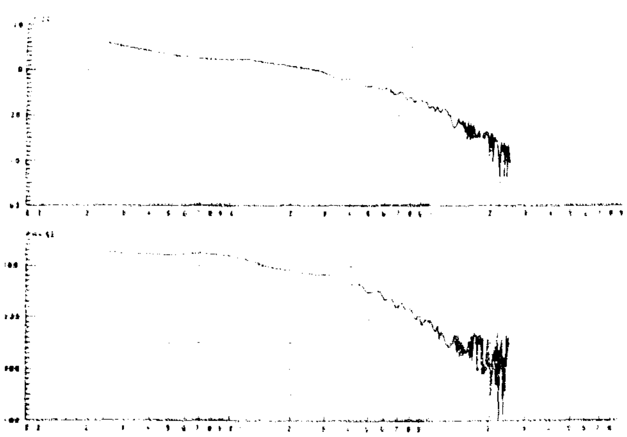


Figure 4 Roll Transfer φ/θ_{1c} (Hover)

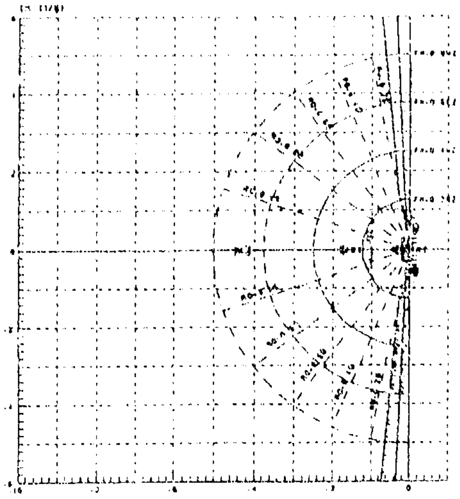


Figure 6 : Open Loop Stability (O kts < vi < 40 kts)

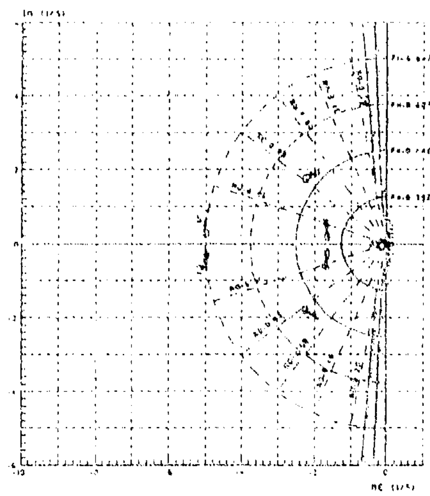


Figure 8 : Closed Loop Stability (O kts < vi < 40 kts)

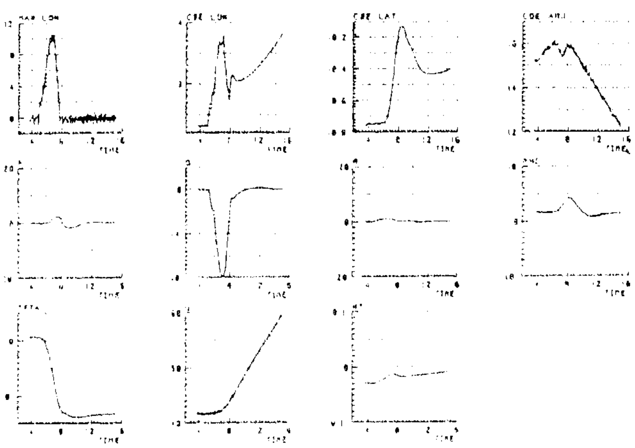


Figure 10 : Step On Pitch Axis (vi = 80 kts)

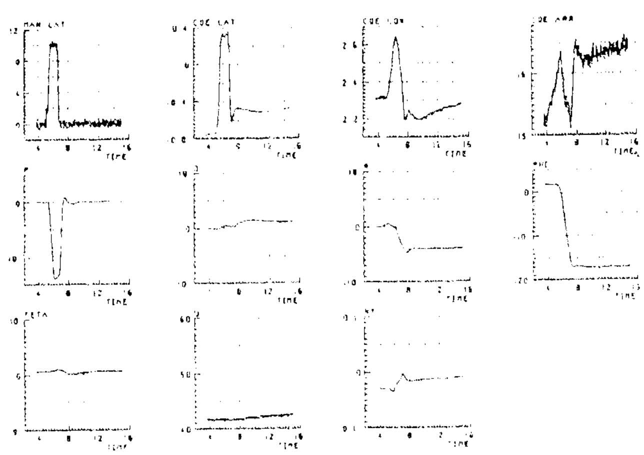


Figure 12 : Step On Roll Axis (vi = 80 kts)

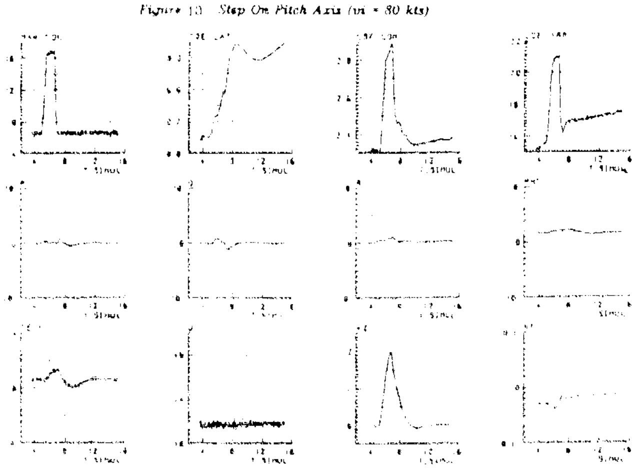


Figure 11 : Step On Collective Axis (vi = 80 kts)

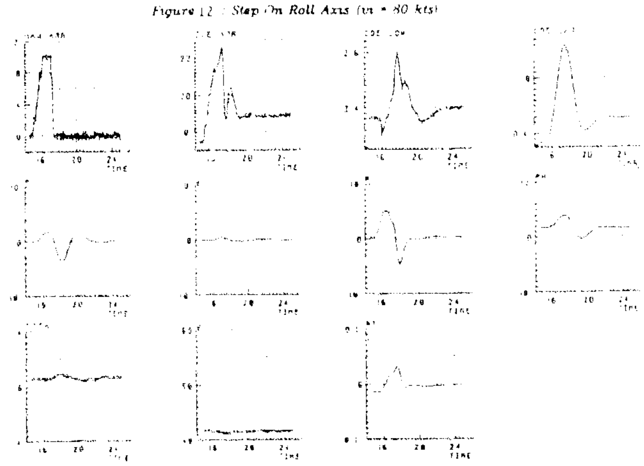


Figure 13 : Step on Yaw Axis (vi = 80 kts)

Integration of Handling Quality Aspects into the Aerodynamic Design of Modern Unstable Fighters

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1. Summary

The principal flightmechanical duties during the several development phases of any aircraft have always been concerned with the vital aspects of "Flying Qualities" in a general sense. The provision of Stability and Control, Maneuverability, sufficient trim capabilities and acceptable control forces throughout the usable flight envelope has been forming the solid bases for the design and sizing of horizontal and vertical tails, ailerons, rudder and elevator power. For stable unaugmented fighter aircraft, where pilot inputs are almost directly linked with the movement of a (single) control surface, the criteria, which lead to a proper sizing of the stabilizers and controllers, can be calculated from dynamic and static Handling Quality requirements of MIL-Spec without detour. - The flightmechanical design of highly augmented aircraft with unstable basic characteristics however is no longer straight forward. The realization of superior Handling Qualities, Maneuverability and Agility up to high angles of attack is mixed with other design goals like optimum performance and/or the observance of structural limitations. The task to integrate the requirements from different disciplines is focused on a single "Black Box", that is to say on the Flight Control System. The various subtasks which have to be managed by the FCS may roughly be split into two parts: One is related to the "Control of Steady-State and Instantaneous Maneuvers", which were formerly managed by the pilot himself with his direct links to the control surfaces of the aircraft. The other part covers the vital aspects of "Stabilization of the Aircraft" which for conventional unaugmented configurations used to be guaranteed by the stabilizing surfaces without artificial help. Now both tasks have to be settled via the Flight Control Computers within the narrow frame of realistic hardware and the necessary scheduling of aerodynamic controls required for increased performance and load alleviation.

In order to maintain the chance to fulfil the challenging requirements for optimum performance <u>and</u> superior Handling Qualities it is necessary to define a set of flightmechanical criteria which translate the most important aspects derived from Handling, Agility and Safety points of view into aerodynamic requirements. These criteria have to generate the necessary link between the disciplines Control Law Design, Flightmechancis and Aerodynamcis within the Pre-Development phases of modern fighter aircraft.

The paper in hand illustrates that the "Time to Double Amplitude T_2" of the basic aircraft may act as key characteristic which describes the problems associated with stabilization in Pitch. If related to the available Pitch Control Power and Control Power Buildup Rate limits for "Minimum permissible T_2" may be established. These limits are easily convertible into aerodynamic requirements and therefore applicable within the early design phases of modern fighter design. - At high angles of attack additional pitch down control power is needed to cancel the effects of inertial coupling due to roll maneuvers. The acceleration to be provided is dependent on angle of attack and attainable rollrate around the velocity vector.

As far as the lateral/directional axes are concerned the FCS is mainly used for stability augmentation and for optimum coordination of the control surfaces. Therefore the flightmechanical design still aims at stable basic characteristics. To achieve this design goal $C_{n\beta dyn}$ is very often the only parameter which is used for optimization during windtunnel testing. The discussion at the end of section 4 shows that especially at high angles of attack stable <u>static</u> derivatives are not sufficient to maintain dynamic stability. The dynamic derivatives "roll and yaw damping" have to be considered as well and at least negative $C_{\ell p}$'s and C_{nr}'s are necessary to complete the listing of desired characteristics. - Agility around the velocity vector at higher angles of attack is mainly a matter of roll and yaw control power. As angle of attack increases the rudder effectiveness gets more and more important; if bank angle requirements deducted from MIL-Spec shall be performed in a well coordinated manner at high angles of attack, the body fixed yaw potential has to exceed the roll control power according to the inertia ratio $(i_z/i_x)^2$ multiplied by $tg\alpha$.

Section 5 of this paper presents some peculiarities of configurations and configuration details if the criteria discussed before are considered. Among other aspects it is shown that the (in-) stability characteristics of a Canard configuration are more complex compared to those of an Aft-tail concept. This will require considerably more effort with respect to the optimization work in windtunnel and may lead to more problems during Control Law design.

2. Flightmechancial Design of Unaugmented Stable Configurations

Ever since aircraft have been designed and built the original flightmechanical task within the several development phases has been to provide good and safe Flying Qualities. Necessary margins for static and dynamic stability, required control power in connection with acceptable forces and sufficient trim capabilities in all axes have always been regarded as key characteristics from which criteria for a proper sizing of stabilizers and control surfaces could be derived. - For stable unaugmented fighters of the past ailerons, rudder and elevator were almost directly linked to the pilot by cables, rods and levers via stick and pedals in the cockpit. So right from the start of the design the transfer functions "Aircraft behaviour due to Pilot or Gust Input" and the characteristic equations of the total system were analytically defined.
The desired static and dynamic features of the aircraft could directly be related to the size and positioning of horizontal and vertical stabilizers and the several control surfaces.
With respect to pitch axis most of the applicable criteria which can be deducted from the Handling Qualities document MIL-F-8785 without detour, turn out to form corner stones for the design for the horizontal tail (stabilizer and elevator). Fig. 1 for example summarizes dynamic requirements for the short period and phugoid characteristics and presents the relevant equations which define the relations to the aerodynamic derivatives. Parametric variations in size and position of the horizontal tail lead to continuous parameter changes and if the minimum and maximum limits from MIL-Spec are considered, permissible forward and aft centre-of-gravity positions due to criteria for dynamic stability can be evaluated as a function of tail area or volume. Together with boundaries derived from other requirements like "Nose Wheel Lift-Off at a given airspeed (1.0 V_S)", "Minimum Static Stability", "Trim in several flight conditions", etc. Design diagrams like the example given in Fig. 2 can be plotted and so the main goal of Horizontal-Tail-Design, to define and open a permissible flightmechanical c.g. range, may be achieved straight away.
The same principle procedure is applicable in lateral/directional axes as shown in Fig. 3 and 4. The correlations between the Handling Quality Requirements for Dutch Roll, Spiral-, Roll mode, etc. and the aerodynamic derivatives are again analytically linked by the characteristic equations and by relatively simple transfer functions. So particular variations of configuration details, as for example vertical tail size and/or position, will directly lead to changes in the dynamic behaviour of the aircraft and with the Handling Quality Requirements under consideration stabilizer, rudder and ailerons can be defined.

3. Design Criteria for modern Configurations with Unstable Characteristics in Pitch

The flightmechanical design of highly augmented aircraft with unstable basic characteristics is not longer straight forward. In addition to handling quality aspects, maneuverability and agility which have to be provided up to high angles of attack, further design goals like optimum point performance, observance of structural limits and carefree handling has to be integrated into the flightmechanical considerations. The task to integrate the requirements from different disciplines is transferred to a single "Black Box", that is to say on the Flight Control System. The various subtasks, which have to be managed by the FCS may roughly be split into two parts: One is related to the control of steady-state and instantaneous maneuvers which includes surface scheduling for different modes, optimum coordination of the available controllers and the surveillance of structural and physiological limitations. The other part covers the vital aspects of stabilization throughout the permissible flight envelope. - So the growing use of the possibilities of modern flight control systems has partly relieved both the fixed stabilizers of the aircraft from their task to provide stability and the pilot from his responsibility to optimize the manoeuvres of the aircraft. On the other hand the requirements for control potential in all axes have to be increased and it is necessary to relate the available control power to the tolerable basic (in-)stability characteristics of the aircraft. - The summary in Fig. 5 shows in which direction the flightmechanical design aspects have to be changed if stability augmentation or artificial stabilization is introduced: Sufficient static and dynamic stability and, at high angles of attack, acceptable departure characceristics have to be replaced by the limitation of basic dynamic instabilities. The provision of control potential for maneuvers and trim has to be supplemented by additional power for stabilization purposes and if required for exotic modes. Acceptable stick and pedal forces need no longer be taken into account within early design phases. They can be treated separately and optimized artificially in combination with the cockpit design. - So, in contrary to unaugmented aircraft for which sizing of the aerodynamic surfaces could directly be done by consideration of level-1 Handling Quality requirements, it is now necessary to design for the capabilities of the FCS which will be developed in a later phase to provide the desired handling. Horizontal-/vertical tails, ailerons and rudder can no longer be treated as completly seperate segments. They have to be considered as multi-functional, integrale stability and control units which are linked via FCS for optimized management of trim, maneuver and limitation tasks.
As the capabilities of any existing and future Flight Control System will be restricted by unavoidable technical inadequacies of realistic hard- and software, it is necessary to design for basic unaugmented characteristics which can be handled by the FCS even under adverse conditions. Therefore requirements which include the aspects of control law design have to be developped and prepared for use in the aerodynamic optimization process in order to restore the early link between Handling and Aerodynamics which has been broken by the FCS.

As complete design loops are always assigned by the four parameters "Mass", "Performance", "Cost" and "Risk", the idea of "Performance" has to be revised by including requirements from agility, handling and ride quality and by introducing essential aspects from safety point of view. As demonstrated in section 4.1 and 4.2, where the development of possible criteria for maximum instability in pitch axis are desribed, simplified assumptions are necessary to fix the corner stones. It has to be kept in mind that any criterion must be convertible into aerodynamic characteristics to enable the design team ....

... to define feasible aerodynamic instability levels,
... to fix trim schedules which leave sufficient control power in pitch, roll and yaw,
... to optimize the basic aerodynamic characteristics in windtunnel
... to size and position the control surfaces and stabilizers.

4. Development of Specific Criteria for Modern Fighter Configurations

The road which has to be taken in order to develop a suitable set of criteria for maximum instability and necessary control power in pitch will start with the definition of aerodynamic and flightmechanical key characteristics, which can be analysed and systematically varied within early design phases. Via the introduction of realistic, principles of control law design and under the consideration of handling quality and safety aspects limits for maximum dynamic instability levels, which are easily transferable into aerodynamic characteristics, will be evaluated.

4.1 Key Characteristics for Unstable Design in Pitch

From the very beginning all the design phases of "New Generation" fighter aircraft are dominated by the attempt to find an optimum balanced concept within the frame of maximum performance, defined mass figures and limited costs. Especially the field of performance encompasses aspects on at least three planes, which are defined by the headlines "Mission-, Point- and Manoeuvre Performance". Requirements derived from these different items are often rather contradicting.
A suitable tool to overcome some of the contradicting requirements is the introduction of Unstable Design in pitch which has remarkable effects on performance as demonstrated in Fig. 6: The trim characteristics of the sample aircraft (i.e. a tailless configuration: the principles apply for any tailed configuration as well) show that the stable version will have negative slopes in the pitching moment-lift diagram for controls fixed. Therefore it is necessary to trim the configuration with negative (i.e. upwards) flap deflections. An unstable design with the centre of gravity aft of the aerodynamic centre, has a positive $\partial C_m/\partial C_L$- (and $C_{m\alpha}$) slope and therefore requires positive (i.e. downwards) flap settings from trim. The sketch of the polars on the righthand side of Fig. 6 shows the resulting beneficial effect on trimmed performance data:
Typical supersonic fighter wings are characterized by a relatively small aspect ratio and high leading-edge sweep. Especially for those the induced drag for a given lift coefficient is much smaller with positive than negative flap deflections. This leads on one hand to a remarkable reduction in overall drag at a desired turn rate and on the other to much larger trimmed maximum lift coefficient. If the full technically feasible potential of unstable design is used, then relative to a conventionally stable aircraft maximum lift can be increased by roughly 25 % and induced drag at a typical lift coefficient for manoeuvre (say $C_L \approx 0.7$) can be reduced by about 20 %.
This means that unstable configurations when designed for the same performance requirements and under the same flight mechanical constraints, will be remarkably smaller than their stable "brothers".
But it has to be kept in mind that, a pure optimization for maximum point performance (i.e. sustained and instantaneous turn rates) which requires maximum lift or minimum drag respectively may not be advantageous for a desired superior agility, because the preloaded aerodynamic controls do not leave enough power to initiate and stop manoeuvres in a way which lead to sufficient handling qualities.
A qualified parameter which indicates the potential for drag reduction and higher maximum lift is the static instability level (Static Margin).

$$SM = -\frac{\partial C_m}{\partial C_L} = -\frac{C_{m\alpha}}{C_{L\alpha}} = \frac{x_{ac} - x_{cg}}{\bar{c}}$$

If only aerodynamic aspects had to be taken into account, it would make sense to increase the negative static margin to a point where the resulting trimmed flap schedule leads to optimum drag polars.
From the flight mechanical point of view a reasonable interpretation of SM is only possible in linear areas at smaller angles of attack. It defines nothing else than the lever arm of aerodynamic forces and is only useful for the flight mechanical design of unaugmented aircraft, which have to be equipped with a certain margin of static stability. As maximum lift is approached ($C_{L\alpha} \rightarrow 0$) the physical evidence of SM ($SM \rightarrow \infty$) is lost. - More definite parameters which can be interpreted at high angles of attack as well, are summarized in Fig. 7. The "Aerodynamic Key Characteristics" in terms of typical C_m versus α plots (coefficients and derivatives) with curves for "zero" and "full nose down" controls point out that one limiting factor for unstable design will be given by the definition of a necessary pitch recovery moment which above all has to guarantee a safe return from high angle-of-attack manoeuvres. The basic design instability covering only performance aspects, will usually be chosen at low and medium angles-of-attack. This instability has to be checked against the capabilities of the flight control system. The same applies to the allowable pitch-up at higher angles-of-attack in trimmed conditions.

Unfortunately neither the static instability SM nor the pitching moment derivativa $C_{m\alpha}$ are sufficient to describe the dynamic problems associated with the stabilization of unstable configurations. According to the equations of Fig. 1 and 15 the highly dynamic short period motion of any aircraft is dependent on much more factors as for example moment of inertia, damping derivatives, wing area, mean aerodynamic chord and dynamic pressure. All these parameters contribute to a "Time to Double Amplitude" T_2, during which, with controls fixed the aircraft will double a distortion in angle of attack. As the simplified schematic graphs of Fig. 8 (pitching moment versus time) show, it is now essential to counteract this aperiodic movement by an appropriate control input. Build up rate $\dot{M}$ and magnitude M of the stabilizing moment must be large enough to stop and reverse the sign of pitch acceleration so that the aircraft returns to its original, trimmed condition. The Time Delay T_t between disturbance input and stabilizing control reaction can be identified as a further important parameter which will increase the problems of control law design if it exceeds a certain percentage of T_2.
So the key characteristics for unstable design in pitch may be summarized, as done in Fig. 9:

- Static Margin SM or $C_{m\alpha}$ for the aerodynamicists
- Time to Double Amplitude T_2, Maximum Pitch Control Moment M and Build-up Rate $\dot{M}$, Time Delay T_t for the control law designers and flight mechanics people

In practice it is necessary that control law people and aerodynamicists can communicate and understand each other in order to end up with a well balanced design. So once dynamic limits for T_2 have been identified they have to be translated into aerodynamic characteristics $C_{m\alpha}$, $C_{L\alpha}$ or SM. A good approximation for the transcendental relation between T_2 and the aerodynamic derivatives is presented in Fig. 15. - For the control moments the simple algebraic equations

$$M_{max} = \pm \Delta C_{m_{max}} (\eta) \cdot \frac{\rho}{2} V^2 \cdot S \cdot \bar{c}$$

$$\dot{M}_{max} = \pm C_{m\eta} \cdot \dot{\eta}_{max} \cdot \frac{\rho}{2} V^2 \cdot S \cdot \bar{c}$$

may be used. The amount of pitch control authority M can be regarded as the sum of moments which is available from all reliable, primary controllers. For both, build-up rates $\dot{M}$ and authority M, limitations due to hinge moments and load restrictions have to be considered.

4.2 Development of a Criterion for Maximum Dynamic Instability in Pitch

The summarizing discussion about the key characteristics of unstable design in pitch from section 4.1 has confirmed that it might be possible and helpful to develop relationships between maximum permissible dynamic instability T_2 and required control authority M and build-up rate $\dot{M}$. In order to take concrete steps towards such a criterion it is now necessary to bother his head with parametric attempts to optimize control law parameters for different dynamic instability levels of modern fighter configurations.
The procedure has to be done under realistic (but simplified) assumptions including essential requirements and methods which are normally used within a thorough design of a Flight Control System:

- realistic control law structure
- realistic hardware assumptions
- optimization of Control Law parameters with respect to handling and ride qualities
- Consideration of safety margins in control law design
- Variation of "Time to Double" and Control Moment characteristics by realistic, relevant aerodynamic data

• Control Law Structure and Aircraft Model

To correct the short period divergence of an aircraft with basically unstable characteristics in pitch and at the same time, significantly improve the aircraft flying qualities a variety of Flight Control Systems can be constructed. The system finally selected will depend on many considerations as for example problems of instrumentation and sensing including biases and sensor excitation, control system compensation needed for flight condition changes, nonlinearities, scheduling, boundary control etc.
To permit the development of a criterion for maximum dynamic instability, a typical example of a Pitchrate Command System, which could be regarded as the essence of an FCS for modern fighter aircraft, has been chosen [Lit. 1]. The flight control system selected is shown in Fig. 10. This system performs six main functions as follows:

- Creates a high degree of effective static stability
- Positions the short period roots via the feedbacks RA and RQ
- Gives sufficient damping of the phugoid by integral q-feedback (RQI)
- Provides pitch rate/attitude hold platform
- Shapes $T_{\theta 2}$ according to the pitch rate filter
- Regulates against external disturbances with emphasis on pitch attitude maintance

To account for the time delay due to sensors, computers, filters etc. a representative equivalent value of 0.02 s has been added to the system by Padé approximation.
The overall model of the "Augmented Aircraft" which was finally used to develop the criterion is shown on the righthand diagram of Fig. 10. Together with the basic aircraft (4. order), the control system incl. Padé approximation (4. Order) and the actuator dynamics for canard, flaps and aft tail (1. Order, $T_M = 0.05$ s) the overall model added up to a system of 9. Order.

• Handling Qualities Requirements

The discussions within AGARD Working Group 17 have lead to the conclusion that many existing criteria in the frequency and time domain are qualified to optimize and assess the handling qualities of augmented aircraft. It has been made clear that a single criterion will in any case not be sufficient. Therefore it is necessary to establish a set of criteria which is individually tuned for the aircraft under consideration.
For our purposes the criteria of the Low Order Equivalent system (MIL-F-8785) have been applied as design goal for the optimization of the Control Law parameters discussed above. The requirements for the short period motion are summarized in three subcriteria, as there are:

- Minimum and maximum damping
- Short period frequency characterized by the CAP-parameter
- Time constant $T_{\theta 2}$ according to ω_{osp}

Roughly spoken the CAP parameter describes the aircraft behaviour after a step input in pitch stick in terms of required 'g'-onset $\dot{n}_{Req}$ and steady state load factor (Fig. 11). Good handling qualities are characterized by a reasonable time after which the steady state load factor has to be achieved. Furthermore no excessive overshoots or oscillations are allowed which leads to the design goals selected for the (CAT.A) flight phases of our study:

- CAP = 1.0 / ω_{osp}
- Short period damping $\zeta = 0.75$
- $T_{\theta 2}$ according to ω_{osp} ($\omega_{osp} \cdot T_{\theta 2} = 3.6$)

To limit the actuator activity it was necessary to add the requirement that the damping of all complex roots should be greater than 0.75. - In addition it was required to take care of the gust responses during the optimization of control parameters in a way that the gust response should not be significantly worse than that of a conventional aircraft.

• Safety Aspects, Robustness of the FCS

Experience in practical engineering has always proven that differences between the best models and reality (i.e. the aircraft in operation) cannot be avoided. Uncertainties with respect to data and model accuracy have to be taken into account and a certain robustness of the FCS is required to maintain stability and control even if the aerodynamic efficiencies for example, deviate considerably from the values predicted in windtunnel.
A qualified method to assess the robustness and safety of the augmented aircraft system is based on the Nyquist criterion (see Fig. 12) which predicts the stability of the closed loop by analysis of the frequency response of the open control loop. According to this criterion the closed control loop is stable, if for a gain of 0 dB of the open loop the phase shift is less than -180°.
The distance from this stability limit may be regarded as a measure for the quality of stabilization. So the distance of the phase from -180° at a gain of 0 dB is called "Phase margin" whereas the "Gain Margin" is defined as difference in gain to 0 dB at -180°. - A gain margin of - 6 dB and a phase margin of 45° is usually considered to be adequate.
For a basically unstable system two stability limits have to be taken into account as pointed out in Fig. 12. If the overall gain is too high the control system is suspicious to turn unstable again while for too low gains the task of stabilization may fail at all. Therefore in addition to the required gain margin of -6 dB a further margin of +6 dB has to be introduced. - Practical application has shown that for modern fighters a phase > -145° at ±3 dB (35° margin) is more appropriate which finally leads to the trapezoidal area of "Insufficient Stability Margins" shown in Fig. 12. - The consideration of this requirement therefore forms an essential assessment tool within the control law design for modern fighters.

• Analysis of typical Aerodynamic Data for modern Fighters

Inspite of the challenging requirements which have been formulated for future fighters based on the European Szenario, it is astonishing that solutions based on all tail concepts have been proposed. The configurations, which are roughly sketched in Fig. 13, are all equipped with relatively low aspect ratio wings (Λ < 3) and the common understanding of the design teams has been that an unstable design together with a horizontal tail (aft-tail, vector nozzle, canard) is needed to achieve a high performance standard and agility.

The table of Fig. 14 summarizes typical data for these configurations including geometrical, aerodynamical and mass/inertia characteristics, which should be valid for a preliminary dynamic analysis in the subsonic region up to maximum lift. The first line of the table shows that the design instabilities (static margin SM) of the tailed aircraft are considerably larger than that of the tailless configuration. This is due to requirements concerning the pitch recovery margin (section 4.3) which for tailless concept is hard to achieve together with an unstable design.
In order to get some deeper insight into the dynamic effects of the data presented, it is necessary to have a look at the equations which determine the location of the short period roots in the complex plane and the key parameter T_2. Fig. 15 illustrates that the positions of the two unstable roots are defined by two major parts: $(\zeta \cdot \omega_0)$ settles the line of symmetry from which the two aperiodic roots s_1 and s_2 will seperate once static stability is lost. This first part is influenced by the lift curve slope $C_{L\alpha}$ and by the dynamic derivatives but it is not dependent on the design instability.
The shift of the roots on the Real Axis is dominated by the term ω_0^2 and therefore directly related to $C_{m\alpha}$.
If the data from Fig. 14 are analysed by the dynamic equations of Fig. 15 it gets evident that the relatively high mass density of the configurations $\mu_L > 70$ (50 for tailless aircraft) deminishes the influence of the dynamic aerodynamic derivatives and the lift curve slope considerably. The line of symmetry $(\zeta \cdot \omega_0)$ is therefore situated between -0.2 for small subsonic Machnumbers near maximum lift and -2.8 for high subsonic speeds at low angles of attack. These values are valid for the whole data bandwidth; the differences between the three tail concepts are negligible.
The term ω_0^2 is clearly dominated by $C_{m\alpha}$; as above the other derivatives are of minor importance because they are all devided by the high mass density.
So with respect to the aerodynamic derivatives the key parameter T_2 will mainly be altered by $C_{m\alpha}$; $C_{L\alpha}$ is of minor importance but should be considered; the bandwidth of the dynamic derivatives may in any case be neglected.
Reviewing the pitch control data of Fig. 14 major differences between the configurations can be identified for the effectiveness of the controllers. For aft tail and flaps a negative ratio $C_{m\eta}/C_{L\eta}$ has to be taken into account whereas for the control surface Canard positive values can be expected.

- Evaluation Procedure for the T_2-Criterion

The definition of control law structure, aircraft model, handling quality and safety requirements for the optimization of control law parameters and the identification of the relevant aerodynamic data have set the necessary conditions to run the evaluation procedure for the T_2 criterion (Fig. 16): Starting from general data of modern fighters represented by three different tail concepts, the relevant aerodynamic parameters ($C_{m\alpha}$, $C_{L\alpha}$, $C_{m\eta}/C_{L\eta}$), Machnumber and dynamic pressure have been varied in order to achieve different values in Time-to-Double Amplitude T_2. In a second step the control law parameters have been defined by optimizer strategy in accordance with the handling quality and safety requirements defined above; simulated flights with test inputs have been performed in order to evaluate the required control power and control moment build-up rate.
A vertical gust ramp of 66 ft/s with 50 m ramp length has been selected for "Test Input", as shown in Fig. 17. As the probability of such a gust is remote, no further control power for additional pilot commands has been required. (Some simultaneous control inputs, smaller gust plus pilot command, have been tested as well. But after some discussions about reasonable maximum load factor - or angle-of-attack-onset in connection with Command Shaping, after a stick input, it was decided to stay with the gust ramp as well defined requirement from MIL-Spec.).

- T_2-Criterion and Discussion of Results

The results, presented in Fig. 18 confirm that it is possible to generate limiting functions "Required Pitch Control Power M" and "Required Pitch Control Power Build-up Rate $\dot{M}$" versus "Time-to-Double Amplitude T_2". As T_2 decreases the required control authority and rate increases rapidly. Especially for smaller Mach-numbers the continuous lines are limited by sharp edges which mark the point where the safe phase- and gain margins, defined above, could no longer be achieved. Differences found for the different tail concepts can be neglected. So the limits shown in the two diagrams should be valid for all modern fighter configurations with a mass density $\mu_L > 50$ and a control system with an equivalent overall time delay of $T_t = 0.02$ s.
In general a larger time delay will not require increased control authority or rate, as pointed out by the time histories in Fig. 19. It could be shown that the optimum gains within the control law parameters will decrease as time delay grows which leads to a more sluggish behaviour of the aircraft and to a delayed reduction of external disturbances. Of greater importance however is the loss in Phase-on Gain margin as shwon in Fig. 20. Especially high gains, which will hurt the lower boundary, become critical which substantiates the statement that the optimum gains will be lower with increasing T_t.
As far the limits of the T_2-criterion (Fig. 18) are concerned no increase in the required control power would be necessary if the overall time delay exceeds 0.02 s. The sharp edges for insufficient Phase- and gain margins, however have then to be shifted towards higher T_2 values.

A further analysis of Fig. 18 shows that configurations with time to double T_2 < 0.2 s may not be feasible for a production aircraft. Even a small unfavourable error concerning T_2 (caused by aerodynamic uncertainties for example) would require excessive additional control power and/or would hurt the achievable phase and gain margins. This does not mean that such an aircraft cannot be realized or equipped with reasonable handling qualities (see for example X 29). But the statement does point out that such a high dynamic design instability introduces a lot of risk which does not pay-off and that during development and operation major difficulties with respect to safety and handling may be encountered.

4.3 Margins for Pitch Recovery from High Angles of Attack

The minimum pitch recovery control power which has to be installed at high angles-of-attack near $C_{L_{max}}$ cannot only be defined by sufficient nose down acceleration which has to provide a safe return from maneuvers near stall. A more detailed analysis of the problem leads to the conclusion that the required nose down control power can roughly be split into two parts (Fig. 21):

1) basic demand for stabilization, for counteracting gusts and for sufficient pitch handling qualities during high angle-of-attack manoeuvres

2) additional control power for increased agility at high angles-of-attack

The basic demand, which has to be provided in the nose-up as well as in the nose-down direction, could probably be defined by design charts like those developped in section 4.2. The criterion presented in Fig. 18, however is based on handling quality requirements of CAT.A flight phases and a heavy gust ramp. Both are not applicable for high angle of attack maneuvers. So the whole criterion has to be recalculated on a modified basis which has not be done up to now. - As a rule of thumb the required pitch acceleration could be fixed at about $\ddot{\theta}$ = ±0.3 rad/s². This margin which should be designed for in any case, is supported by several simulation studies and recent work within several fighter projects.
Additional pitch control power for increased agility at high angles-of-attack is directly combined with the requirements for maximum roll rate in this region. The sketch on the lefthand side of Fig. 21 shows that any roll rate around the velocity vector is combined with a pitch-up moment. The aircraft acts like a dumb-bell and the resulting inertial coupling produces a nose-up acceleration which is given by:

$$\ddot{\theta}_{iC} = \frac{1}{2}\, p_V^2 \cdot \sin 2\alpha$$

So beneath the basic recovery margin additional pitch down control power is needed to counteract the inertial coupling during roll manoeuvres. As soon as the angle-of-attack for maximum lift (i.e. roughly the location of minimum nose down control power) is known it is possible to draw a design chart of required pitch down acceleration versus roll rate, as shown on the righthand side of Fig. 21. The fix of a roll rate requirement at a certain calibrated airspeed leads us straight forward towards the nose down recovery margin in terms of $\ddot{\theta}$ or pitching moment coefficient $\Delta C_{m_{Rec}}$ which has to be installed. It is important to point out that a certain loss in pitching moment due to differential flaps has to be taken into account; this leads to the slightly transverse line in the design chart if the recovery moment is defined to be derived from the configuration with all pitch controls deflected fully down.

4.4 Design Criteria for Lateral/Directional Stability

Considerations about requirements for the lateral/directional basic characteristics of a modern fighter design have to start with the evidence that an unstable design in roll/yaw will not lead to such remarkable gains in performance as destabilization in pitch. Furthermore a dynamically unstable aircraft in pitch and yaw may multiply the complexity of the flight control system and hence is not very likely to pay off.
The consequence is that at low as well as at high angles of attack the design should aim towards coefficients and derivatives which produce at least indifferent roots in the dynamic analysis (slightly unstable spiral mode excluded).
The critical area for low angle-of-attack characteristics, control fixed, may be found at high supersonic Mach numbers. In the region of maximum dynamic pressure the elastic factors usually diminish the stabilizing contribution of the vertical tail.
For the low speed/high angle-of-attack region stable directional/lateral derivatives ($+C_{n_\beta}$, $-C_{\ell\beta}$) with smooth behaviour versus sideslip, avoidance of yaw and roll departure tendencies, sufficient margin for spin resistance and effective rudder/roll control power highlight the optimization goals.
To assess departure and spin resistance, the "Dynamic Directional Stability Parameter" $C_{n_{\beta_{dyn}}}$ and the "Lateral Control Departure Parameter" LCPD have been developed and proposed as a prediction method by Weisman [see Ref. 2, 3, 4]. The resulting Weisman Criterion (taken from [Ref. 3]) in Fig. 22 specifies regions of stable and unstable behaviour in the $C_{n_{\beta_{dyn}}}$-LCDP plane. $C_{n_{\beta_{dyn}}}$ itself has been derived from the characteristic equation as it is summarized in Fig. [illegible], using the experience that divergence usually occurs if the C-coefficient becomes negative. As shown in [Ref. 7] many of the terms, contained in C are usually small enough to be neglected. The result of the evaluation leads to the conclusion that directional divergence is likely to occur, if

$$C_{n\beta dyn} = C_{n\beta} \cdot \cos\alpha \quad \left(\frac{I_z}{I_x}\right)^2 \cdot C_{\ell\beta} \cdot \sin\alpha$$

approaches zero or gets negative. This tendency was checked against the behaviour of several high performance aircraft and the correlation turned out to be fairly good. So it has become common use for preliminary design to set a certain minimum positive margin for $C_{n\beta dyn}$ to make sure that spin tendency at high angles-of-attack is excluded. Meanwhile many papers have been published (see for example [Ref. 4 and 6]) where the clear evidence was pronounced that some key phenomena may not be covered by the criterion. Especially the regions, marked with "affected by secondary factors" in Fig. 22, and the many examples of current modern fighters, whose behaviour at high angles of attack were found to be different from the $C_{n\beta_{dyn}}$-prediction, shows the need for revision of this criterion.

In examining the data used by Weisman it was found in [Ref. 4] that some important features of fighter aircraft have changed since the criterion was developed. Especially maximum lift and usable angle of attack have shifted to considerably higher values ($\alpha_{CL_{max}}$ ~ 20° - 30°) which implies that the static and dynamic lateral/directional derivatives are now dominated by forebody vortices from nose, strakes or canard. For the older aircraft the dynamic data were of minor influence and the departure characteristics in Weisman's correlation were dominated by the static derivatives. - High angle of attack characteristics of modern aircraft are more dependent on the dynamic derivatives which are heavily influenced by forebody geometry.
In spite of the fact that fighter configurations and usable flight envelopes have changed the characteristic equation of Fig. 23 is still valid: Stable behaviour may be expected if all the coefficients B, C, D and E are positive and if the Routh discriminent $(C \cdot B - D) \cdot D - EB^2$ keeps larger than zero. - So with typical data from modern fighter configurations under consideration some design rules for the aerodynamic optimization in windtunnel may be derived, as done in Fig. 24:
A rough estimation shows that according to the geometric, mass and inertia figures of typical modern fighters (Fig. 15 with radius of inertia i_x = 1.5, i_z ≈ 3.4; span ≈ 10 m) the following characteristic values may be assumed:

- mass density: $\mu_s > 80$
- inertia ratios: $K_x \approx 0.1 < K_z \approx 0.5$
- flightmechanical time: $t_F < 0.1\ \mu_s$

Furthermore the results of various windtunnel tests show that ...

... $C_{y\beta}$ will always stay negative

... $C_{\ell r}$ will always stay positive ($C_{\ell r} \sim C_L$, $C_{\ell r} > 1.0$ at C_{Lmax})

Keeping these assumptions in mind the design rules for the aerodynamicists are straight forward:

- The B-coefficient stays positive if autorotation is avoided ($C_{\ell p} < 0$) and yaw damping is maintained ($C_{nr} < 0$).
- The C-coefficient is dominated by $C_{n\beta dyn}$; this parameter must be kept larger than zero.
- To keep D- and E-coefficient and the discriminant positive it is essential to maintain $C_{n\beta_{dyn}} > 0$ with negative $C_{\ell\beta}$ and only slightly positive $C_{n\beta}$ (≈ +0).

In any case it is necessary to emphasize that the dynamic aerodynamic data has to be included into the design process as early as possible and that stable and linear characteristics for the essential derivatives, as listed above, have to form a fundamental design goal within the optimization of modern fighters.

4.5 Design Rules for Lateral/Directional Controllers

The essential factors which influence the control power requirements in roll and yaw directly may be deducted from MIL-Spec., as for example from requirements for "Time to Bank", "Engine failure during Take-off" and "Take-off/Landing in Crosswind".
Control power for stabilization or stability augmentation of the lateral/directional axis is dependent on the chosen basic stability characteristics, as discussed above. But as long as no excessive instability in roll or yaw has to be covered the control power deducted from the other criteria should be sufficient.
The capability to initiate and maintain coordinated rolls especially at high angles-of-attack represents a major point of interest especially for future fighter aircraft with high agility in this part of the flight envelope. Already during preliminary design phases these aspects may be covered. Fig. 25 illustrates within three sketches in the time domain the essential parameters which afterwards will lead to roll and yaw power, required from aerodynamic or thrust vector devices.

For preliminary design the roll performance of an aircraft may be sufficiently described by the Roll Time Constant T_R, the Maximum Roll Rate p_{MAX} and a "Time to Bank to Ø degrees". Especially at high angles-of-attack most of the control law designs try to avoid sideslip and therefore prefer a well coordinated roll around the velocity vector. So the "pitch recovery margin" which has been provided according to the discussion in chapter 4.3 sets the first corner stone by defining the maximum achievable roll rate p_{VMAX} (roll rate around the velocity vector). A rough calculation shows that the roll power necessary to maintain this roll rate is far too small to get sufficient handling qualities: The resulting time constant T_R and as a consequence the "Time to Bank" to an arbitrary bank angle is much too large for an agile aircraft. So once a requirement for "Time to Bank" or a certain "Roll Time Constant" is settled some additional roll acceleration, $\Delta\dot{p}_V(t)$ will be necessary during the first few seconds of the manoeuvre. In practice a "Time to Bank" requirement leads straight towards a "Time Constant" requirement if the maximum roll rate is fixed. Thus, to initiate a coordinated roll manoeuvre the necessary roll acceleration (around velocity vector) can be simply defined as:

$$\dot{p}_{V0} = \frac{p_{VMAX}}{T_{R\ REQ.}}$$

Fig. 26 now points out how the roll acceleration requirement has to be transfered into body fixed yaw and roll control power. The definition of angle-of-attack and calibrated airspeed/dynamic pressure, where the agility is required, leads to the deduction of the body fixed roll and yaw control power requirements. Some further analysis shows that for any coordinated roll manoeuvre onset the relation

$$C_{n0} = C_{\ell 0} \cdot \frac{i_z^2}{i_x^2} \cdot \mathrm{tg}\,\alpha$$

must be satisfied. Because of $i_z > i_x$ good roll performance at high angles may only be achieved if sufficient yaw control power can be provided.
The summary of all the discussions above is presented in Fig. 27 showing a "design chart" for yaw and roll controllers at high angles-of-attack. The diagram (body fixed yawing moment versus body fixed rolling moment) contains the line of coordination (defined by equation above) and an arbitrary minimum requirement for C_n and C_ℓ (deducted from Fig. 26).
The aileron and/or flaperons at high angles-of-attack usually produce an adverse yaw/roll characteristic. Starting from this characteristic it is now necessary to meet the coordination line above the requirement by providing the appropriate yaw control power. It gets evident that this does not only require a certain yawing moment C_n but also a C_n-C_ℓ characteristic of the yaw controller. Once the yaw/roll control behaviour is fixed by configuration details it is of no use to increase the yaw potential beyond the "line of coordination". The capabilities for a well coordinated roll manoeuvre will not improve.

5. Aerodynamic/Flightmechanical Pecularities of Configurations and Configuration-Details

Using the stability and control requirements derived in the previous sections, the aerodynamicists together with the overall design specialists have to look for a well balanced design. During the different predesign phases several (tail-) concepts will be investigated and compared and solutions based on all three horizontal tail arrangements will usually be proposed, as already illustrated in Fig. 13 and 14. In general it is not very difficult to provide the necessary pitch control power together with a reasonable design instability by qualified configurative means as shown in Fig. 28. Only for a merely tailless aircraft a destabilization SM ≤ 0 % $\bar{c}$ will probably be impossible to realize, unless exotic wing concepts (gotic wings) are used which introduce a lot of natural pitch-down at higher angles of attack. The introduction of a "Pitch-Thrust-Vector Nozzle" as an internal horizontal tail (Fig. 28 right side) offers an attractive alternative to regain the advantages of an aerodynamically unstable tailless aircraft (Ref. 8).
If the typical shifts of aerodynamic centre versus Machnumber for the different tailconcepts are analysed, as done in Fig. 29, critical areas for maximum dynamic instability levels for small angles of attack may be identified. For the Canard Configuration the a.c. keeps moving aft continuously within the whole subsonic Machnumber range so that the maximum static instability for low angles of attack may be expected in the low speed region.
Transferred into dynamic characteristics the minimum Time to Double T_2 may be found close to Mach = 0.7. - As a consequence of decreasing tail increments the aerodynamic centre of an aft tail configuration tends to move foreward and will reach its most critical position (SM_{max}) at high subsonic Machnumbers. The dynamic instability in terms of T_2 will in general follow which leads to a minimum Time to Double near Mach = 0.9. - For the tailless configuration only a slight (forward) a.c. shift may be expected; therefore maximum dynamic instability will also be found in the high subsonic range.
It is of course not sufficient to confine the analysis of maximum instability to low angle of attack data, because the dominant $C_{m\alpha}$ derivatives may deviate considerably from the chosen design values. Especially a canard tends to induce lifting forces in front of the centre of gravity and is likely to introduce a lot of pitch up close to maximum lift as pointed out in Fig. 30. Consequently the dynamic instability will be increased and the key parameter T_2 will get well below the original design values. Together with fading control power this characteristic close to C_{Lmax} could lead to some difficulties with respect to stabilization and handling. In this context the behaviour of an aft tail configuration seems to be more promising: A decreasing downwash increases the

stabilizing contribution of the tail versus angle of attack which leads to a continuous pitch-down effect. The maximum dynamic instability may therefore be expected at low angles of attack where the largest control potential will be available. This favourable characteristic will meet the general wishes, to introduce instability in order to minimize drag up to medium angles of attack. As maximum lift is approached the instability is reduced considerably which helps to optimize handling and to avoid complexity of the control laws. - The $C_{m\alpha}$-characteristics of a tailless configuration are dominated by the geometry of wing and strake which should be optimized in a way that only minor pitch-up tendencies up to medium angles of attack are to be expected.

The effects of a wing strake on the basic pitch behaviour are shown in Fig. 31. As strake size increases the aerodynamic centre is shifted forward which implies that for a given design instability the maximum $C_{m\alpha}$ will get larger and the pitch recovery margin will be reduced. In addition some increase in maximum lift (probably not usable) and a shift towards higher $\alpha_{CL_{max}}$ may be expected.

A not very obvious but nevertheless big effect on maximum instability and recovery margin may come from the choice of the proper vertical tail configuration. Fig. 32 illustrates the loss in pitch down capability and the increase in $C_{m\alpha_{max}}$ when replacing the single vertical by a twin tail. The flow breakdown at higher angles of attack produces a download between the two vertical fins and therefore a positive contribution to the pitching moment coefficient. Windtunnel tests have shown that this effect is almost independent on the cant angle of the verticals.

In addition to tail concepts, strakes etc., the choice of the wing planform itself may be crucial for the danger of too large pitch-up. As shown in the C_m-α diagrams on the left side of Fig. 33 a "normal" trapezoidal wing will obtain its most critical SM and $C_{m\alpha}$ at low Machnumbers. In transonic and supersonic regions those planforms tend to restabilize and pitch down; the aerodynamic centre moves continuously aft and instability disappears. - Other wing planforms like cranked wings may exhibit a different unfavourable behaviour (see righthand graph of Fig. 33): At low angles of attack the usual aft movement of a.c. is found whereas at medium α a considerable pitch-up tendency is extended to trans- or even supersonic Machnumbers because of a vortex burst at the Kink-station.

Once a first wing-body-tail concept with its geometrical wing data "aspect ratio" and "sweep" has been selected, it may be possible that the resulting pitch-up tendencies turn out to be unsatisfactory. A proper possibility to improve the situation may be found by looking for alternatives in the wing planform itself. - Similar to the so called "NACA-Pitch-up Line" it is possible to draw diagrams of "similar pitch-behaviour at high angles of attack" for trapezoidal wings depending on the geometrical parameters "aspect ratio Λ" and "leading edge sweep ϕ" (→ Fig. 34). If, starting from a baseline configuration, more "Pitch-down" is required at high angles of attack, it doesn't make much sense to change the wing geometry parallel to these lines. It is more promising to introduce changes perpendicular to the curves of "Similar Pitch-up" which means to decrease the product $\Lambda \cdot \phi$. The same principal relationships may be found for wing-strake-combinations; the lines are then shifted parallel towards smaller Λ-ϕ-values.

As already discussed in section 4.4, the original design philosophy for the lateral/directional unaugmented behaviour has not changed. The general design goal is still to maintain natural stability which just has become more difficult to achieve because of the need to go to higher angles of attack. Here induced forces and moments from the forebody (nose/strake/canard) tend to dominate the static and dynamic aerodynamic data and the aft-body surfaces (vertical tails) will generally loose their importance. The diagrams in Fig. 35 (taken from [Ref. 4]) and Fig. 36 show that there are fundamental interchanges between static and dynamic derivatives if the shape and/or length of the forebody is altered: It is true that variations towards elliptical cross sections for example will improve the static directional data $C_{n\beta}$. On the other hand the yaw damping C_{nr} and the lateral stability $C_{l\beta}$ may be detoriated which can lead to departure and spin tendencies (see analysis of characteristic equation Fig. 23/24). - Advices for optimum forebody-geometry are difficult to be defined because the interactions of the different emerging vortices and the resulting forces and moments can hardly be predicted. The analyses of numerous windtunnel tests however have shown that it may be most promising to design for a round nose cross section with a forebody length of $\ell_N/d \approx 5.5$ (d = fuselage diameter at wing apex).

As the lateral/directional data at high angles of attack is heavily dependent on the forebody geometry the scheduling and control deflections of a canard may hurt the important aerodynamic derivatives as well (see Fig. 37). Especially close to maximum lift $C_{l\beta}$ is strongly influenced by and often highly nonlinear versus canard setting which imposes a lot of severe problems both on the aerodynamisists and the flight control/flight mechanics people. - For an aft-tail concept the data is generally much smoother as shown on the right hand side of Fig. 37: Once the forebody shape has been optimized for sufficient lateral/directional stability, the deflections of the horizontal aft-tail will not deteriorate the relevant data.

6. Conclusions

The discussions about recommendations, requirements and limits which have to be taken into account in order to cover "Flightmechanical" and "Flight Control System" points of view, have shown that it is mandatory to involve these aspects already into a preliminary design process of a modern fighter aircraft. The criteria which have been derived are of major influence for the overall configuration.

Maximum allowable instabilities and control power requirements, will set remarkable constraints to the freedom of aerodynamic design and influence essential components of the aircraft. Because of the complex aerodynamic effects at high angles-of-attack it will be necessary to approach the "basic configuration" by some optimization loops especially in low speed wind tunnel tests. During the whole process specialists from flightmechanics, aerodynamics and overall design departments have to form a close team in order to end up with an excellent well-balanced design.
Furthermore the discussion about flightmechanical criteria which can and have to be used within the pre-design phases of modern fighters, has shown that no homogenious set of requirements is available up to now.
Especially the impact of unstable design together with high angles of attack maneuvering is not covered sufficiently. Therefore further research is urgently needed to develop criteria which show the inter-relationship between attainable Flying Qualities, Design Instability and required Control Power within the several flight phase categories or angles of attack regions.

7. Nomenclature

B, C, D, E		Coefficients of characteristic equation, see Fig. 23
b		Wing span
CAP	[s^{-2}]	Control Anticipation Parameter
C_D	[-]	Drag Coefficient
$\bar{c}$	[m]	Mean Aerodynamic Chord
C_L	[-]	Lift Coefficient
$C_{L\eta}$	[rad^{-1}]	Lift due to Deflection of Pitch Controller
$C_{L\alpha}$	[rad^{-1}]	Lift Curve Slope
$C_{L\dot{\alpha}}$	[rad^{-1}]	Lift due to Angle of Attack Rate
C_{Lq}	[rad^{-1}]	Lift due to Pitch Rate
C_{ℓ}	[rad^{-1}]	Rolling Moment Coefficient (Body Fixed)
$C_{\ell\beta}$	[rad^{-1}]	Rolling Moment due to Sideslip
$C_{\ell p}$	[rad^{-1}]	Rolling Moment due to Roll Rate
$C_{\ell r}$	[rad^{-1}]	Rolling Moment due to Yaw Rate
C_m	[-]	Pitching Moment Coefficient
$C_{m\eta}$	[rad^{-1}]	Efficiency of Pitch Controller
$C_{m\alpha}$	[rad^{-1}]	Pitching Moment derivative
$C_{m\dot{\alpha}}$	[rad^{-1}]	Pitching Moment due to Angle of Attack Rate
C_{mq}	[rad^{-1}]	Pitching Moment due to Pitch Rate
C_n	[-]	Yawing Moment Coefficient (Body Fixed)
$C_{n\beta}$	[rad^{-1}]	Directional Stability
$C_{n\beta_{dyn}}$	[rad^{-1}]	Spin Resistance Parameter (Def. see Fig. 22)
C_{np}	[rad^{-1}]	Yawing Moment due to Roll Rate
C_{nr}	[rad^{-1}]	Yawing Moment due to Yaw Rate
$C_{n\zeta}$, $C_{\ell\zeta}$	[rad^{-1}]	Rudder efficiency
C_y	[-]	Sideforce coefficient (Body Fixed)
$C_{y\beta}$	[rad^{-1}]	Sideforce due to Sideslip
C_{yp}	[rad^{-1}]	Sideforce due to Roll Rate
C_{yr}	[rad^{-1}]	Sideforce due to Yaw Rate
d	[m]	Diameter of Fuselage at Wing Apex
H	[m]	Altitude
i_x	[m]	
i_y	[m]	Radii of Inertia (Body Fixed)
i_z	[m]	
K_x	[-]	
K_y	[-]	Ratios of Inertia (see Fig. 15/23)
K_z	[-]	
l	[m]	Length of Vertical Gust Ramp
LCDP		Lateral Control Departure Parameter (see Fig. 22)
M	[m]	Pitching Moment
$\dot{M}$	[Nm/s]	Pitching Moment Build-up Rate
Ma	[-]	Mach Number
m	[kg]	Mass
n, n_z	[g]	Load Factor
p	[deg/s]	Roll Rate
q	[deg/s]	Pitch Rate — Body Fixed
r	[deg/s]	Yaw Rate
RA	[-]	
RAC	[s]	Control Law Parameters see Fig. 10
RQ	[s^{-1}]	
RQI	[-]	
S	[m^2]	Reference Area
s_1, s_2	[s^{-1}]	Short Period Roots
SM	[% $\bar{c}$]	Static Margin (negative = unstable)
t_f	[s]	Flightmechanical Time (see Fig. 23)
T	[s]	Time Constant
T_2	[s]	Time to Double Amplitude
T_N	[s]	Time Constant of Lead Lag Filter
T_R	[s]	Roll Time Constant
T_Z	[s]	Time Constant of Lead Lag Filter
T_t	[s]	Time Delay
t	[s]	Time
V	[m/s]	Airspeed

w_{wg}	[m/s]	Vertical Gust Component
$x_{c.g}$	[% $\bar{c}$]	Centre of gravity
$x_{a.c}$	[% $\bar{c}$]	Aerodynamic centre
α	[deg]	Angle of Attack
β	[deg]	Angle of Sideslip
ζ	[-]	Damping Ratio
ζ	[deg]	Rudder Deflection
η	[deg]	Deflection of Pitch Controller
θ	[deg]	Pitch Attitude
$\ddot{\theta}$	[rad/s²]	Pitch Acceleration
φ	[deg]	Bank Angle
ϕ	[deg]	Leading Edge Sweep of Wing
μ_{L}	[-]	Mass Density (see Fig. 15)
μ_{S}	[-]	Mass Density (see Fig. 23)
ρ	[kg/m³]	Density
ω	[rad/s]	Frequency
ω_0	[rad/s]	Undamped Frequency

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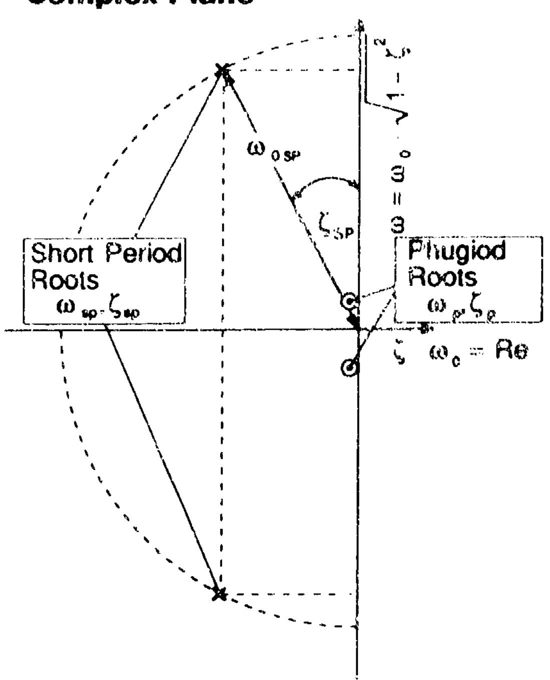
Fig. 1 Correlations between Handling Quality Requirements and Aerodynamic Parameters for an Unaugmented Aircraft (Pitch Axis)



$$\omega^2_{0SP} \approx \frac{V^2}{\mu_L \cdot i_y^2} \cdot \left(C_{m\alpha} + \frac{C_{L\alpha} \cdot C_{mq} - C_{Lq} \cdot C_{m\alpha}}{\mu_L} \right)$$

$$\zeta_{SP} \approx - \frac{C_{mq} - k_y \cdot C_{L\alpha} + C_{m\dot{\alpha}}}{2[k_y(-C_{L\alpha} \cdot C_{mq} - \mu_L C_{m\alpha})]^{1/2}}$$

$$\omega^2_{0P} \approx \left(\frac{V}{\mu_L \cdot \bar{c}} \right)^2 \cdot \frac{C_L(2C_L + C_{Lu})}{1 - C_{Lq}/\mu_L}$$

$$\zeta_P \approx \frac{C_{Du}(1 - C_{Lq}/\mu_L)}{2[C_L(2C_L + C_{Lu})(1 - C_{Lq}/\mu_L)]^{1/2}}$$

$$k_y = (i_y/\bar{c})^2; \quad \mu_L = \frac{2m}{\rho \cdot S \cdot \bar{c}}$$

Requirements from MIL-F-8785 C (Level 1 CAT. A):

$0.35 \leq \zeta_{SP} \leq 1.3$

$3.6 \geq \dfrac{\omega_{0SP}}{n/\alpha} \geq 0.28$

$\zeta_P \geq 0.04$

Fig. 2 Integration of Handling Quality Requirements into the Sizing of a Horizontal Tail (Unaugmented Aircraft)

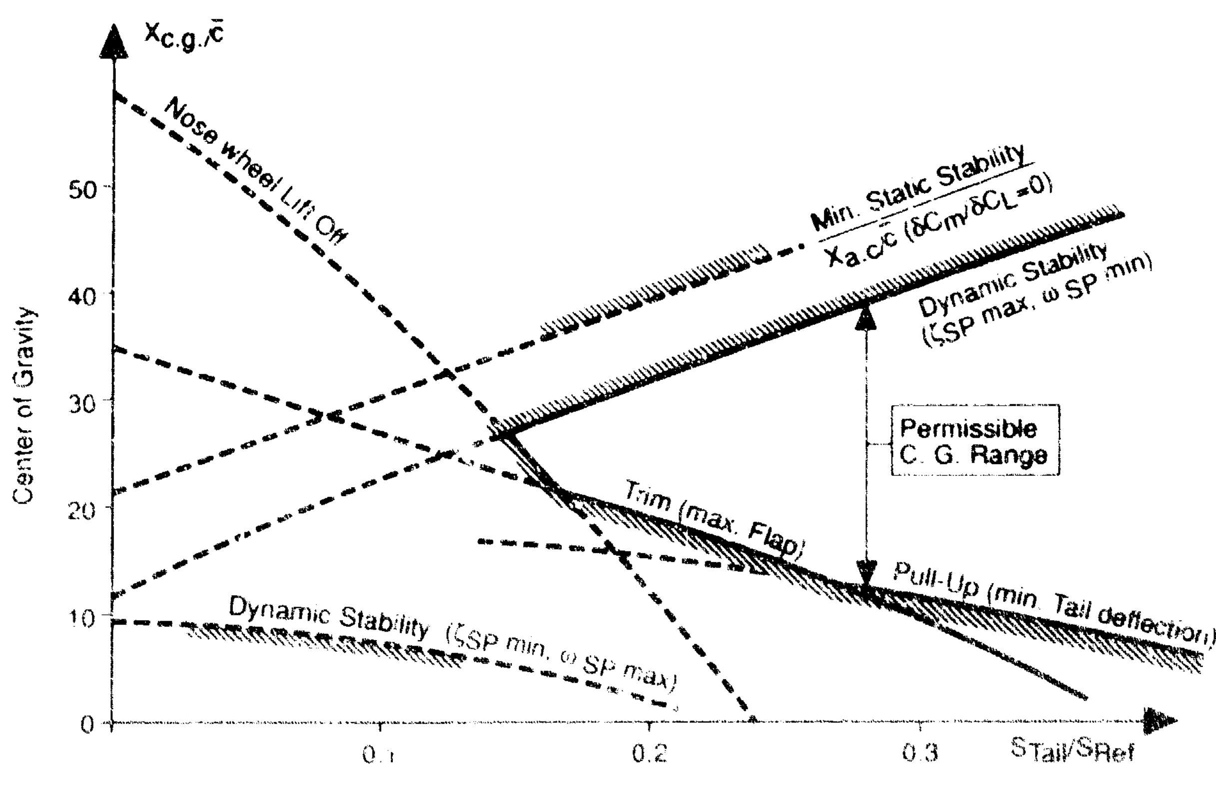
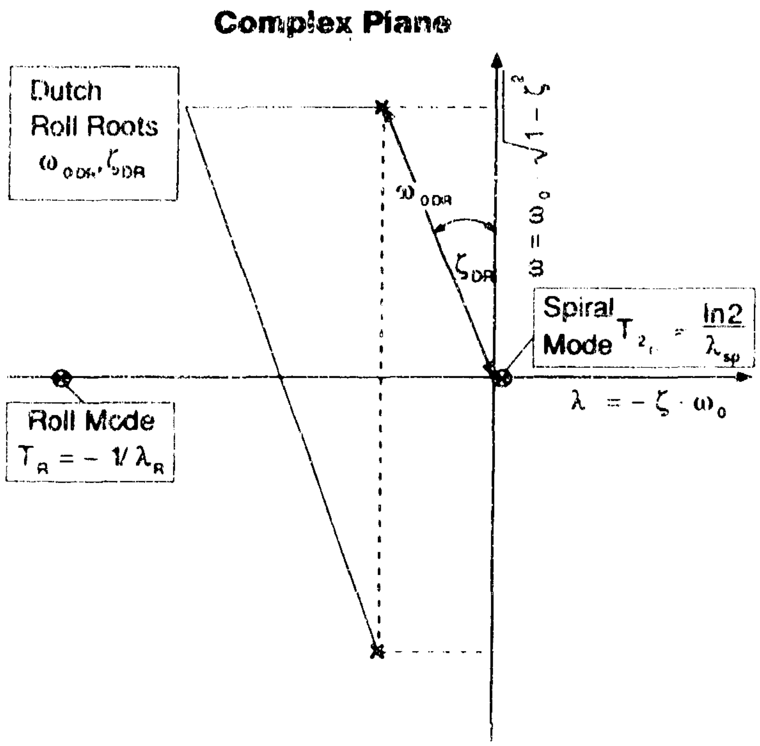


Fig. 3 Correlation between Handling Quality Requirements and Aerodynamic Parameters for Unaugmented Aircraft (Lat./Dir. Axes)



$$\omega_{0DR}^2 \approx \frac{V^2}{\mu_s \cdot i_z^2} \cdot C_{n\beta}$$

$$\zeta_{DR} \approx -\frac{1}{4}\frac{b}{i_z} \cdot \frac{C_{nr}}{\sqrt{\mu_s \cdot C_{n\beta}}}$$

$$T_R \approx -\frac{2 \cdot \mu_s \cdot i_x^2}{b \cdot V \cdot C_{lp}}$$

$$\lambda_{sp} \approx -\frac{g}{V} \cdot \frac{C_{lr} \cdot C_{n\beta} - C_{l\beta} \cdot C_{nr}}{C_{lp} \cdot C_{n\beta}}$$

$$\mu_s = \frac{4m}{\rho \cdot S \cdot b}$$

Requirements from MIL-F-8785 C (Level 1 CAT. A):

$$\left.\begin{aligned} \zeta_{DR} &\geq 0.19 \\ \omega_{0DR} &\geq 1.0\ \text{rad/s} \end{aligned}\right\} \zeta_{DR} \cdot \omega_{0DR} \geq 0.35$$

$T_R \leq 1.0$ s

$T_{2_{sp}} \geq 12$ s

Fig. 4 Integration of Handling Quality Requirements into the Sizing of the Vertical Tail (Unaugmented Aircraft)

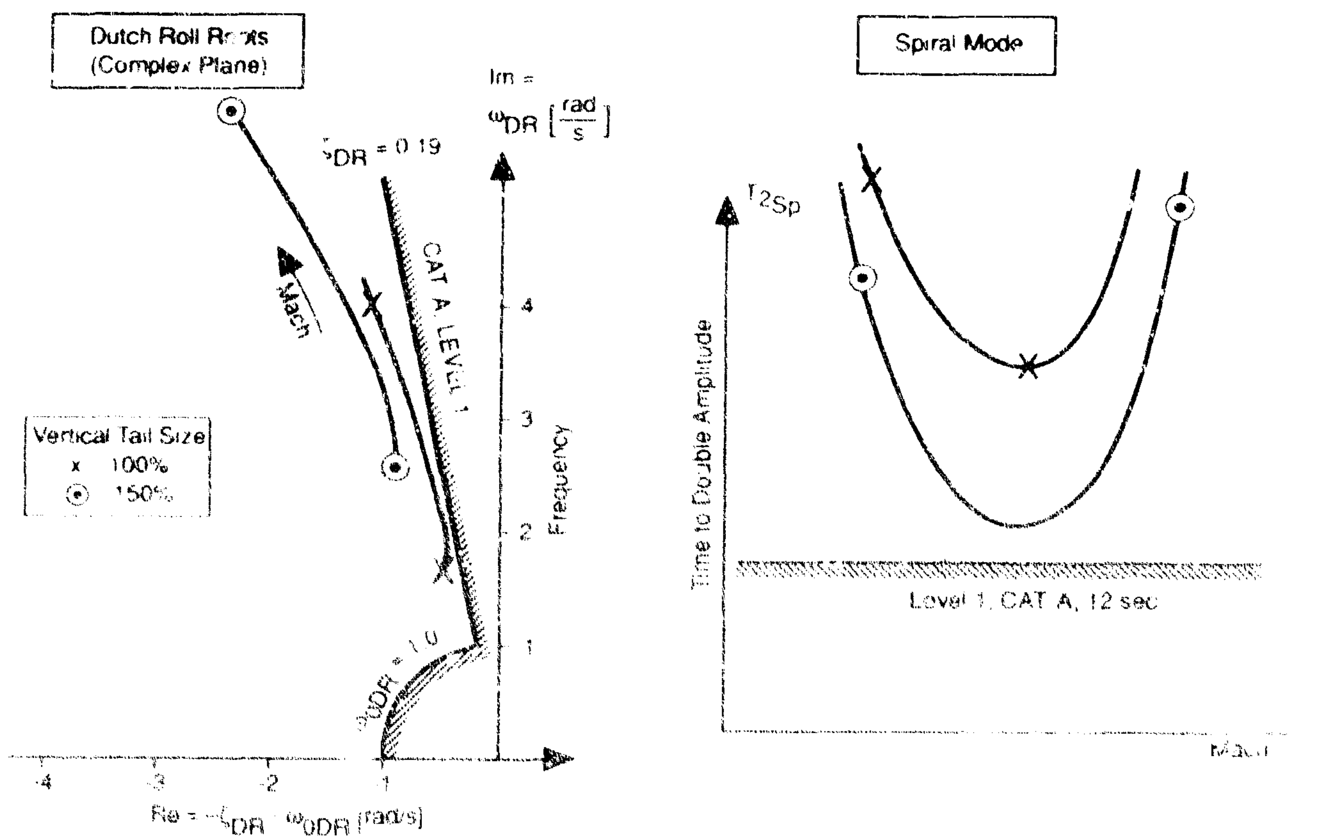


Fig. 5 Flightmechanical Design Aspects

Unaugmented A craft

- Sufficient S d Dynamic Stability
- Acceptable ure Characteristics
- Sufficient Control Potential for Manoeuvres
- Sufficient C rol Potential for Trim
- Acceptable Stick and Pedal Forces

Design for Level-1 Handling Qualities in the Operational Envelope ($\alpha \leq \alpha_{C_{L_{max}}}$)

Horizontal Tail — Ailerons — Vertical Tail/Rudder

Influence of ...

... Wing/Body

... (fixed) Flaps

... etc

Augmented (Basically Unstable) Aircraft

- Limitation of Basic Dynamic Instabilities
- Provision of Control Potential for Stabilization, Trim and (Agile) Manoeuvres
- Artificial Optimum Stick and Pedal Forces

Design for / Check the Capabilities of the Flight Control System to provide Level -1 Handling Qualities upto $\alpha > \alpha_{C_{L_{max}}}$

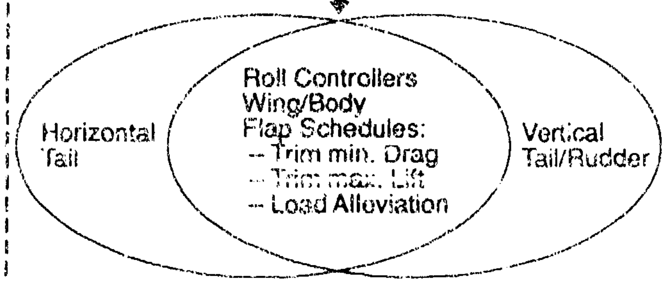


Fig. 6 Key Characteristics for the Aerodynamicists Effect of Destabilization on Performance

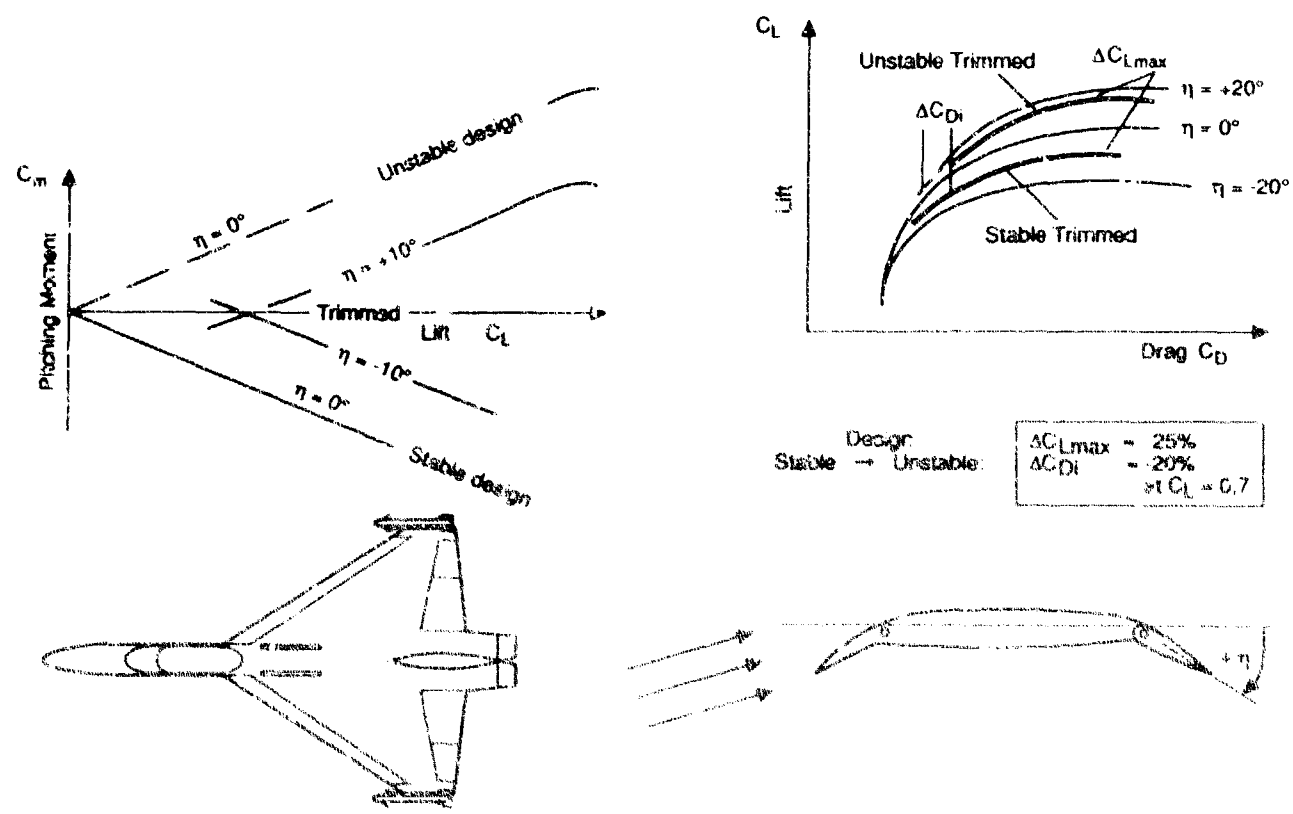


Fig. 7 Key Characteristics for the Aerodynamicists Pitching Moment versus Angle of Attack

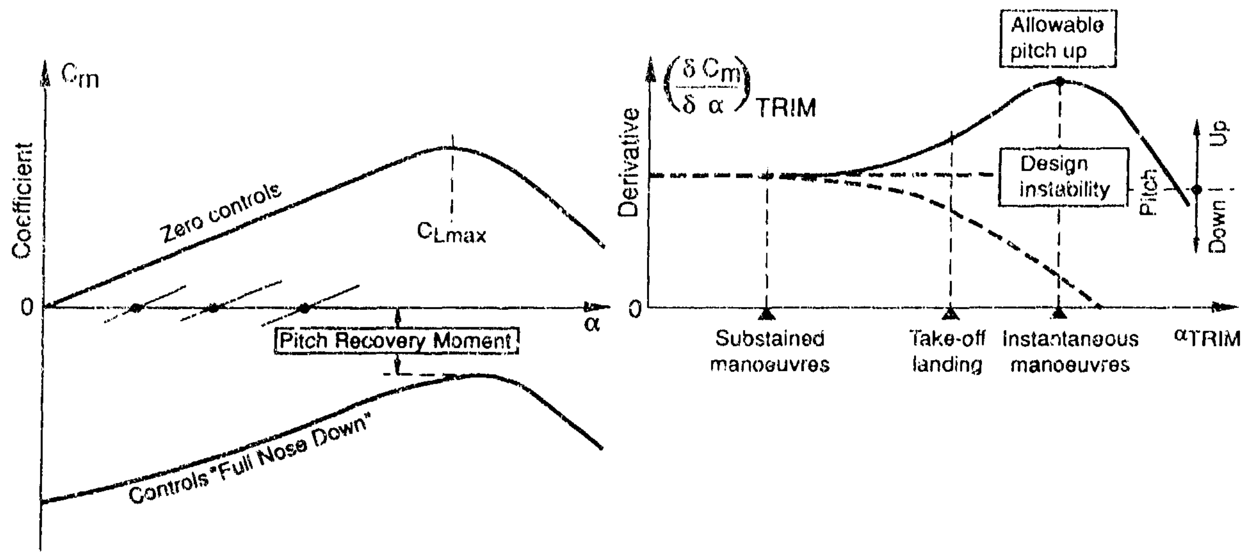


Fig. 8 Principle Problems of (Pitch) Stabilization for an Unstable Designed Aircraft

T_1 = Time Delay
T_2 = Time to Double Amplitude
M = Pitching Moment
$\dot{M}$ = Build-Up of Control Moment

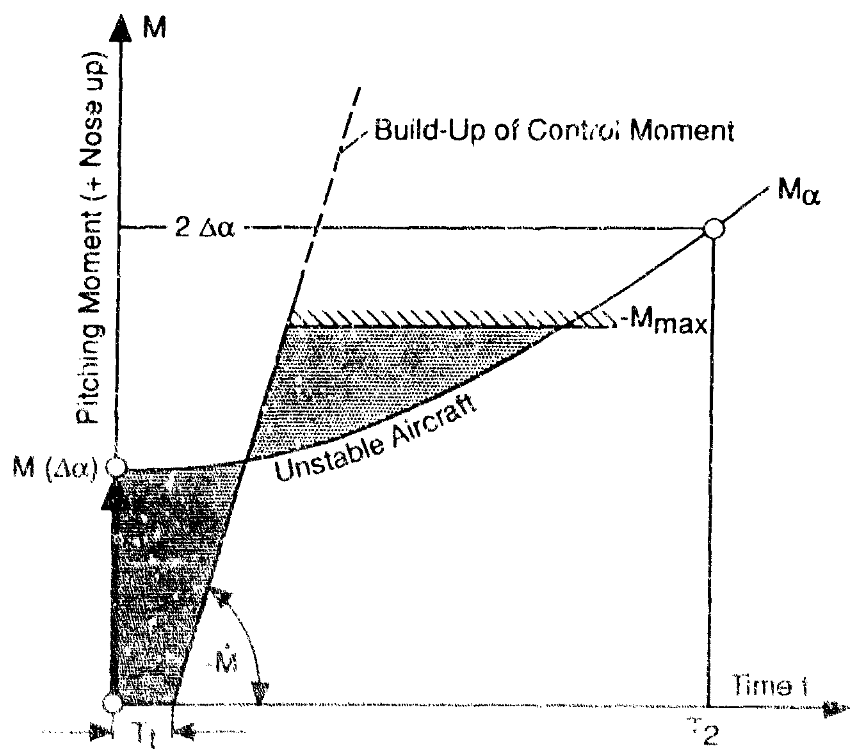
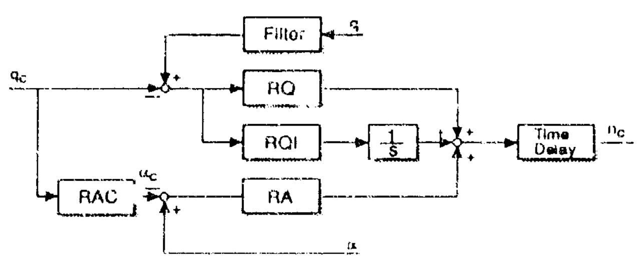


Fig. 9 Key Characteristics of Unstable Design in Pitch

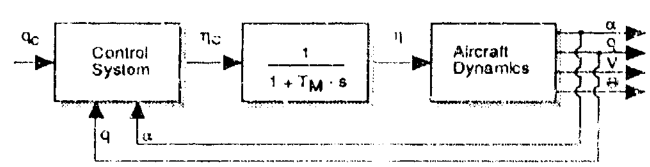
Parameter	Remarks
Static Margin $SM = -\frac{\delta C_m}{\delta C_L} = \frac{x_{a.c.} - x_{c.g.}}{\bar{c}}$	● Aerodynamic Characteristic of Instability ● Lever Arm of Aerodynamic Forces ● Indicates Potential for Drag Reduction ● Indicates Potential for Higher Max. Lift ● Flightmechanical Design Criterion for stable Unaugmented Aircraft ● No Indication for Higher Agility
Time to Double Amplitude T_2	● Quantification of Dynamic Problems → Stabilization → Optimization of Flying Qualities
Maximum Pitch Control Power $\pm M_{max}$ Rate of Control Power Build-up $\pm \dot{M}_{max}$	● Defines Potential to Solve Problems

Fig. 10 Control Law Structure and Aircraft Model

Control Law Structure



Model of the "Augmented Aircraft"



- Positioning of Short Period Rocts (RA, RQ)
- Integral q-Feedback for Sufficient Damping of Phugoid (RQI)
- Transfer of Pitch Rate into Angle of Attack Signal (RAC)
- Pitch Rate Filter for $T_{\theta 2}$ Shaping

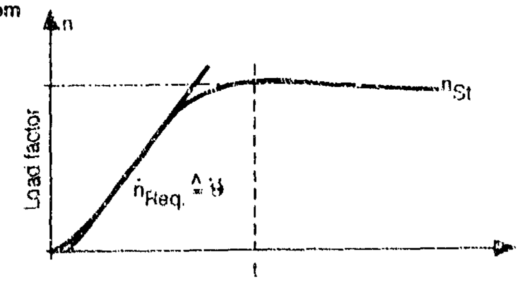
$$\text{Filter} = \frac{1 + T_Z \cdot s}{1 + T_N \cdot s}$$

- Representative Time Delay (Sensors, Computers, Filters, etc.) ≈ 0.02 sec. i.e. State of the Art

●	Basic Aircraft Model	4. Order
●	Control System incl. Pade Approximation for Time Delay	4. Order
●	Actuator Dynamics (Canard, Flaps, Aft-Tail)	1. Order
	Overall System	9. Order

Fig. 11 Handling Quality Requirements Used for Optimization of Control Laws

- **Low Order Equivalent System Requirements from MIL-F-8785 C for Level-1 Handling Qualities**
  - CAP ≈ 1.0 → ω_{osp}
  - Short Period Damping $\zeta \approx 0.75$
  - $T_{\theta 2}$ according to ω_{osp} ($\omega_{osp} \cdot T_{\theta 2} \approx 3.6$)
- **Additional Requirements**
  - Damping of All Complex Roots > 0.75 (limited actuator activity)
  - Gust Response not worse than conventional aircraft



Requirement: n_{St} to be reached after t sec.

$$\mathrm{CAP} = \frac{\ddot{\Theta}}{\Delta n_{St}} \approx \frac{\omega_{osp}^2}{\frac{V}{g} \cdot \frac{1}{T_{\theta 2}}} \approx \frac{\omega_{osp}^2}{n/\alpha}$$

Fig. 12 Safety Aspects covered by Sufficient Robustness of Control System

- **Nyquist Criterion:**

  If at 0 dB gain of the open loop the phase is smaller than -180 deg., the closed loop is....

  ...."**Unstable**"

- **Margins to Stability Limits cover variations of parameters:**
  - Phase Margin: 35 deg. at ± 3 dB
  - Gain Margins: ± 6 dB at -180 deg.

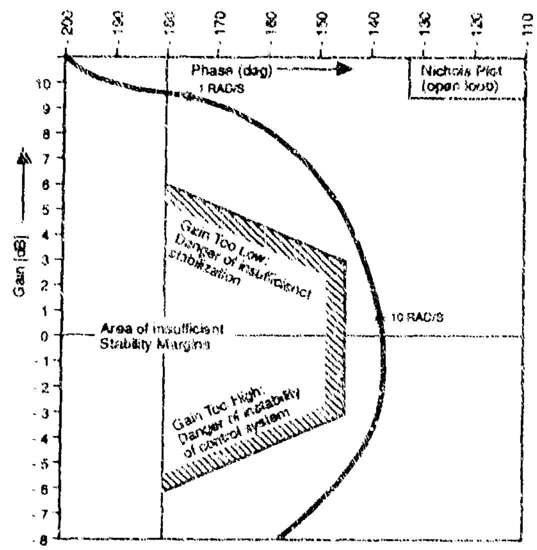


Fig. 13 Typical Modern Fighter Configurations (Designed for European Mission and Manoeuvre Requirements)

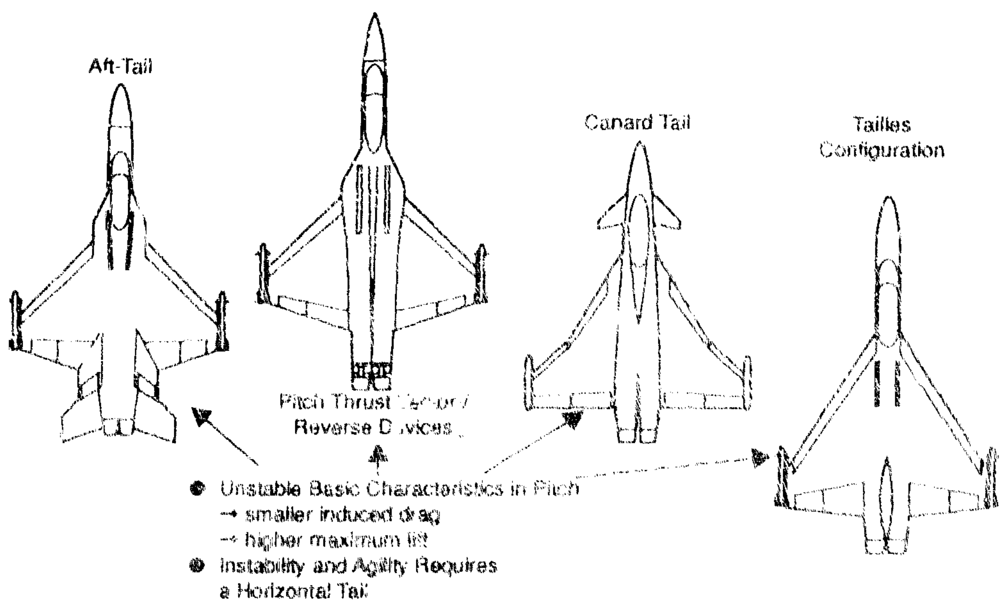


Fig. 14 Typical Data of Modern Fighter Aircraft in the Subsonic Region (Longitudinal Motion);

Parameter	Horizontal Tail Arrangement: Aft-Tail	Vector Nozzle	Canard	Tailless
Static Instability Margin (small α)	SM ≈ −10 %	SM ≈ −8 %	SM ≈ −8 %	SM ≈ 0 %
Medium Combat Mass m	ca. 12500 kg	ca. 12500 kg	ca. 12500 kg	ca. 12500 kg
Reference Wing Aerea S	45 m^2	45 m^2	50 m^2	60 m^2
Mean Aerodynamic Chord $\bar{c}$	5.4 m	5.4 m	5.7 m	6.7 m
$C_{m\alpha}$ ($\alpha = 0° ▷ \alpha_{CLmax}$; Mach < 0.6)	+0.4 ▷ 0.0	0.28 ▷ 0.35 ▷ 0.0	0.26 ▷ 0.5 ▷ 0.0	0.0 ▷ 0.2 ▷ 0.0
$C_{L\alpha}$	+4.0 ▷ 0.0	+3.5 ▷ 0.0	+3.3 ▷ 0.0	2.5 ▷ 0.0
C_{mq}	−1.5 ▷ −2.5	−0.5 ▷ −1.0	−0.5 ▷ −1.0	−0.4 ▷ −0.8
C_{Lq}	+1.6 ▷ +2.0	+1.0 ▷ +1.5	1.2 ▷ +1.9	+1.2 ▷ +1.9
$C_{m\dot{\alpha}}$	−1.0 ▷ −1.0	+0.1 ▷ +0.2	−0.1 ▷ 0.0	−0.1 ▷ 0.0
$C_{L\dot{\alpha}}$	+0.6 ▷ +2.5	+0.6 ▷ +1.5	+0.7 ▷ +1.5	+0.7 ▷ +1.5
C_{Lmax}/α_{CLmax} (Low Speed)	1.50/30°	1.50/30°	1.35/30°	+1.15/≥30°
Tail Efficiency $C_{L\eta}/C_{m\eta}$	+0.45/−0.5	$\ddot{\theta}_{max} \approx \pm 2.0$ rad/s^2	+0.03/0.12*	-/-
Flap Efficiency $C_{L\eta K}/C_{m\eta K}$	+0.63/−0.13**	+1.0/−0.30	+1.15/−0.35	+1.15/−0.35
Mass Density μ_L	84	84	72	50
Radius of Inertia i_y	3.4 m	3.3 m	3.0 m	3.2 m

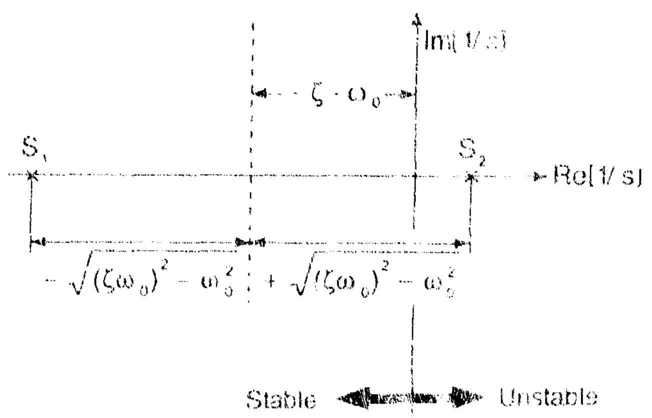
* incl. Wing Interference

** incl. Tail Interference

$\mu_L = \frac{2m}{\rho \cdot S \cdot \bar{c}}$

Fig. 15 Correlations between Aerodynamic Derivatives and Dynamic Instability (Time to Double Amplitude T_2)

$$(\zeta \cdot \omega_0) = -\frac{V \cdot \bar{c}}{2 i_y^2 \cdot \mu_L} \cdot \frac{(1 + \frac{C_{L\dot{\alpha}}}{\mu_L}) \cdot C_{mq} - k_y \cdot C_{L\alpha} + C_{m\dot{\alpha}} \cdot (1 - \frac{C_{Lq}}{\mu_L})}{(1 + \frac{C_{L\dot{\alpha}}}{\mu_L})}$$



$$\omega_0^2 = \frac{V^2}{i_y^2} \cdot \frac{(-C_{m\alpha} + \frac{C_{Lq} \cdot C_{m\alpha} - C_{L\alpha} \cdot C_{mq}}{\mu_L})}{(\mu_L + C_{L\alpha})}$$

$$\mu_L = \frac{2 \cdot m}{\rho \cdot S \cdot \bar{c}}$$

$$k_y = (i_y/\bar{c})$$

$$S_1, S_2 = -(\zeta \cdot \omega_0) \pm \sqrt{(\zeta \cdot \omega_0)^2 - \omega_0^2}$$

$$\boxed{T_2 = \frac{1}{S_2} \ln (2 \frac{S_1 - S_2}{S_1})}$$

$$\boxed{\begin{array}{l} -\zeta \cdot \omega_0 = F(C_{mq}, C_{L\alpha}, C_{Lq}, C_{m\dot{\alpha}}, C_{L\dot{\alpha}}) \\ -\zeta \cdot \omega_0 \ll F(C_{m\alpha}) \quad \omega_0^2 = F(C_{m\alpha}, \ldots) \end{array}}$$

$\rightarrow$

$$\left.\begin{array}{ll} C_{m\alpha} & \text{1. Order} \\ C_{L\alpha} & \text{2. Order} \\ C_{m\dot{\alpha}}, C_{mq} & \text{3. Order} \\ C_{L\dot{\alpha}}, C_{Lq} & \text{4. Order} \end{array}\right\} \text{Parameter}$$

Fig. 16 Evaluation Procedure for the T_2-Criterion
(Required Control Power versus Time-to-Double)

- General Data of Modern Fighters (3Tail Concepts)
- Rough but Reasonable Hardware Assumptions

↓

Variation of Relevant (Aerodynamic) [illegible]acteristics
($C_{m\alpha}$, $C_{L\alpha}$, $C_{m\eta}/C_{L\eta}$, Mach Number,

↓

Varation of Time to Double

↓

Optimization of Control Law Parameters (Optimizer Strategy)
- HQ-Requirements
- Phase- and Gain Margins
- Limited Actuator Activity

↓

Simulated Flights with Test Imputs

↓

Required Pitch Control Moment M
Required Pitch Control Moment Build-up $\dot{M}$

↓

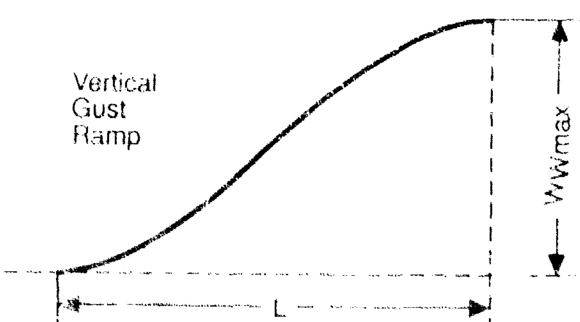
Design Diagrams versus
Time-to-Double T_2

Fig. 17 "Test Inputs" for the Evaluation of the Necessary Control Power

- **Gross Pitch-Stick Command**
  - Maximum Angle of Attack Rate $\dot{\alpha}$?
  - Maximum Load Factor Onset $\dot{n}_L$?
  - "Command Shaping" according to Basic Aircraft Characteristics Possible

- **Heavy Gusts**
  - Defined in MIL-F-8785
  - To be Demonstrated for Certification
  - Cannot be "Shaped"

Test Input



L = 50 m
W_{Wmax} = 66 ft/s
1-cos Form

$\dot{\alpha}_{max}$ = 36 deg/s
$\dot{\alpha}_{AV}$ = 23 deg/s

Fig. 18 T_2-Criterion for Unstable Fighter Aircraft

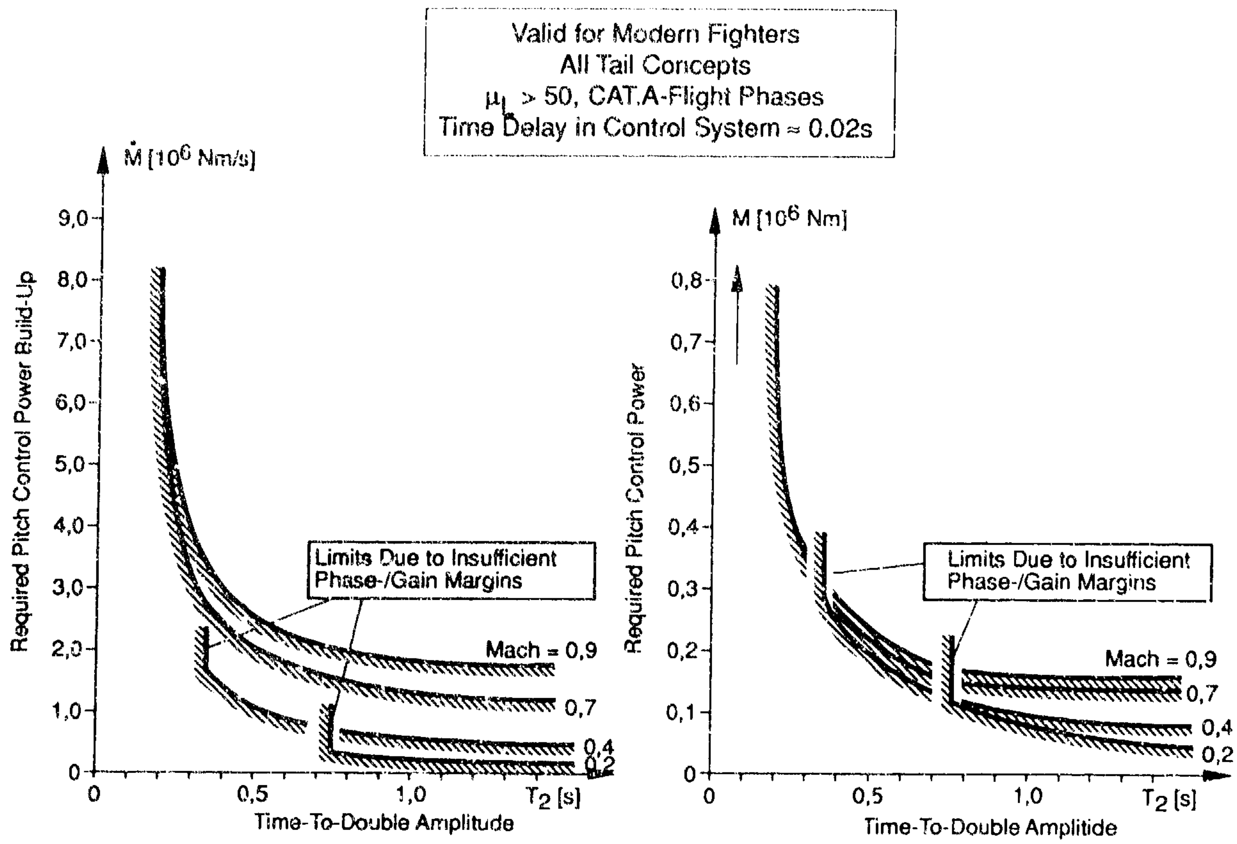


Fig. 19 Response on Test Gust for Different Overall Time Delay

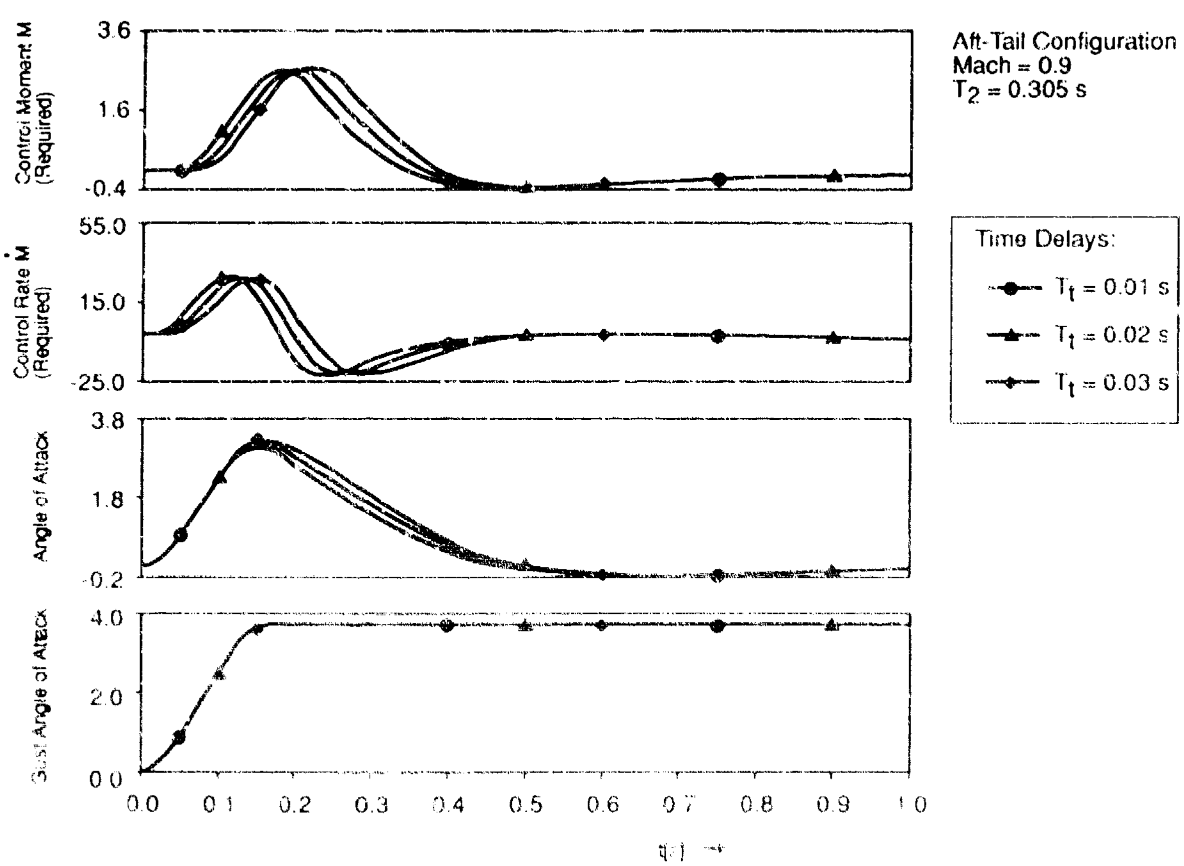


Fig. 20 Effect of Overall Time Delay on Phase and Gain Margins

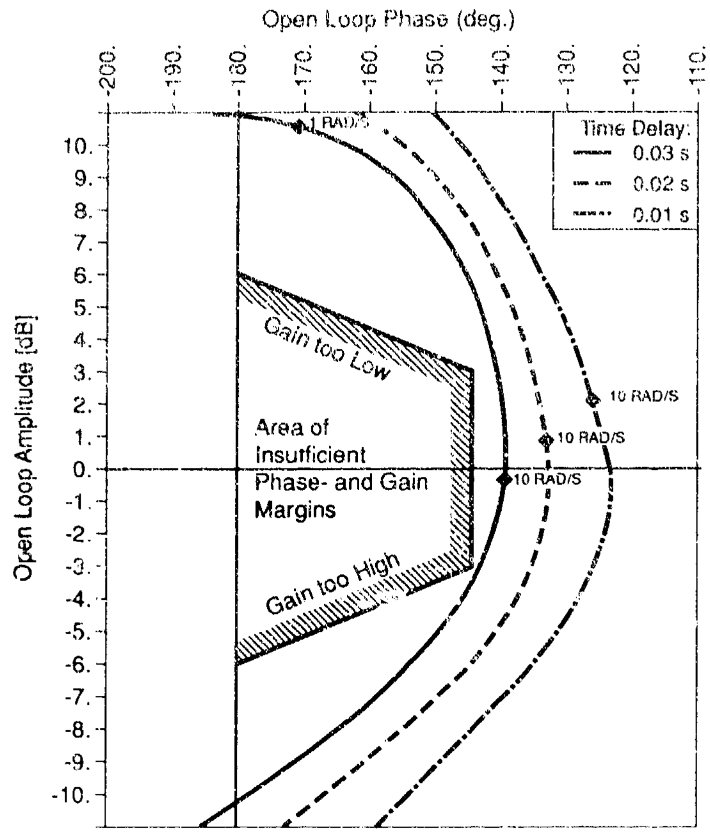


Fig. 21 Definition of Pitch Recovery Margin at High Angles of Attack by Roll Rate Requirement

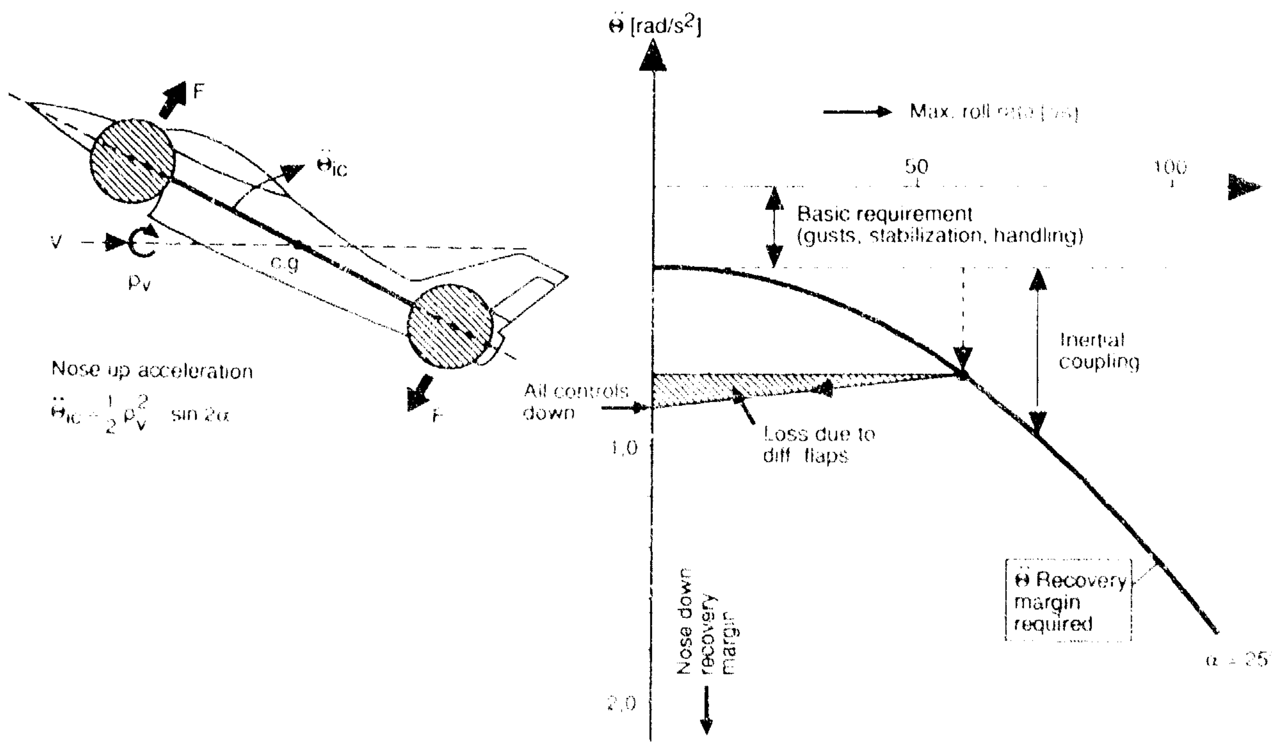
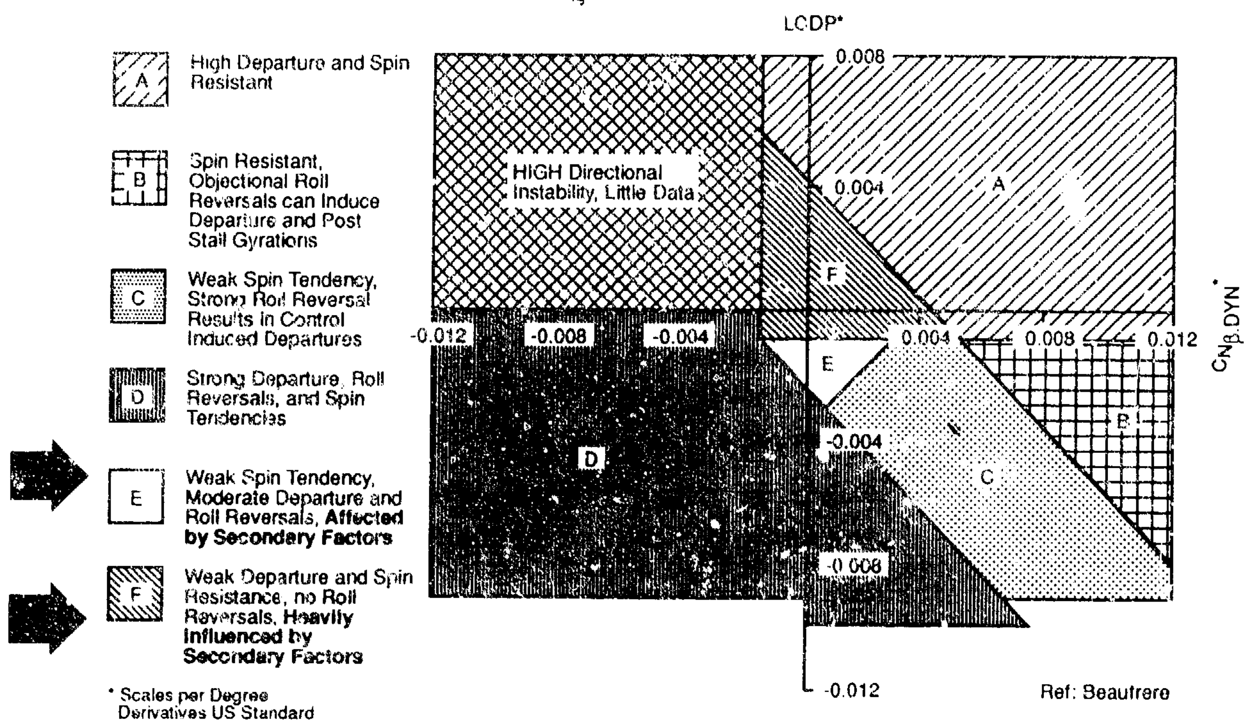


Fig. 22 Weissmann Criterion for Lateral/Directional Stability

$$C_{n\beta_{dyn}} = C_{n\beta} \cdot \cos\alpha - \left(\frac{i_z}{i_x}\right)^2 C_{l\beta} \cdot \sin\alpha$$

$$LCDP = C_{n\beta} - \frac{C_{n\xi}}{C_{l\xi}} \cdot C_{l\beta}$$

Bare Aircraft



* Scales per Degree Derivatives US Standard

Fig. 23 Characteristic Equation of Lateral/Directional Motion (Derivatives in Body Axes)

$$\lambda^4 + B\lambda^3 + C\lambda^2 + D\lambda + E = 0$$

Stable if: B, C, D, E > 0 And $(C \cdot B - D) D - EB^2 > 0$

$$B = \frac{1}{t_r \cdot K_x \cdot K_z} \cdot (K_x \cdot C_{nr} + K_z \cdot C_{lp} + K_x K_z \, C_{y\beta})$$

$$C = \frac{1}{t_r^2 \cdot K_x \cdot K_z} (\mu_a \cdot K_x \cdot C_{n\beta} \cdot \cos\alpha - \mu_a \cdot K_z \cdot C_{l\beta} \cdot \sin\alpha + C_{lp} \cdot C_{nr} - C_{np} \cdot C_{lr} + K_z \cdot C_{y\beta} \cdot C_{lp} - K_z \cdot C_{yp} \cdot C_{l\beta} + K_x \cdot C_{y\beta} \cdot C_{nr} - K_x \cdot C_{yr} \cdot C_{n\beta})$$

$$D = \frac{1}{t_r^3 \cdot K_x \cdot K_z} \Big[\mu_a \cdot (C_{l\beta} \cdot C_{np} \cdot \cos\alpha - C_{n\beta} \cdot C_{lp} \cdot \cos\alpha - C_{l\beta} \cdot C_{nr} \sin\alpha \ldots) + t_r^2 \cdot \frac{g}{s} \cdot (-K_z \cdot C_{l\beta} \cos\alpha - K_x \cdot C_{n\beta} \cdot \sin\alpha) + C_{y\beta} \cdot C_{lr} \cdot C_{np} + C_{l\beta} \cdot C_{yp} \cdot C_{nr} + C_{n\beta} \cdot C_{lp} \cdot C_{yr} - C_{y\beta} \cdot C_{lp} \cdot C_{nr} - C_{l\beta} \cdot C_{np} \cdot C_{yr} - C_{nr} \cdot C_{yp} \cdot C_{lr}\Big]$$

$$E = \frac{g}{t_r^2 \cdot s \cdot K_x \cdot K_z} (C_{l\beta} \cdot C_{nr} \cdot \cos\alpha - C_{nl} \cdot C_{np} \cdot \sin\alpha + C_{nl} \cdot C_{lp} \sin\alpha - C_{n\beta} \cdot C_{lr} \cdot \cos\alpha)$$

$$\mu_a = \frac{4m}{\rho \cdot S \cdot b} \qquad t_r = \frac{2m}{\rho V \cdot S} \qquad k_x = \left(\frac{2i_x}{b}\right)^2 \qquad k_z = \left(\frac{2i_z}{b}\right)^2$$

Ha[illegible]an

Fig. 24 Design Criteria for Lateral/Directional Characteristics (at High Angles of Attack)

Assumptions:

$\mu_s \gg 0$ (typical values ≥ 80)

$K_z > K_x$ ($K_x \approx 0.1$, $K_z \approx 0.5 \div 0.8$)

$\mu_s \gg t_F$ ($\mu_s = \frac{2V}{b} \cdot t_F$; $\mu_s > 10 t_F$)

$C_{y\beta}$ always negative

C_{lr} always positive ($C_{lr} \sim C_L$, $C_{lr} \geq 1.0$ at $C_{L_{max}}$)

Coefficients of Characteristic Equation	Design Rules for Dominating Parameters		
B	Avoid Autorotation	➡	$C_{lp} < 0$
	Maintain Yaw Damping	➡	$C_{nr} < 0$
C	Dominated by $\mu_S \cdot K_X \cdot C_{n\beta_{dyn}}$	➡	$C_{n\beta_{dyn}} > 0$
D, E	Avoid Large Positive Directional Stability	➡	$C_{n\beta} \approx +0.$
	Maintain $C_{n\beta_{dyn}} > 0$ by Lateral Stability	➡	$C_{l\beta} < 0$
$(C \cdot B - D)\, D - EB^2$	Keep C Larger Than D	➡	$C_{n\beta_{dyn}} > 0.1$ with slightly positive $C_{n\beta}$

Fig. 25 Definition of Roll Control Power by Roll Performance Requirements

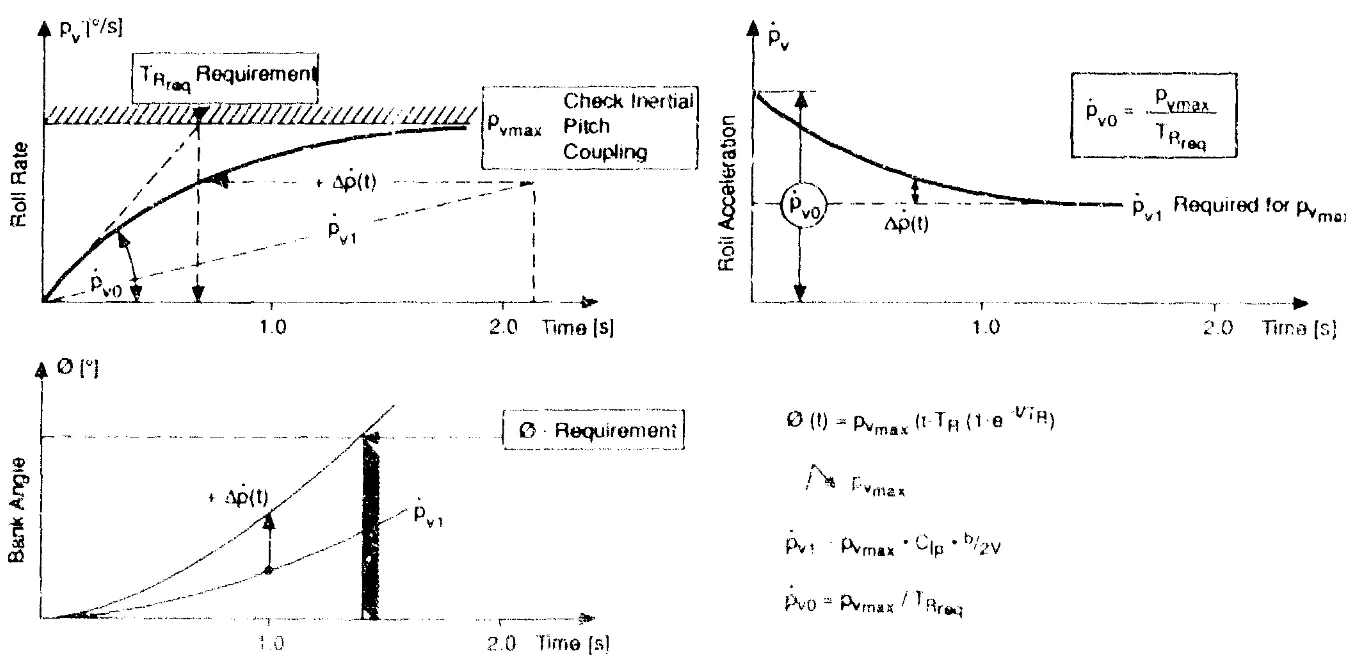
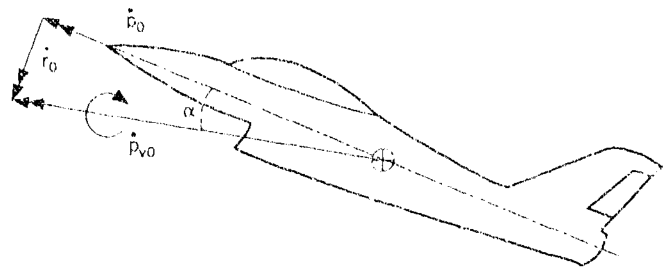


Fig. 26 Definition of Body Fixed Roll and Yaw Control Power (Coordinated Rolls at High Angles of Attack)



Transformation of Angular Accelerations:

Velocity Vector → Body Fixed

$$\dot p_{v0}^2 = \dot p_0^2 + \dot r_0^2$$

$$\dot p_0 = \dot p_{v0} \cdot \cos\ \alpha$$

$$\dot r_0 = \dot p_{v0} \cdot \sin\ \alpha$$

$$\boxed{C_{n_0} = C_{l_0} \cdot \frac{i_z^2}{i_x^2} \cdot tg\ \alpha}$$

Deduction of Aerodynamic Control Potential

$$\dot p_0 = K_p \cdot p_{v_{max}} \cdot \cos\alpha / T_R$$

$$\dot r_0 = K_r \cdot p_{v_{max}} \cdot \sin\alpha / T_R$$

$K_p, K_r > 1. = f$ (Actuators, Time Delay, etc.)

1. Shot: $K_p, K_r = 1.25$

Aerodynamic Coeffients $\Delta C_n, \Delta C_l$

Fig. 27 Design Diagram for Yaw and Roll Control Power at High Angles of Attack

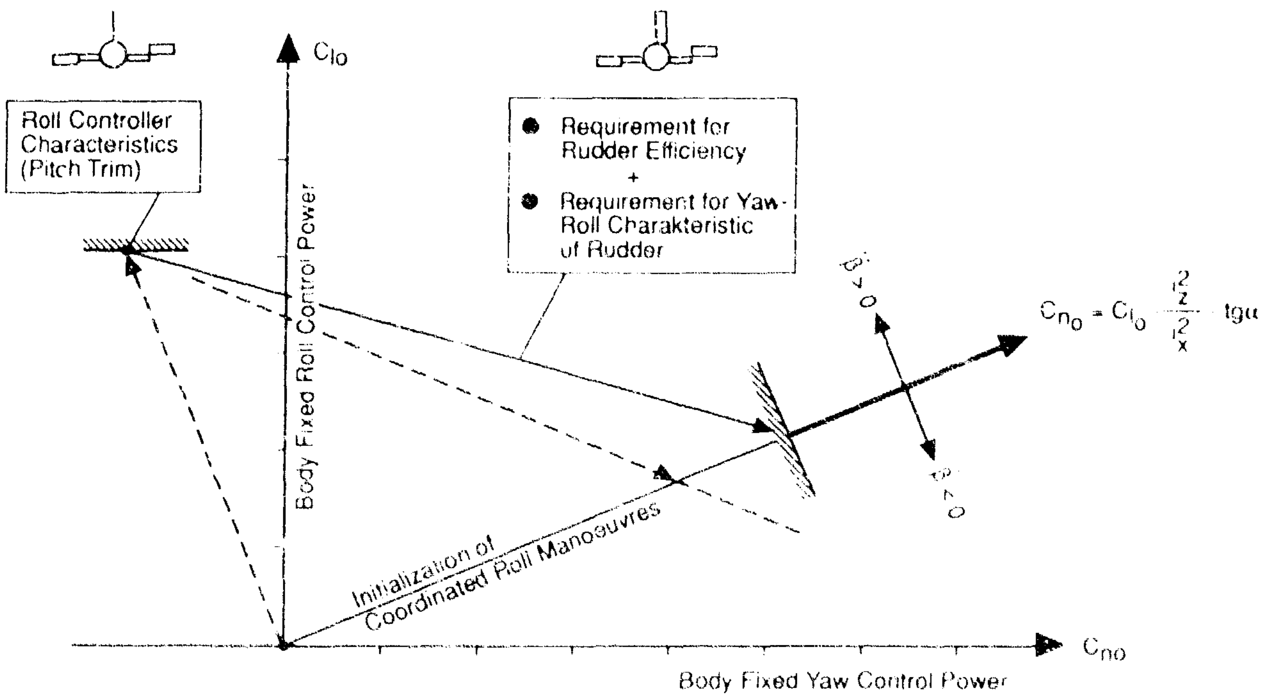
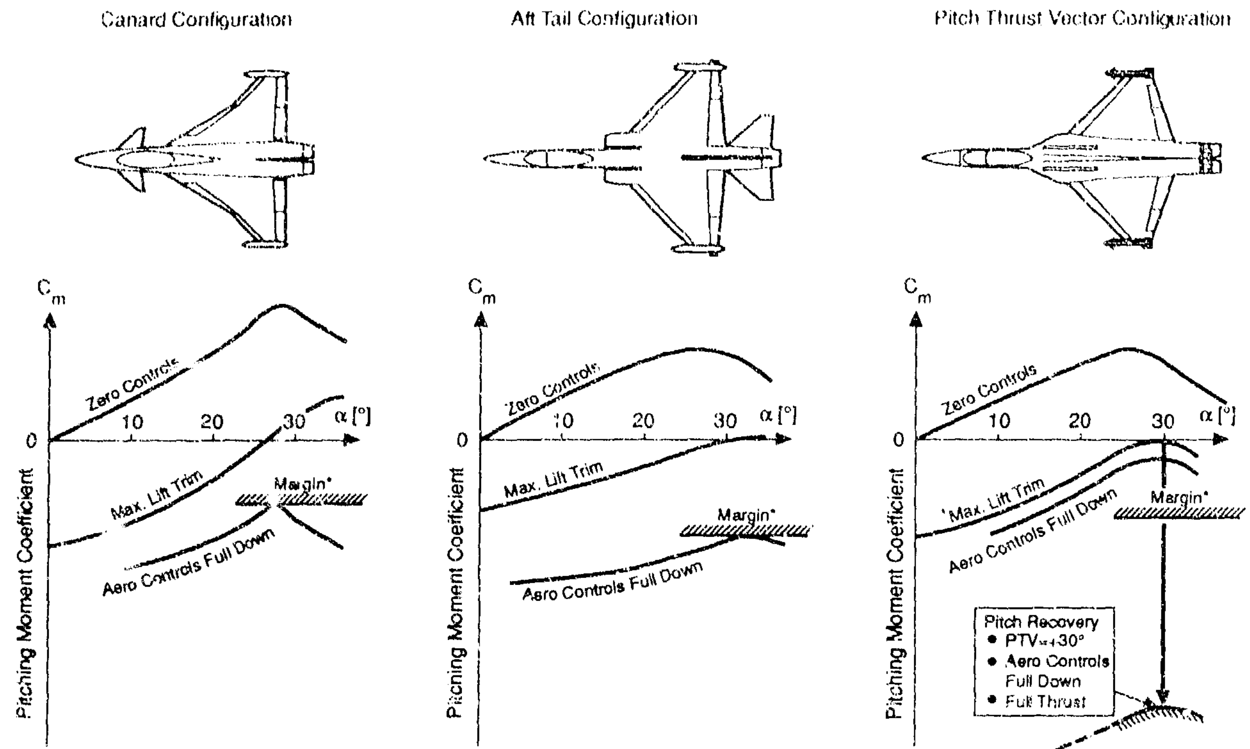


Fig. 28 Typical Pitching Moment Characteristics of Modern Fighters in the Subsonic Region (Mach = 0.3)



* Margin: $\ddot{\theta}_A \leq -0,80\ rad/s^2$ (200 KEAS: Pitch Recovery + MAX.- Rollrate = 65deg/s)

Fig. 29 Shift of Aerodynamic Centre and Time to Double T_2 (Small Angles of Attack)

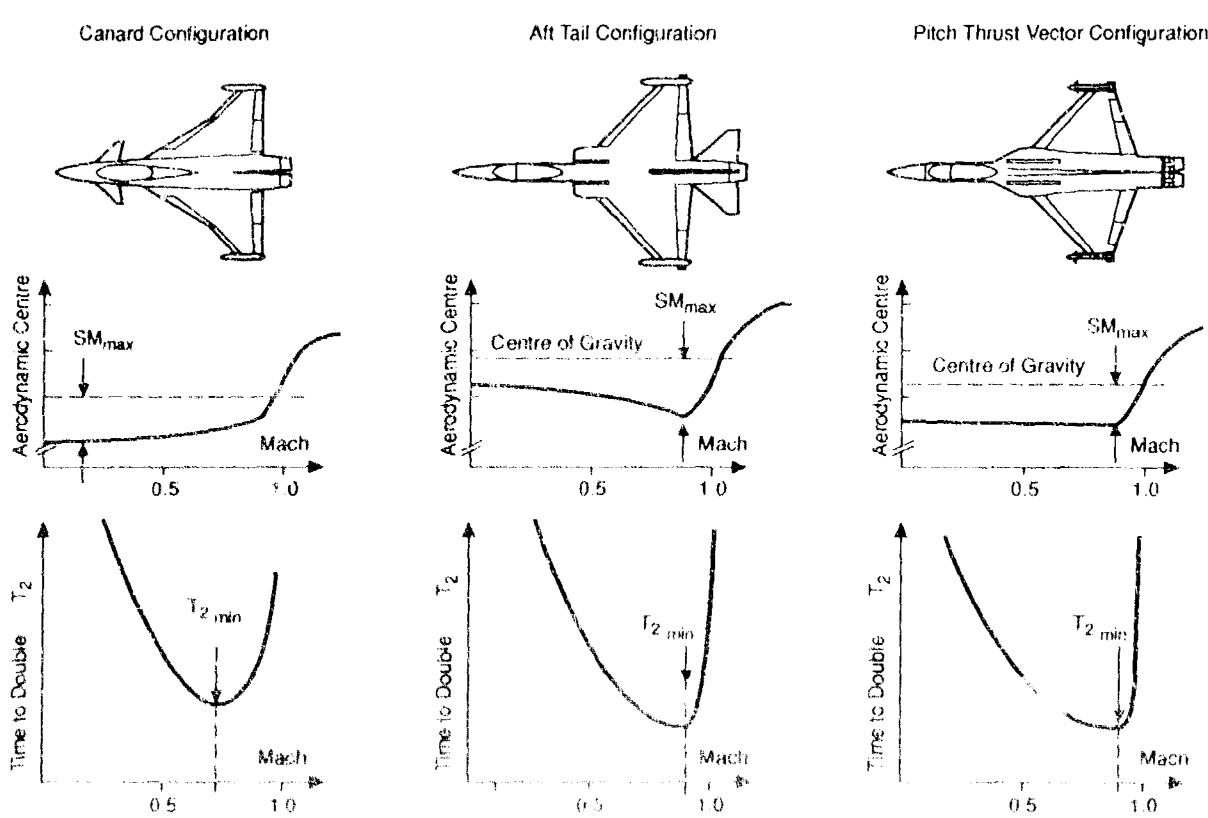


Fig. 30 Static Instability and Time to Double T_2 versus Angle of Attack

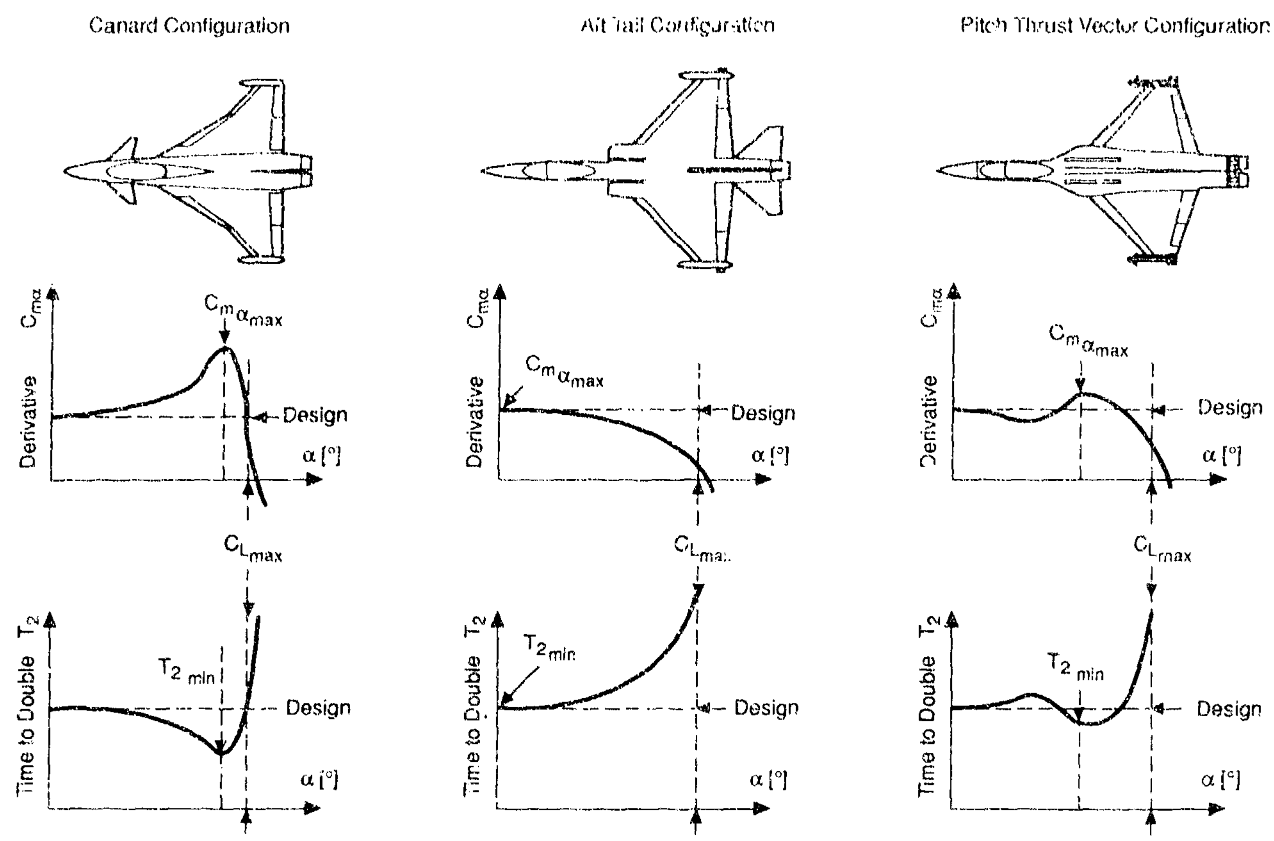


Fig. 31 Influence of Wing Apex Strakes on the Pitching Moment Characteristics

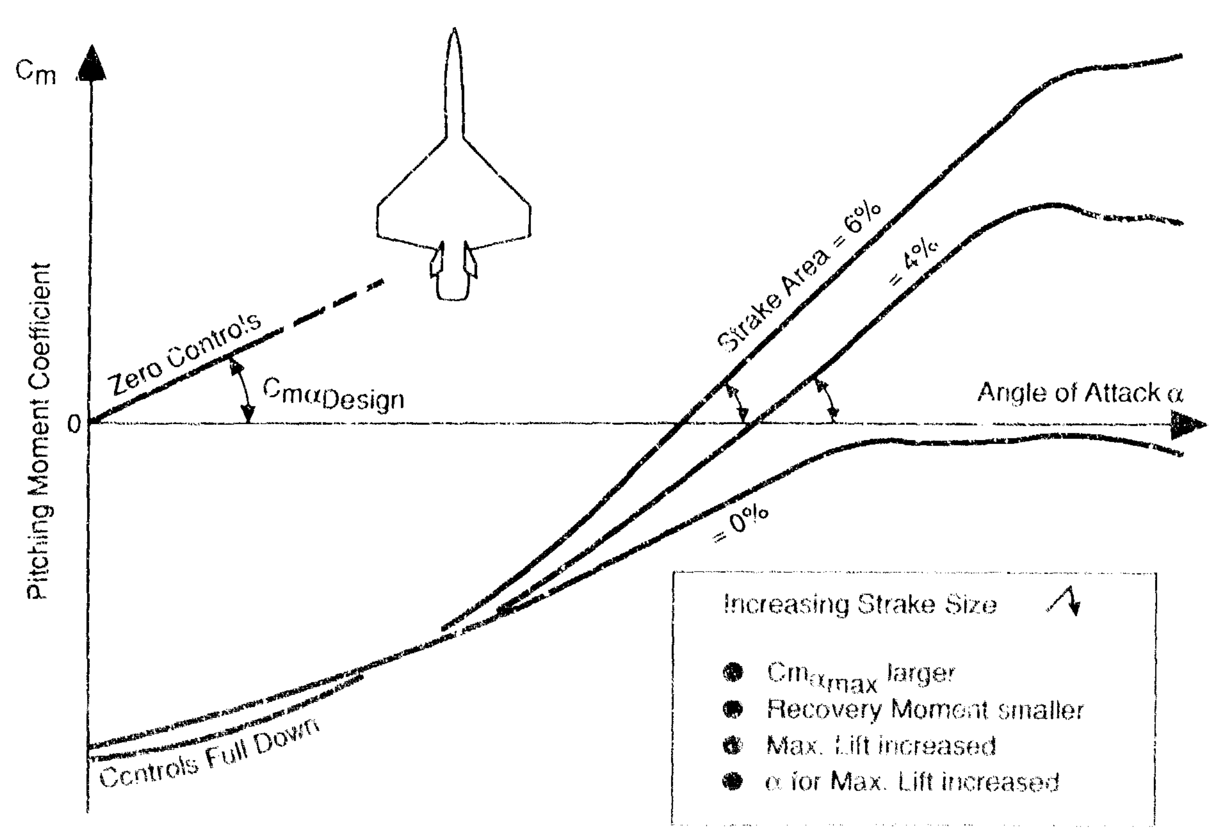


Fig. 32 Influence of Vertical Tail Arrangement on the Pitching Characteristics

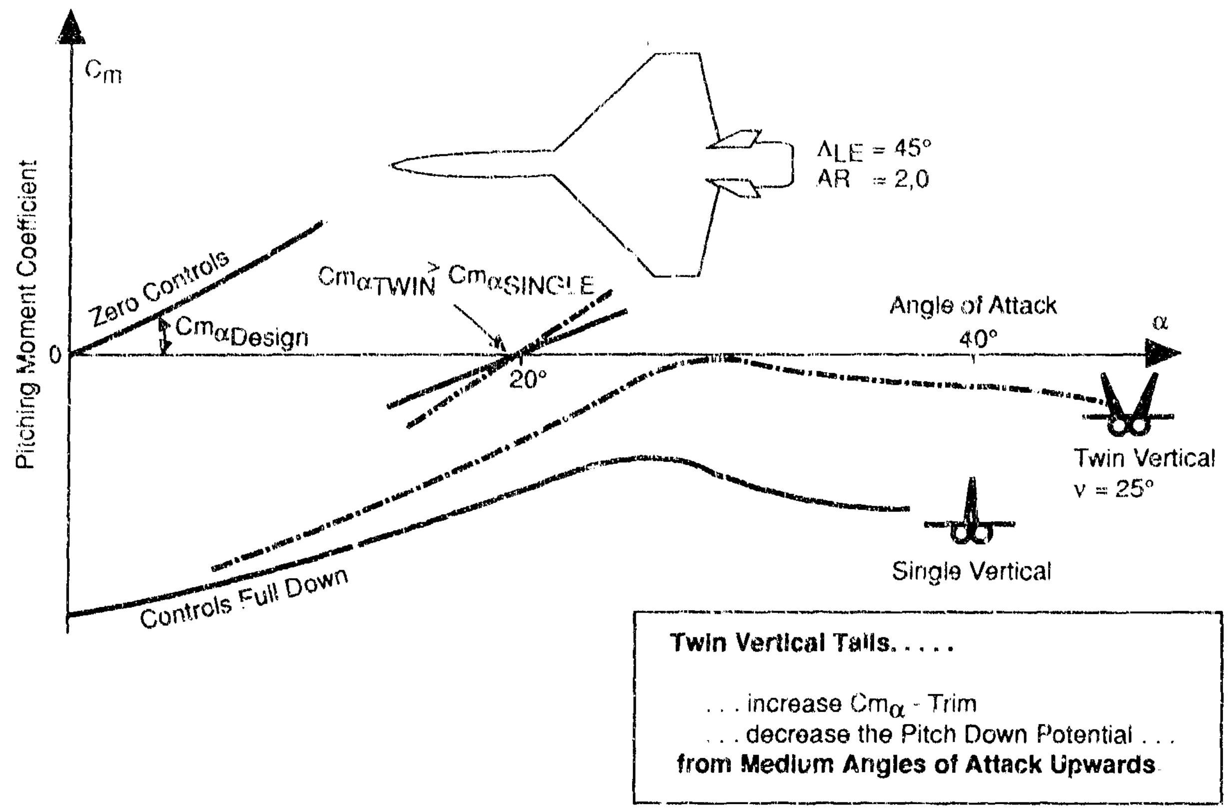


Fig. 33 Influence of Wing Planform on Pitch Behaviour (Zero Controls)

Trapezoidal-Wings: $Cm_{\alpha max}$ in Low Speed Region
Cranked Wings: Danger of Transonic Pitch-up!

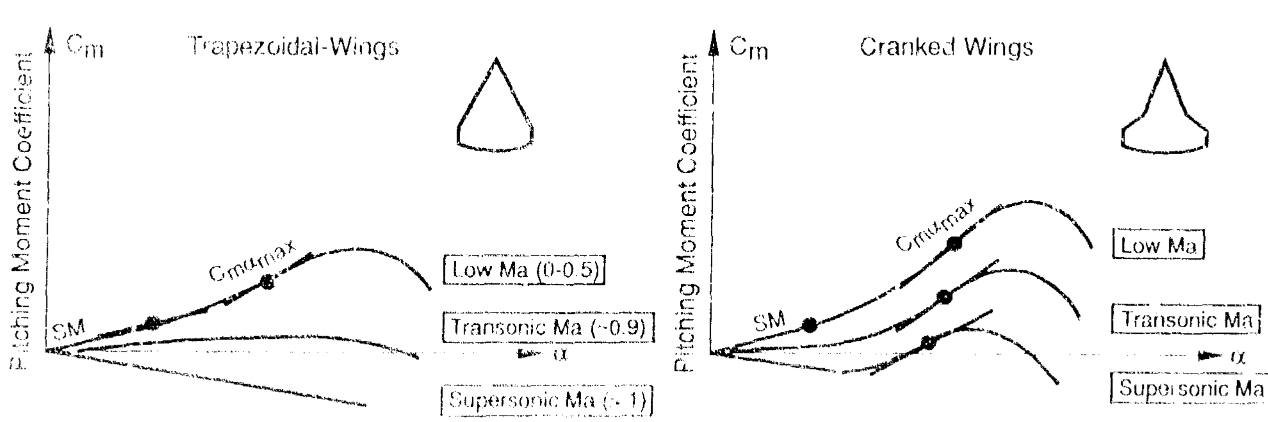


Fig. 34 Pitch-up Behaviour at Higher Angles of Attack as a Function of Wing Planform

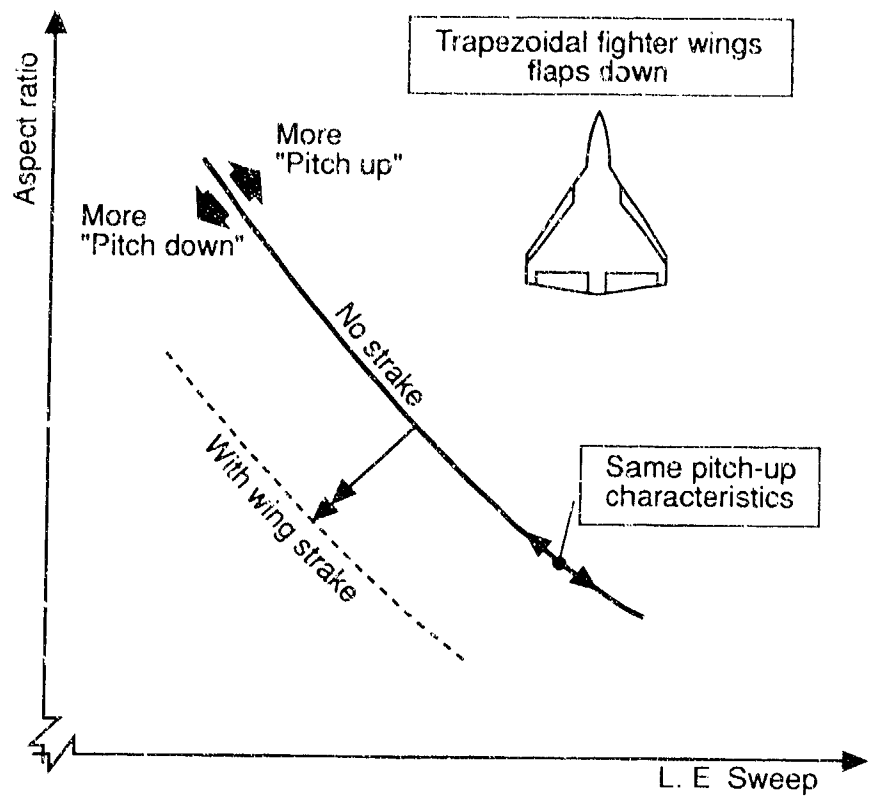


Fig. 35 Influence of Forebody Shape on Directional Stability and Yaw Damping
(Ref.: J. Hodgkinson)

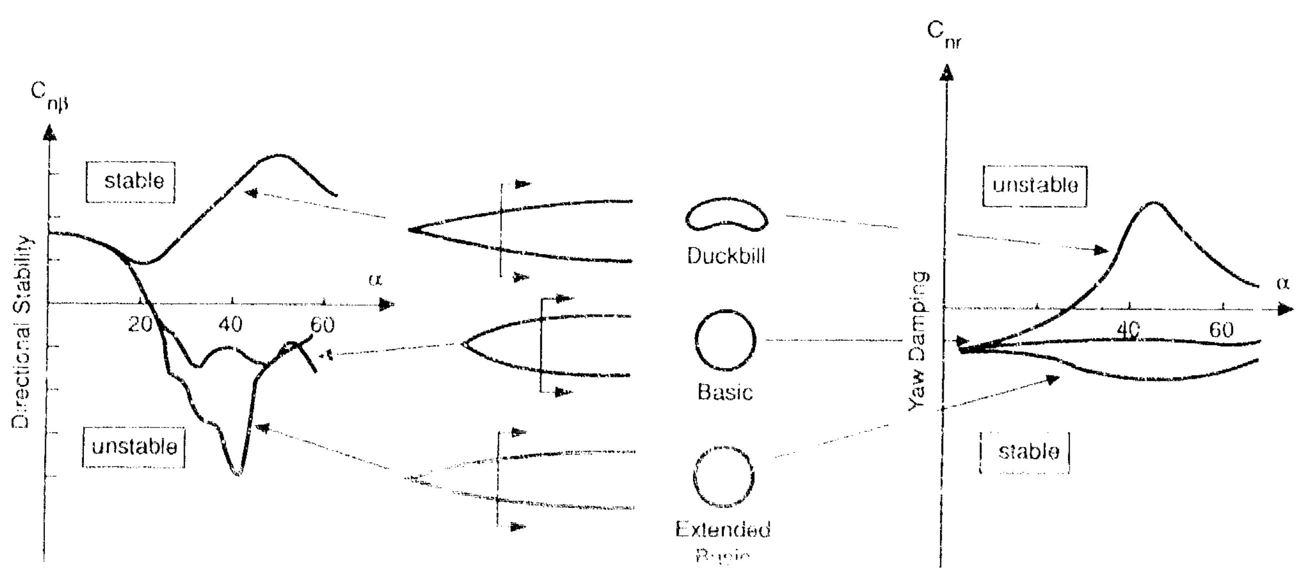


Fig. 36 Influence of Forebody Shape on Static Lateral/Directional Deriatives (β = 10°)

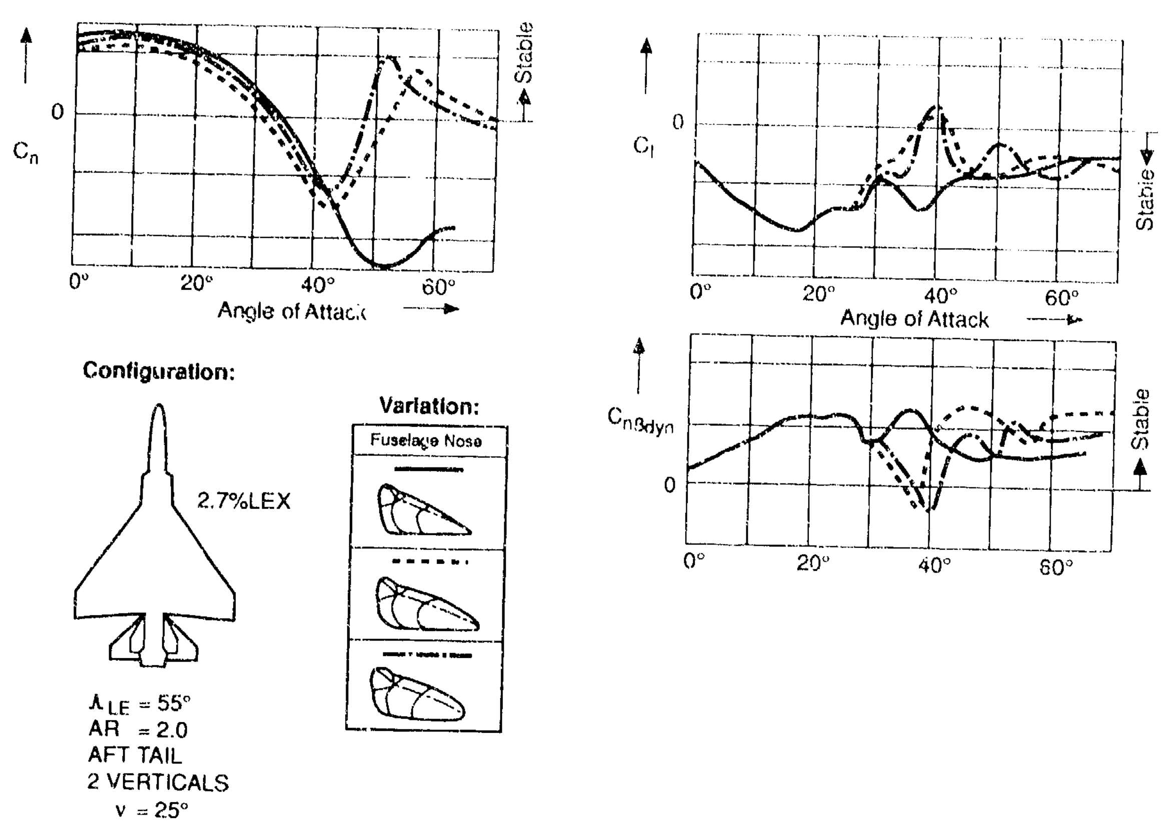
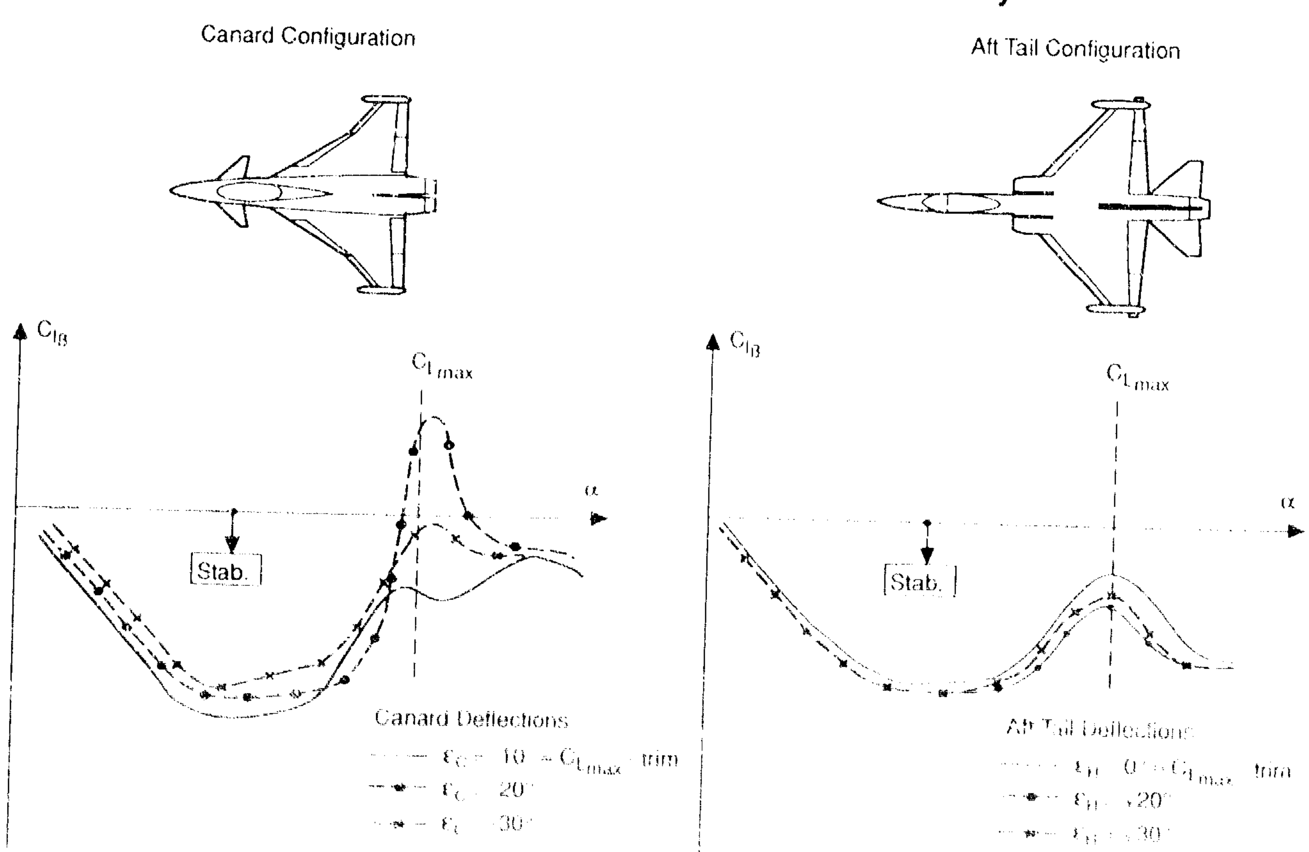


Fig. 37 Influence of Horizontal Tail Deflections on Lateral Stability



B-1B HIGH AOA TESTING IN THE EVALUATION OF A STALL INHIBITOR SYSTEM

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SUMMARY

High angle-of-attack (AOA) B-1B flight tests were conducted from September 1987 to June 1989 at Edwards AFB, California for the purpose of evaluating a flight control Stall Inhibitor System/Stability Enhancement Function (SIS/SEF). The SIS/SEF system was integrated into the basic B-1B flight control system (FCS) because of an inherent stability problem in the B-1 aircraft design. The problem lies in the fact that aerodynamically the B-1 was stability limited and not lift limited. This resulted in a lack of longitudinal stability while operating at high AOA conditions with no warning or natural cues to the pilot that the aircraft was approaching an unstable region. The B-1's stability limitations were made even more pronounced when B-1B mission requirements called for an an 82,000-pound increase in maximum gross weight over the basic B-1A configuration. To safely utilize all the available AOA at the higher gross weights demanded, some means of providing "apparent" stability was essential to provide cues to the pilot of these stability limitations. Also essential was a means of providing departure resistance to protect pilots from inadvertently entering a hazardous poststall situation. This capability was provided by the SIS/SEF system which produced a marked improvement in flying qualities and a significant increase in operational capabilities over the original B-1A FCS. This paper presents significant test results of the SIS/SEF flight test program.

BACKGROUND

Initial B-1A flying qualities tests (Reference 1) completed in May 1979, identified high AOA flight conditions where the B-1 had an inherent stability problem causing uncommanded aircraft pitchup with inadequate natural stall warning to the pilot. A follow-on B-1A flying qualities evaluation (Reference 2) completed in August 1984, provided flying qualities test results of a B-1A modified with a B-1B prototype FCS when the strategic bomber program was revived in January 1982. The B-1B mission requirements called for an 82,000-pound gross weight increase to be incorporated to increase payload and range without changing the original B-1A wing and tail design. Test results indicated that the B-1's stability problem was compounded because of the higher AOAs demanded by the heavier weight requirements with diminishing longitudinal flying qualities occurring within a significant portion of the B-1B flight envelope. Uncommanded pitchup tendencies were even more in evidence as well as unacceptable pitch stick control reversals ($dF_{stk}/dN_z < 0$) during maneuvering flight. The B-1 bomber's inherent stability problem lies in the fact that aerodynamically the B-1 was stability limited and not lift limited. That is, the maximum achievable AOA was defined by the aircraft's longitudinal stablity limitation and not by typical maximum lift stall characteristics. This was clearly made evident in B-1 wind tunnel predictions (see Figure 1) where there was ample lift available at AOAs well beyond the point at which the aircraft became longitudinally unstable.

The B-1's stability limitation was directly linked to the basic airframe aerodynamics characteristics. By electing a variable sweep wing design, the B-1 required large wing pivot joints which were blended into the wing/fuselage body interface (see Figure 2). At this point, the outer fuselage skin, commonly called the "wing glove," served not only as a good aerodynamic fairing but an excellent lifting surface in front of the main wing as well. During high AOA flight the wing tips stall first, thereby decreasing the lift component from the wings. As higher AOAs were achieved the lift component from the "wing glove" area acted as a lifting body causing the center of lift to move forward until neutral stability occurred. Also, the basic aircraft flying qualities were such that there were virtually no natural cues to the pilot that the aircraft was approaching the stability limit which made it very easy to inadvertently exceed the limit.

This lack of natural cues and possible aircraft departures ultimately constrained the B-1 to a very conservative stall warning margin to protect pilots from inadvertently entering a hazardous poststall situation. Stall warning was set at an AOA value of 80 percent of the way from zero lift AOA to neutral stability AOA, thus making the stall warning margin 20 percent below the neutral stability AOA. As a result, this reduced operational AOA envelope (stall warning envelope) was deemed unsatisfactory and modifications to the basic B-1B FCS were recommended (Reference 2) to improve flying qualities and to utilize the bomber's intended full design performance capability. An interim stall inhibitor system, SIS (phase I) or SIS-1, was developed and flight tested (Reference 3) and subsequently was installed in the first 18 production B-1Bs. The SIS-1 system demonstrated the ability to provide apparent stability throughout an interim B-1B flight envelope while at the same time providing more AOA capability by reducing the stall warning margin to 5 percent below the neutral stability AOA limit. Based on the demonstrated ability to implement this interim system, a follow-on SIS (phase II)/SEF, or SIS-2/SEF, system was designed and tested (Reference 4) as an improvement to SIS-1, to be retrofitted in all operational B-1Bs.

Development testing of SIS-2 began September 1987 and continued until June 1989. A SEF demonstration flight was flown in February 1988 to evaluate SEF gain scheduling up to and including the SIS-2 AOA limits. Actual SEF testing beyond the SIS-2 AOA limits began in June 1988 and represented the first time the B-1B AOA envelope was expanded beyond the SIS-2 limits.

TEST ITEM DESCRIPTION

General Aircraft

The B-1B, designed and manufactured by Rockwell International, was a long-range, aerial-refuelable, strategic bomber with the capability of high, subsonic flight at low altitude. The B-1B incorporated a blended wing-body concept with variable-sweep wings, a single vertical stabilizer with a three-section rudder, and all-moving horizontal stabilators which operated independently to provide both pitch and roll control. The wings were equipped with flaps, slats, and spoilers which provided roll control and also functioned as speed brakes. A structural mode control system (SMCS) reduced longitudinal and lateral structural bending oscillations through movable, canted vanes on each side of the forward fuselage. The aircraft was designed for a crew of four: pilot, copilot, offensive systems officer, and defensive systems officer.

The takeoff/landing or power approach (PA) configurations consisted of flaps, slats, and landing gear extended with a 15- or 20-degree wing sweep; the cruise (CR) configurations were with a 15-, 20-, 25-, 55-, or 67.5-degree wing sweep with the flaps, slats, and landing gear retracted, as defined in Table 1 below.

Table 1
CONFIGURATION DEFINITION

Configuration	Wing Sweep Position (deg.)	Flap, Slat[1] and Gear Position
15 PA	15	Down
20 PA	20	Down
15 CR	15	Up
20 CR	20	Up
25 CR	20	Up
55 CR	25	Up
67 CR	67.5	Up

[1]Slat position up is retracted and down is extended.

The B-1B was powered by four General Electric F101-GE-102 dual rotor, augmented turbofan engines in the 30,000-pound thrust class. The engines were mounted in twin nacelles below the wing near the left and right wing pivot points. The engine air inlets were a fixed-geometry design.

The aircraft was equipped with integral fuel tanks in the fuselage, wing carry through, and wing outer panels. Automatic fuel transfer sequencing provided cg control.

Basic Flight Control System

The B-1B FCS was a combined hydromechanical system integrated with the electrical stability control augmentation system (SCAS), including SIS-2/SEF and the automatic flight control system (AFCS). The primary FCS provided position command to the flight control surfaces in all three axes based on mechanical inputs from the pilot's control stick and rudder pedals, and electrical control stick inputs and stability feedbacks through the SCAS. The secondary FCS included the pilot's controls and actuating devices for variable wing sweep, high-lift surfaces (slats and flaps), SMCS, and deceleration control (symmetrical spoilers). The SIS-2/SEF description included in the following discussion was the final version flight tested and will be referred to as the SIS/SEF system throughout the remainder of this paper. For more detail of the entire flight control system, refer to References 5, 6, 7, and 8.

Stall Inhibitor System/Stability Enhancement Function

The SIS/SEF system was designed as an "add on" to the basic flight control system which consisted of a combined hydromechanical system and an analog electrical SCAS. The SIS/SEF system functioned solely through the pitch SCAS and was designed to operate in two modes: SIS and SEF. The SIS mode was designed to enhance B-1B flying qualities at AOAs up to and including neutral aerodynamic stability. The SEF mode provided additional stability to allow the aircraft to be flown at AOAs beyond neutral stability (anywhere from 1- to 4-degrees AOA above neutral stability depending on aircraft configuration and flight condition) with sufficient pitch control power remaining to safely return the aircraft from excursions beyond the SEF AOA limits. Both modes operated using the same hardware control configuration; however, SEF utilized higher AOAs and therefore imp[illegible] higher gains. The SIS/SEF system was design[illegible] to [illegible] fail operational in the SEF envelope and fail safe in the SIS envelope. The SEF mode was the normal mode of operation with manual reversion to the SIS mode selected for certain failure states. The function of SIS/SEF was to protect the aircraft from exceeding predetermined AOA limits and providing increased apparent stability in neutrally stable or statically unstable regions to achieve an expanded maneuver capability consistent with expanded B-1B mission objectives. Apparent stability was achieved by utilizing an AOA feedback signal to reduce trailing edge up (TEU) horizontal stabilator commands as a function of AOA and AOA rate. Pilot stick force cues were increased by making stick inputs less effective at commanding TEU stabilator deflection as the aircraft approached the AOA limit, thus increasing the aircraft's apparent stability. In order to improve aircraft flying qualities while limiting AOA, SIS/SEF attempted to linearize stick force gradients below stall warning and considerably increase stick force gradients above stall warning. An example of this desired system response is shown in Figure 3 which depicts a 1-g level deceleration maneuver performed at 20 CR, aft cg to the SEF limit (9.5 degrees AOA). Also, note the unstable characteristics of the aircraft as shown by the horizontal stabilator moving in the trailing edge down (TED) direction as the AOA increases from 7.5 degrees to the SEF limit.

Angle-of-attack was determined from pressure differentials on the pitot-static probes. Three sets of side-mounted pitot-static probes accompanied by three sets of digital pressure transducers provided air data information to two Central Air Data Computers (CADCs) for redundancy. From the air data and aircraft configuration information provided, the CADCs independently computed a "standardized AOA" (α_s).

The term "standardized AOA" was a signal processed by the SIS/SEF control laws to provide a means to standardize the AOA of the AOA feedback signal to account for the changing stability limitations with varying wing sweep configuration and flight condition. Thus, regardless of aircraft configuration, the α_s signal had a value of 1.0 at the corresponding SIS or SEF AOA limit and a value of 0.0 at zero lift. This is illustrated in Figure 4 for SIS α_s. Stall warning SIS-OFF was 0.8 α_s and for SIS- or SEF-ON it was 0.95 α_s.

In order to compute α_s, the CADCs each stored AOA limits for both the SIS and SEF envelopes and used the values corresponding to the position of the three-position SIS/SEF select switch located in the cockpit. With the switch in the SIS-OFF or SIS-ON position, SIS AOA limits were used in the computations. With the switch in the SEF-ON position, the SEF limit was used in the computations. The CADCs also calculated the onset of proportional feedback in the SIS/SEF by use of the α_{SSE} parameter. The onset for proportional feedback was designed to occur at the same true AOA independent of the SIS/SEF mode selected. This AOA was approximately equal to 60 percent of the SIS AOA envelope. Proportional feedback gain (K_{SEF}) was also computed by both CADCs and was a function of wing sweep, cg, and dynamic pressure, and is presented in Figure 5.

The SIS/SEF system functioned through three primary control paths:

1. Proportional stability path,

2. Mechanical pitch stick cancellation, and

3. Electrical pitch stick cancellation.

Figure 6 shows the SIS/SEF control paths and the interfaces with the CADC and SCAS. The mechanical cancellation path was from the pitch stick position (x_θ) to $\delta H_{SIS(M)}$. The electrical cancellation path was from x_θ to "SIS/SEF," and the proportional stability path was from α_s (from the CADC) to "SIS/SEF."

The proportional stability path utilized the α_s signal calculated by the CADCs to provide increased apparent stability and AOA limiting by feeding back pitch down commands to the SCAS servo in response to increased changes in AOA; independent of pitch stick position. Initially, the α_s signal from the CADC was enhanced by a term that was proportional to the rate of change of α_s and was referred to as α_s^*. Then, the bias α_{SSE} (computed from the CADC) was summed with α_s^* to provide the onset for SIS/SEF proportional feedback to become active. The biased α_s^* was then multiplied by the stability enhancement nonlinear gain (see Figure 7) that was proportional to the α_s^* magnitude. This gain was a four slope function that provided higher gain as α_s^* increased. Thus, as the AOA increased above the onset, additional gain compensated for the inherent decrease in stability as the aircraft approached and exceeded neutral stability. Pitch rate damping was provided for wing sweeps between 22 and 63 degrees (see Figure 8) via the pitch rate feedback path. Finally, the proportional stability path was gain-conditioned or tailored to the particular aircraft configuration with the K_{SEF} value to produce the net proportional feedback gain for "artificial" or "apparent" stability, i.e.,

$$C_{M_{\alpha_{SIS}}} = C_{M_\alpha} + K_{\alpha_S^*}\alpha_S^*$$

(where $K_{\alpha_S^*}$ represents the net proportional feedback gain). Thus, the proportional control path provided the aircraft with apparent stability when flying in a neutral or unstable region.

Providing additional safety from departure were two more control paths which could potentially add to the overall "SIS/SEF" pitch down command signal of the proportional stability path. These control paths were the integrator and the savior paths. They were designed to prevent excessive AOA excursions above the SIS/SEF limits if abrupt pilot inputs or atmospheric turbulence cause the AOA to increase. The integrator path (labeled "SIS INTEG" in Figure 6) provided TED stabilator command whenever 1.05 α_s^* was exceeded. This loop could provide TED command with a rate of up to 4 degrees per second (see Figure 9) with a maximum command of 10 degrees. Once α_s^* was decreased below 0.95, the integrator synchronizer would fade the integration signal as a 10-second time constant. The savior path (labeled "SAV" in Figure 6) served as the final TED command path whenever α_s^* exceeded 1.3.

This path provided a direct feedback of up to a maximum of 4 degrees TED stabilator command (see Figure 10).

The mechanical stick cancellation path was designed to restrict TEU stabilator command authority by aft pitch stick inputs. Cancellation was provided for aft stick positions greater than 1.25 inches by means of a four slope function (see Figure 11). This cancellation function attempted to model the slopes of the nonlinear gearing curve with an offset provided to allow for a reduced mechanical command signal to pass to the pitch SCAS servos. Similarly, the electrical stick cancellation path provided a TED stabilator command which was equal to 25 percent of the electrical SCAS pitchup command (see Figure 12) for all AOAs below 0.95 α_s^*.

Working in concert with the two stick cancellation paths was the AOA command limiter (see Figure 13). The function of the command limiter coupling with the stick cancellation paths was to provide 100 percent electrical aft stick cancellation and approximately 90 percent mechanical aft stick cancellation when the AOA exceeded 0.95 α_s^* (in other words, above stall warning). The remaining 10 percent mechanical command authority was provided so that the AOA limit could be achieved under a limited maneuvering margin, although aft pitch stick force gradients would be increased beyond stall warning. This increase in pitch stick cancellation was accomplished by means of an integrator (A13I in Figure 13) that followed the pitch stick displacement. Pitch stick command signals from the SCAS were sent to this integrator via an integration loop where stick error signals (the difference between $X_{\theta LIMIT}$ and A13I) were driven to zero below 0.95 α_s^*, resulting in no cancellation contribution (MEAHL and ELAHL = 0) from the AOA command limiter going to the SCAS servo with switch A13 (SWA13) in the closed position (state 1.0). However, when α_s^* was greater than or equal to 0.95, SWA13 would move to the open position (state 0.0) at which moment the integrator value freezes. This frozen state of the integrator value when SWA13 changed to the open position was referred to as the point at which the command limiter became "latched." In this state, a pitch stick error signal was calculated based upon the difference between the current stick position and the stick position required to achieve 0.95 α_s^*. Any additional aft stick commands then resulted in the error signal being sent to two control paths: the mechanical stick cancellation path (MEAHL) and the electrical stick cancellation path (ELAHL). In the ELAHL path, the stick error signal was conditioned with a stick-to-tail cancellation gain of 0.75, with a net result of 100 percent electrical stick cancellation when combined with the initial 0.25 cancellation gain of the electrical stick cancellation path. In the MEAHL path, the stick error signal was conditioned with a gain of 0.7 degree stabilator/inch of aft stick, resulting in an additional 70 percent mechanical stick cancellation above 0.95 α_s^*. To "unlatch" the AOA command limiter or to return SWA13 to the closed position, two conditions must be satisfied:

1. AOA must be decreased such that the α_s^* is below 0.95.

2. Pitch stick command must be returned within 0.75 inch of the position where the system originally became "latched."

The 0.75 inch stick position bias was an outcome of a simulator evaluation to provide acceptable SIS/SEF system performance during inflight encounters with turbulence.

The sum of all SIS/SEF control paths, excluding the mechanical cancellation command, were gain scheduled with Mach number and altitude in the pitch SCAS exactly as the original pitch SCAS feedback signals (see Figure 6). This command was then summed with the mechanical cancellation command. The total signal was transmitted to the pitch SCAS servos and subsequently to the horizontal stabilator actuators. The pitch SCAS servos were limited in command authority to $\pm$ 29.6 degrees.

FLIGHT TEST AND EVALUATION

Test Plan Approach

The complexity of SIS/SEF, coupled with the number of aircraft configuration combinations (i.e., wing sweeps from 15 to 67 degrees and a cg range from 10 to 48 percent MAC), made the flight test program a difficult and lengthy one. Aircraft configurations were limited to previously determined optimized wing sweeps of 15, 20, 25, 55, and 67 CR with takeoff, approach and landing done only at 15 and 20 PA for a total of seven different configurations. For each wing sweep, cgs were tested at the nominal or "target" cg, and the forward and aft cg limits. No spin recovery device was practical on an aircraft the size of the B-1, therefore flight testing was performed in a very conservative build-up manner.

Since the test aircraft was used for radar avionics and terrain-following testing as well as for SIS/SEF flying qualities evaluation, a noseboom for measurement of AOA could not be installed, although desirable for high AOA testing. So, the production pitot-static system, which determined AOA from pressure differentials on the pitot-static probes, was employed.

Prior to first flight, the SIS/SEF system was evaluated on a pilot-in-the-loop simulator by a panel of five test pilots to determine its readiness for flight. Also, the simulator evaluation was invaluable for failure-state analysis, and prediction of system performance and limiting characteristics. Furthermore, the SIS/SEF system's success in markedly improving flying qualities and aircraft capabilities over the original SCAS-only (i.e., SIS-OFF) B-1B flight control system was attributed, in large part, to detailed analyses and evaluations that were performed on the simulator before and during flight test.

Testing was accomplished throughout the B-1B flight envelope covering literally thousands of test points with speeds ranging from the minimum airspeed based on the angle-of-attack (AOA) limit, to 1.2 Mach number, and with altitudes from 2,500 to 27,000 feet pressure altitude. Structural load factor limitations were 3 gs for the 67 CR configuration, and 2 gs for all other configurations. Testing emphasized the slow speed and high AOA/N_z side of the envelope and included level accelerations and decelerations; windup turns; straight pull-ups/push-overs, bank-to-bank rolls; steady-heading sideslips; pitch, roll and yaw doublets; speed brake extensions (which changed the stability); wing sweep reconfigurations; takeoffs and landings; aerial refueling tanker boom tracking and refueling up to maximum gross weight; and automatic terrain following.

Of primary interest and of final proof of the SIS/SEF system's capability, were the maneuvers designed to test the system AOA limiting characteristics. These consisted of 1-g, constant altitude decelerations and constant mach or airspeed windup turns to the AOA limits at which point the aircraft was stabilized and abrupt step pitch stick inputs were applied by approximately 1 inch and held constant. The purpose was to evaluate the capability of SIS/SEF to limit AOA to an abusive control input.

In the interest of safety and the fact that aircraft departure from controlled flight could not be tolerated, a build-up approach was taken in testing the system limiting performance. In addition, all test points were preflown in a pilot-in-the-loop simulator to provide pilot familiarization, as well as predictions of aircraft response. Initially, limiting (via step inputs) was evaluated with SIS-ON at the SIS AOA limits across the B-1B envelope, since previous B-1 testing (References 2 and 3) had expanded the AOA envelope to those limits. However, SEF-ON testing constituted the first time the B-1B AOA envelope was to expand beyond neutral stability. Therefore, rather then proceeding directly to the SEF limit, the AOA was increased in 1-degree increments with data analysis performed between flights before progressing to the next increment or SEF limit. If the data analysis was within prediction and the handling qualities were acceptable at the SEF limits, then the test points were cleared for step inputs to evaluated AOA limiting with SEF-ON.

Since a departure could not be tolerated for obvious reasons, the termination criteria specified in Table 2 was critical. The primary means in the control room used to call termination or continuation of the maneuver was the horizontal stabilator versus AOA cross plot compared real time with a plot of the same maneuver from pilot simulation. There were also specification requirements of not exceeding 1.5 degrees AOA overshoot which applied only to the maneuvers with a 1-inch pitch step input to evaluating limiting characteristics. Test points were performed at target cg prior to forward cg, and then forward cg prior to aft cg. For each test condition, 1-g decels were accomplished first, followed by windup turns.

Table 2
TERMINATION CRITERIA

The maneuver was terminated for:
1. A rapid change in the $\frac{\Delta\delta_H}{\Delta\alpha}$ gradient.
2. Horizontal stabilator equal to 3 degrees TED (this provided 7 degrees of remaining TED command authority).
3. Stabilator SCAS servo equal to 0.9 inch TED (this was 50 percent of the maximum servo travel).
4. A difference of 2 or more degrees of horizontal tail between flight test and predicted data.

Flying Qualities Results

Longitudinal speed stability was positive throughout the B-1B envelope; however, the degree of stability varied among the wing sweeps and configurations. The 15 and 20 PA, and 55 CR configurations provided satisfactory speed stability; 15, 20, and 25 CR provided light but satisfactory speed stability; and the 67 CR configuration provided satisfactory speed stability below stall warning, but did not demonstrate the desired increase in stick force gradients beyond stall warning as per design requirements. Overall, longitudinal speed stability was found to be acceptable.

Longitudinal maneuvering stability was also positive throughout the B-1B envelope with SIS/SEF significantly improving stick force per g flying characteristics. However, as in longitudinal speed stability, 67 CR provided no apparent increase in stick force gradients beyond stall warning. The SIS/SEF system eliminated all characteristics of zero pitch stick force gradients or control reversals as identified in previous B-1A testing. The significance of this is illustrated in Figure 14 which shows a comparison of stick force per g with and without SIS/SEF capability for the same 55 CR windup turn maneuver performed at similar test conditons.

Angle-of-Attack Limiting Characteristics

The SIS/SEF system provided improved departure resistance and provided adequate AOA limiting at most flight conditions for pitch stick step inputs applied at the AOA limit. Figure 15 illustrates the desired time history response to a step input, an AOA overshoot less than 1.5 degrees which was well damped returning the aircraft back to the AOA limit with no oscillatory or secondary undershoot behavior. Generally, the AOA response for all step inputs tested was damped within one cycle.

AOA limiting characteristics in the SEF mode were not acceptable throughout the 67 CR flight envelope. The short term solution was to add restrictions to the B-1B Flight Manual for 67 CR flight. However, this solution was unsatisfactory for the long term and it was recommended that the SEF mode of the SIS/SEF system should be modified, in order to provide a sufficient level of departure resistance in the 67 CR configuration. Limiting was unsatisfactory for 67 CR at 18,000 and 22,000 feet pressure altitude due to excessive AOA overshoots on the order of 2.5 to 3.5 degrees above the SEF limit. Limiting was not tested for 67 CR at aft cg, 0.85 Mach number, 8,000 and 15,000 feet because excessive pitch oscillations occurred above stall warning. These cases were indicative of the SIS/SEF gains being too high.

Limiting testing revealed that SIS/SEF AOA limiting could be defeated if aggressive pilot inputs (i.e., large or abrupt aft pitch stick inputs) were commanded just <u>below</u> stall warning since the AOA command limiter did not become activated or "latched" until 0.95 α_s^*, as discussed previously in the SIS/SEF Description Section. As a result it was possible that abrupt stick inputs at AOAs <u>below</u> stall warning could cause excessively high AOA excursions because the input would go to the horizontal stabilator only partially cancelled before the aircraft AOA had increased enough to cause the command limiter to "latch" (SWA13 changing to the open position). Aircraft responses to step inputs <u>below</u> stall warning were unsatisfactory since it was possible to reach a high AOA (as shown in Figure 16 where the AOA reached 2.5 degrees above the AOA limit) before the command limiter could react and provide additional cancellation of the input. For normal pilot inputs this was not a problem, but was a problem for aggressive or abrupt inputs. This was not unique to any one wing sweep or flight condition and requires further investigation.

A related area of concern for the command limiter was the degraded handling qualities or secondary uncommanded pitchup during certain pitch maneuvering near the AOA limit. Figure 17 shows a time history response to a small step input (approximately 0.5 inch) that increased AOA above 1.05 α_s^* causing the integrator path to become active and reducing the AOA below stall warning. This, in turn, resulted in the AOA command limiter becoming "unlatched" without a change in stick position (stick position was held constant following the step input), sending a pitchup command transient to the horizontal stabilator causing a secondary AOA overshoot. As mentioned in the SIS/SEF Description Section, in order to "unlatch" the command limiter: (1) the pitch stick must be within 0.75 inch of the position at which α_s^* exceeded 0.95, and (2) α_s^* must be below 0.95. If, however, after the AOA had exceeded 0.95 α_s^*, the pitch stick was moved further aft, but less than 0.75 inch from the position where it orginally became "latched" and the AOA subsequently dropped below 0.95 α_s^*, the command limiter will "unlatch" since both conditions to "unlatch" were satisfied. The stick error signal (difference between $X_{\theta LIMIT}$ and A13I) was seen by the SIS/SEF system as a momentary decrease in stick cancellation, resulting in a pitchup command to the stabilator until the stick error signal was again driven to zero. The maximum pitchup transient would be the equivalent of 0.75 inch of pitch stick command.

This "unlatch" characteristic could also occur while flying in turbulence near the AOA limit as shown in Figure 18. In this case, the command limiter became "latched" at stall warning while the pilot continued applying slightly more aft stick (but less than 0.75 inch) to stabilize the aircraft at the AOA limit. However, before the pilot could stabilize at the limit, the aircraft experienced a momentary pitch down gust causing the command limiter to "unlatch" resulting in a pitchup command transient to the stabilator. Subsequently, the resulting AOA overshoot was large enough to activate the integrator path to pitch the aircraft down a second time. Obviously, this characteristic would become even more pronounced if the pilot applied abrupt or large aft stick inputs just <u>below</u> stall warning to counter a sudden pitch down gust.

Test Technique (Operational Maneuver)

In response to the concern of aggressive pitch stick inputs below stall warning, an operational maneuver was used to investigate whether problem areas existed that had not surfaced during the use of standard testing techniques. The AOA limiting characteristics of the system were evaluated by applying a 1-inch pitch step input held constant after the aircraft had been stabilized at the AOA limit. System response with AOA overshoots and oscillations was evaluated to determine system specification compliance. This approach took advantage of design features of the limiter and was not the most likely scenario to be used by the operational command. Thus, the objective of the operational maneuver was to evaluate the effectiveness of stall warning scheduling and to have the pilot assault the AOA limiting features of SIS/SEF by commanding pitch inputs slightly <u>below</u> stall warning.

Test conditions were selected for each of the wing sweep configurations and were flown at the corresponding nominal mission flight conditions and payloads. Testing was accomplished initially by performing a level turn and stabilizing the aircraft at an AOA just below stall warning. Once stabilized, the pilot rolled aggressively to wings level and then performed a pullup maneuver to between 10 and 20 degrees of pitch attitude. During the pullup maneuver, the pilot was to be aggressive with the system but to always adhere to stall warning.

Pilot comments indicated that for both SIS and SEF, and the forward wing sweeps (15 CR, 15 PA, 20 CR, 20 PA and 25 CR), only small maneuver margins existed below stall warning. However, no large overshoots occurred as long as stall warning was observed and the aircraft was controllable. For the aft wing sweeps (55 and 67 CR) at the nominal flight conditions flown, stall warning could not be achieved without exceeding the structural limits of the aircraft and no problem areas were noted.

CONCLUSIONS

Overall, the B-1B flight control system modified with the Stall Inhibitor System/Stability Enhancement Function system provided acceptable flying qualities throughout most of the B-1B operational envelope. The SIS/SEF system produced a marked improvement in flying qualities and aircraft capabilities over the original SCAS-only (i.e., SIS-OFF) B-1B FCS. The SIS/SEF system provided a significant increase in operational capabilities, which have been constrained by stability limitations rather than lift limitations. This improved system provided sufficient apparent stability and departure resistance to fly safely near maximum lift. The addition of SIS/SEF also provided an increase in the use of the operational AOA envelope (stall warning envelope) from 0.80 α_s with SIS-OFF, to 0.95 α_s with SIS- or SEF-ON. With the increased AOA envelope provided by SIS/SEF, two areas of improved aircraft capability were the B-1B's abilities to aerial refuel and to terrain follow at gross weights up to 100,000 pounds heavier over the original SCAS-only (i.e., SIS-OFF) B-1B FCS (see Figures 19 and 20).

The modified B-1B FCS, with SIS engaged, provided acceptable flying qualities including adequate stick force cues for positive speed stability, positive maneuvering stability, and departure resistance throughout most of the B-1B stable AOA envelope. The SEF mode also provided the same acceptable speed stability, maneuvering stability, and departure resistance throughout the expanded AOA envelope (full operational envelope) for all wing sweeps and configurations tested, except 67 CR. Testing at 67 CR revealed unsatisfactory AOA limiting at the SEF AOA limit at altitudes greater than 15,000 feet due to excessive AOA overshoots (greater than 2 degrees for aft stick step inputs). In addition, AOA limiting was untested at 67 CR at altitudes less than 15,000 feet at aft cg due to oscillatory behavior above stall warning. Testing had revealed that SIS/SEF was not a foolproof sytem and that its ability to limit AOA could be defeated if aggressive pilot inputs (i.e., large or abrupt aft pitch stick inputs) were commanded just below stall warning. The response to these inputs as a result of the "latch/unlatch" characteristics of the AOA command limiter was a large overshoot in AOA with a possible departure from controlled flight if stall warning was not observed. This was not unique to any one wing sweep or flight condition and will require crew training to ensure B-1B pilots are aware of these characteristics.

Longitudinal stability limitation is a potential problem that exists with highly mission specialized aircraft such as the U.S. F-111 fighter and B-1B bomber because of basic aerodynamic design. Futhermore, it can be compounded or made even more pronounced by new mission or growth requirements as evident in this paper. SIS/SEF systems will always be needed, but should be incorporated into the initial flight control system design.

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WING SWEEP: 67 DEG, CRUISE
C.G.: 25% MAC

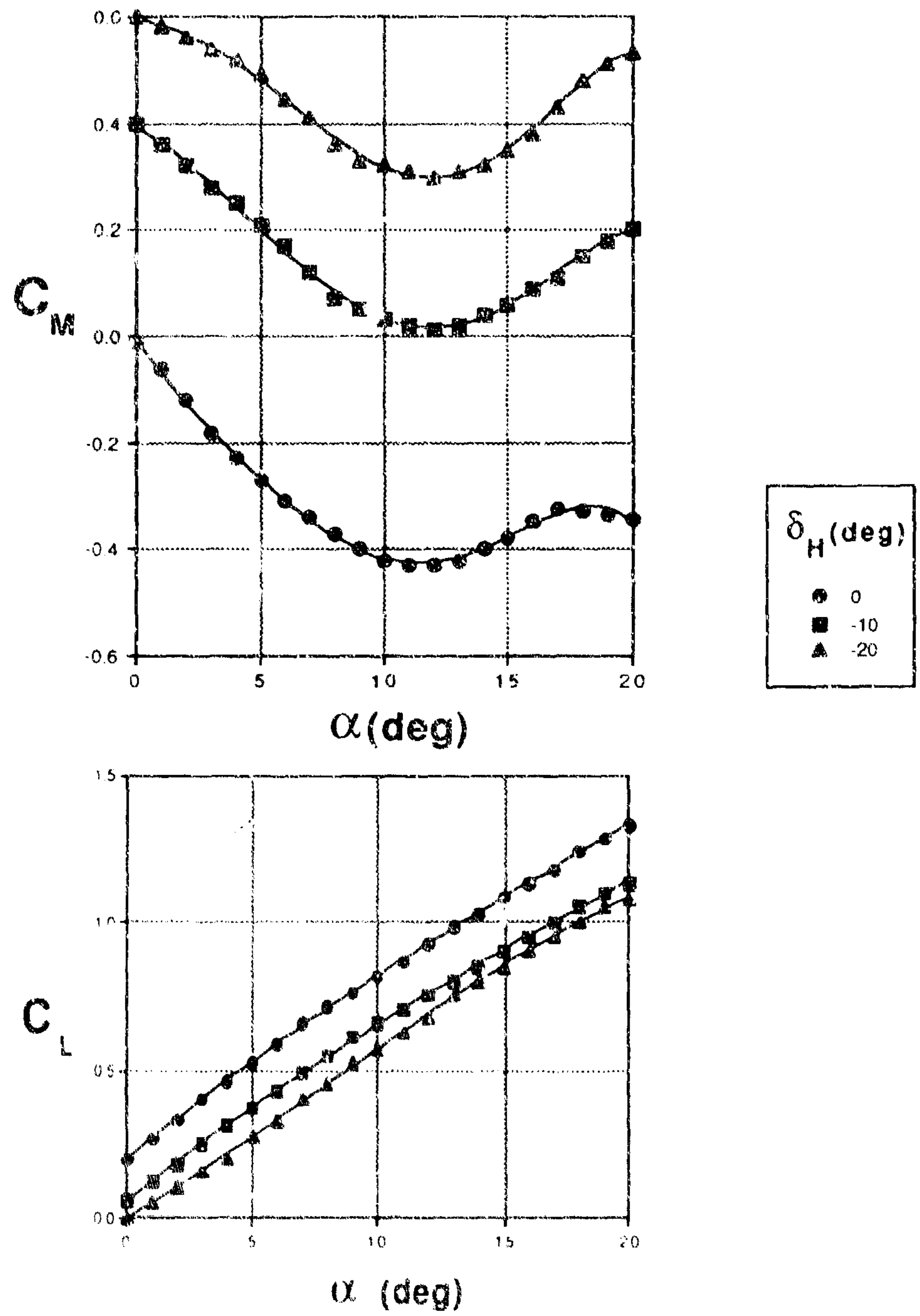


Figure 1 B-1 Lift and Moment Characteristics

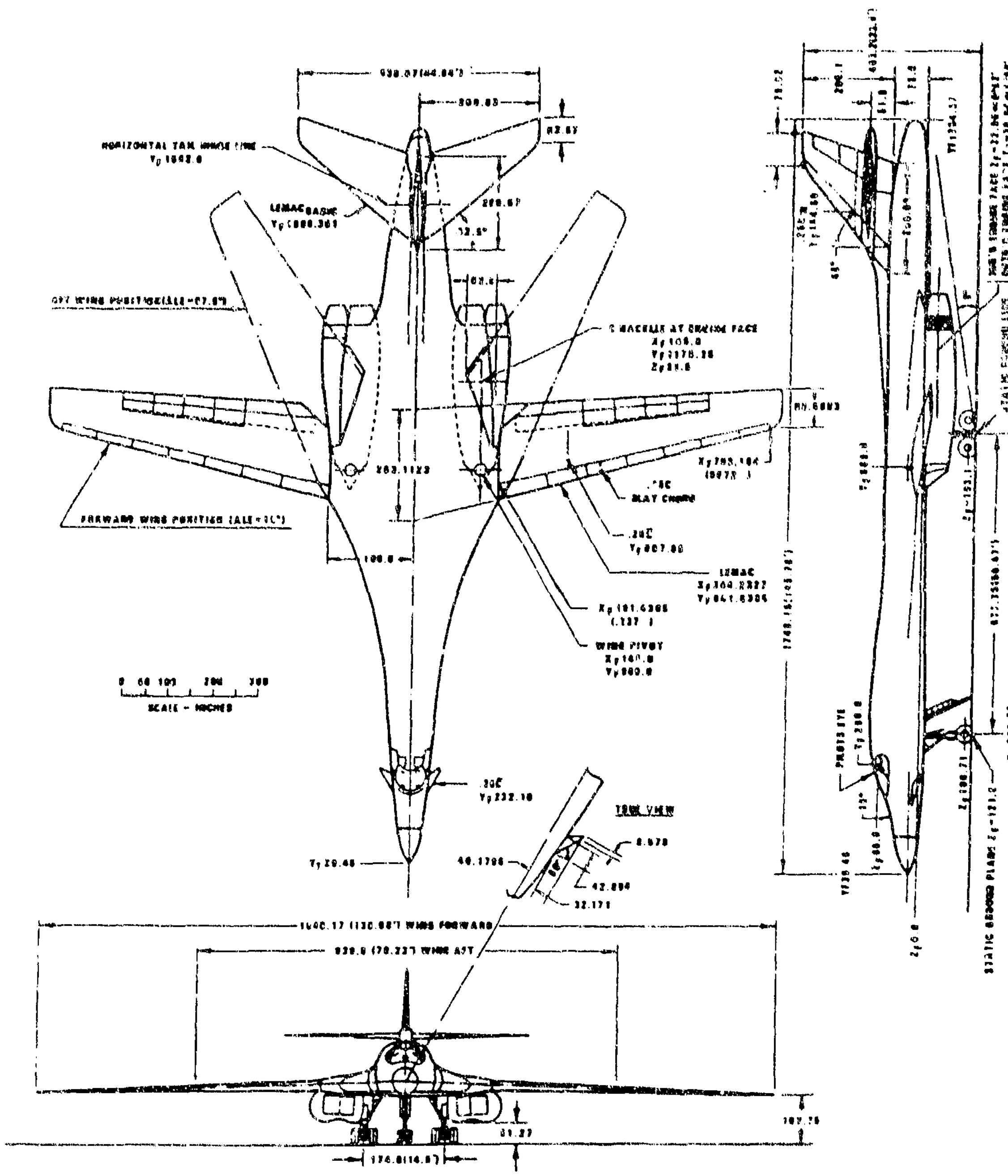


Figure 2 B-1B Three View

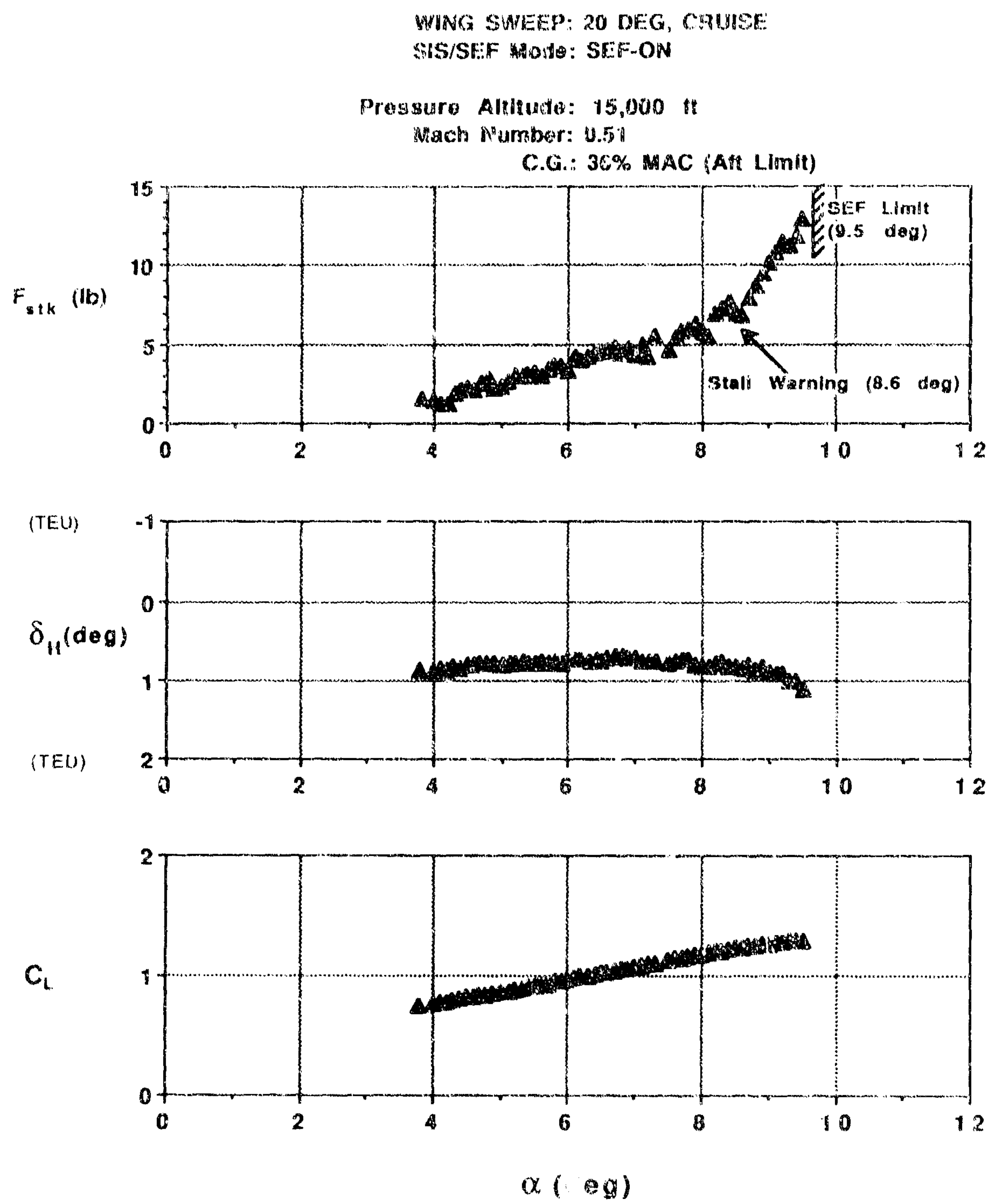


Figure 3 1-g Level Deceleration Flying Characteristic with SIS/SEF - 20 CR

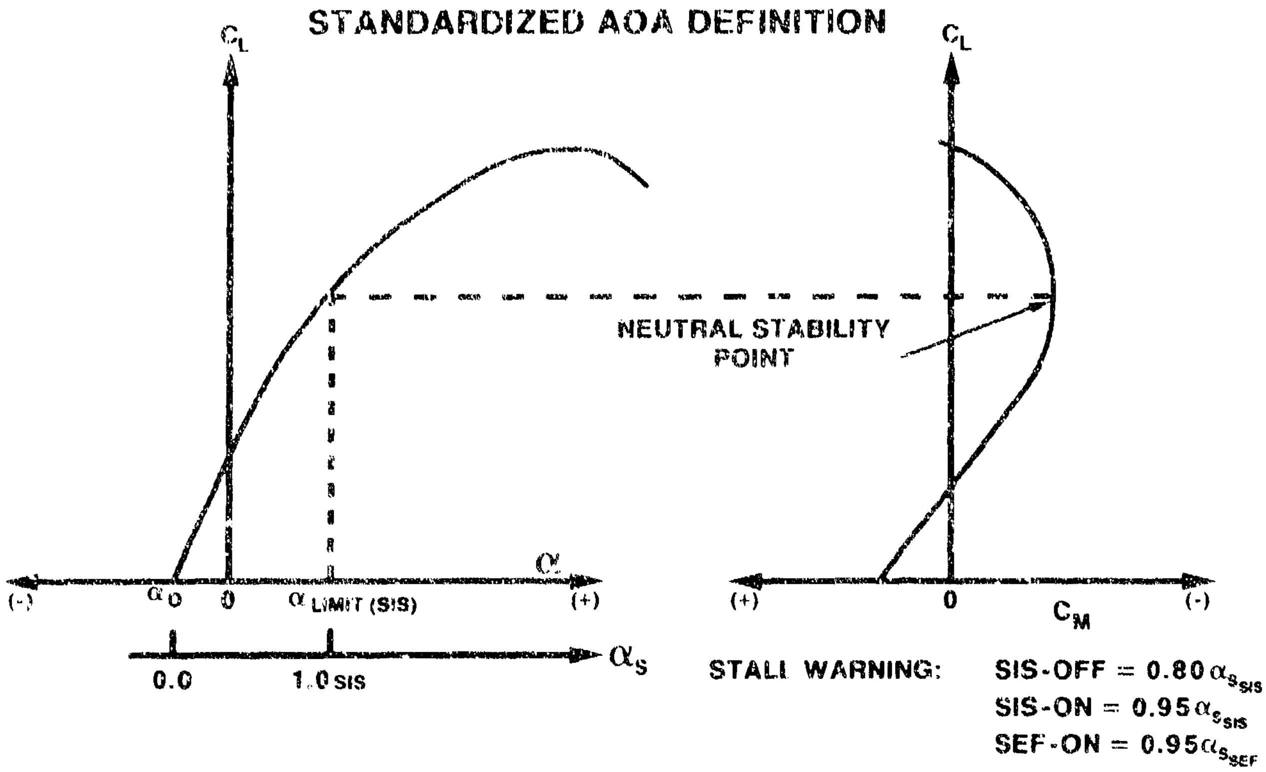
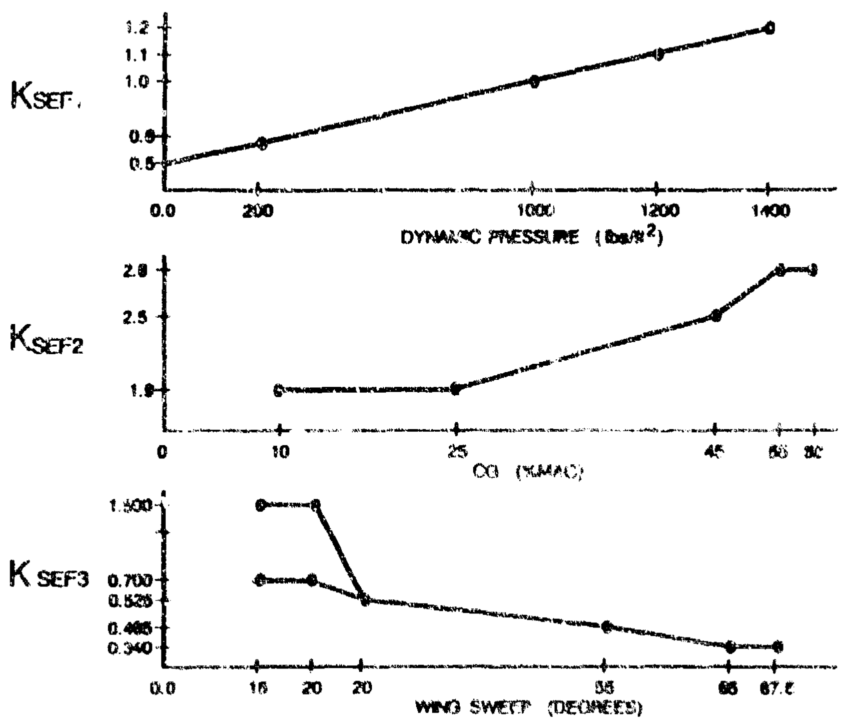


Figure 4 SIS Standardized AOA Definition



$K_{SEF} = K_{SEF1} \times K_{SEF2} \times K_{SEF3}$

Figure 5 CADC Stability Enhancement Gain

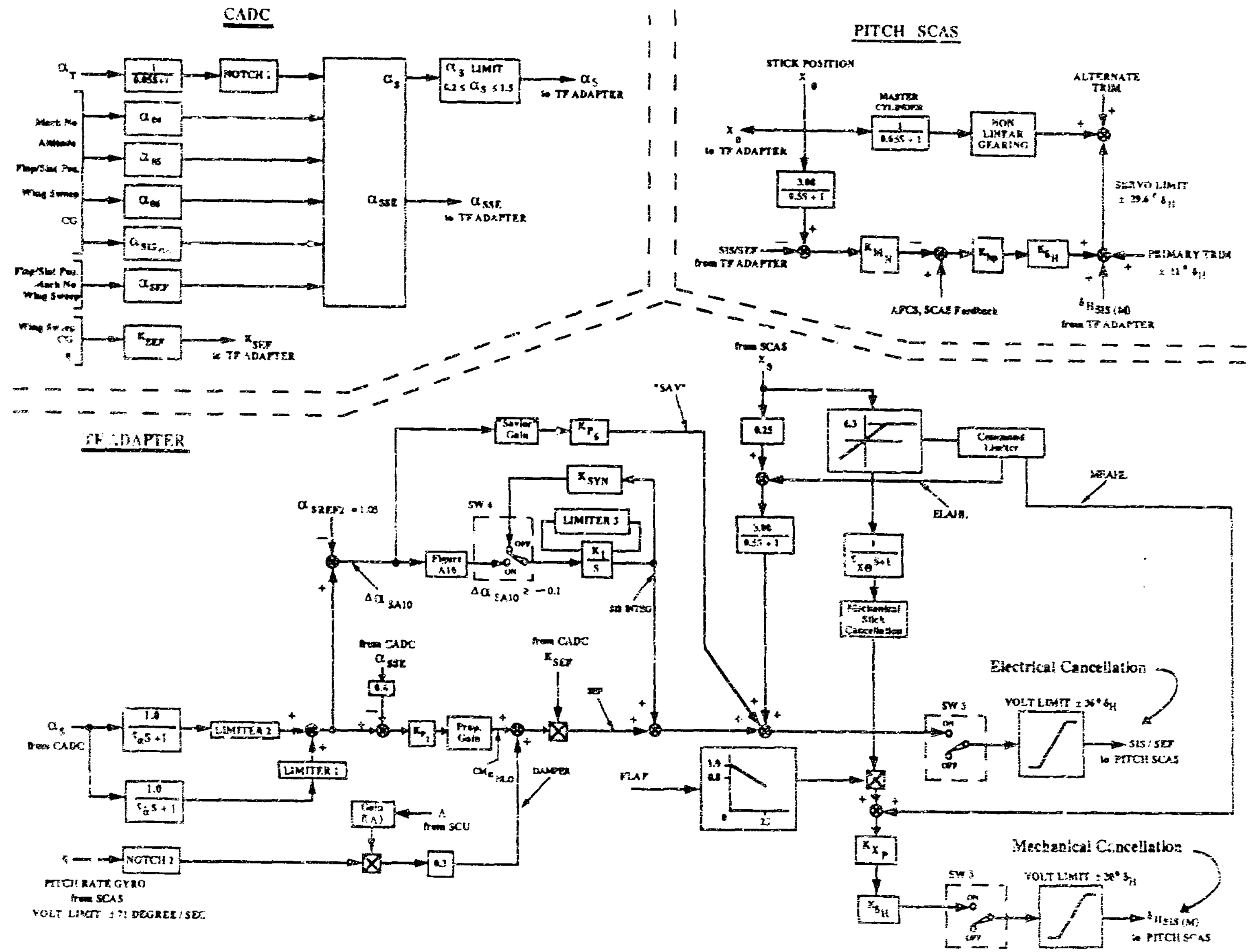


Figure 6 SIS/SEF Functional Diagram

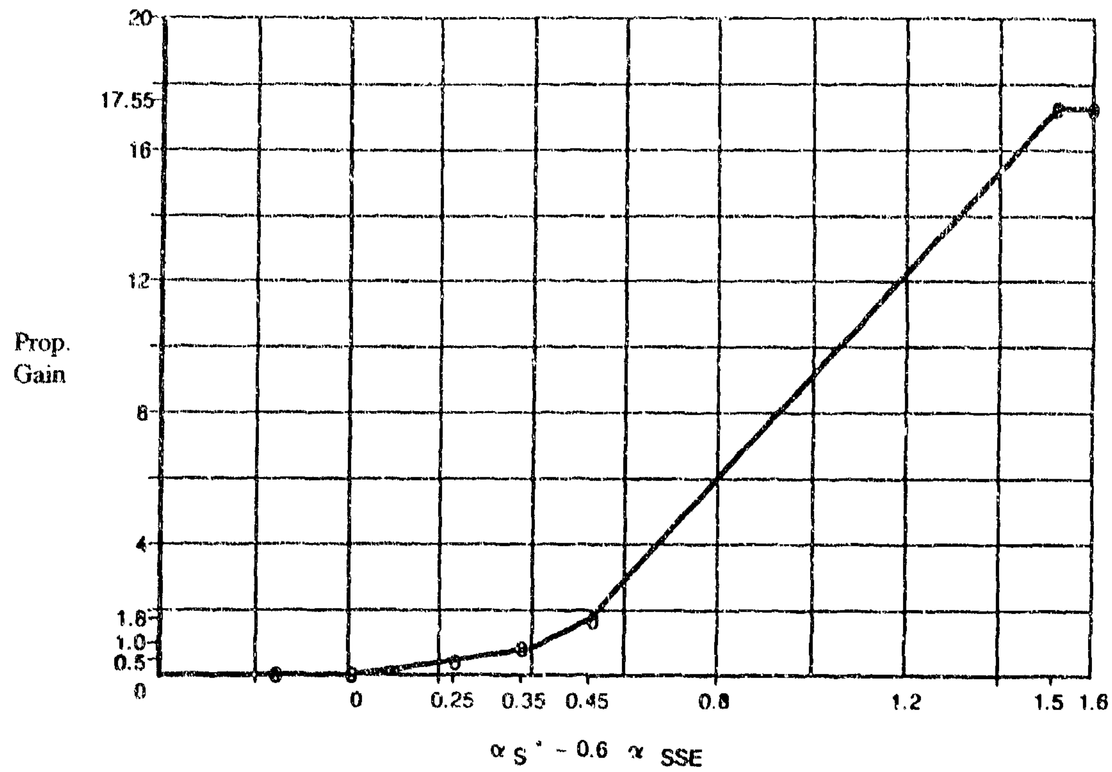


Figure 7 Stability Enhancement Nonlinear Gain

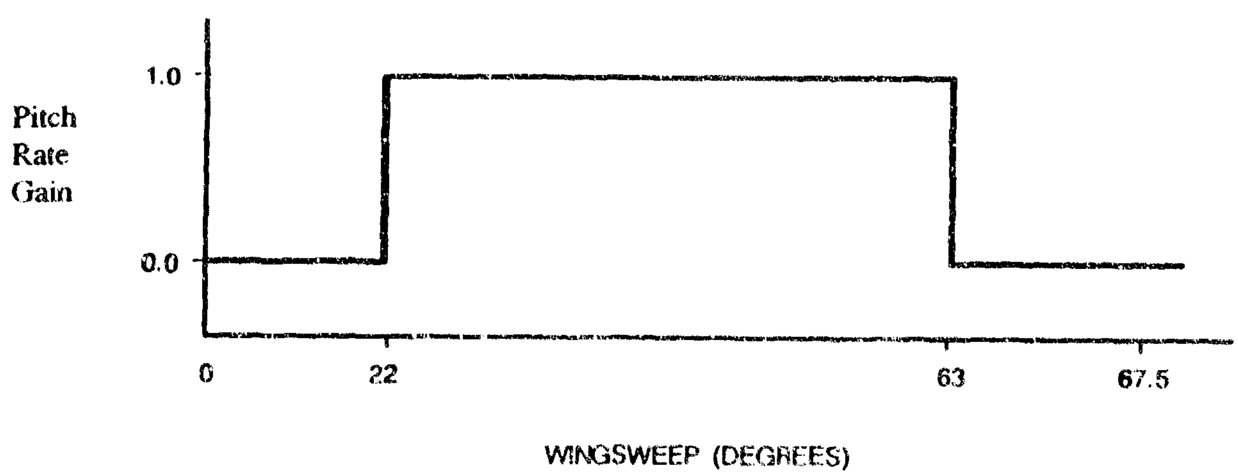


Figure 8 Pitch Rate Damping Offset

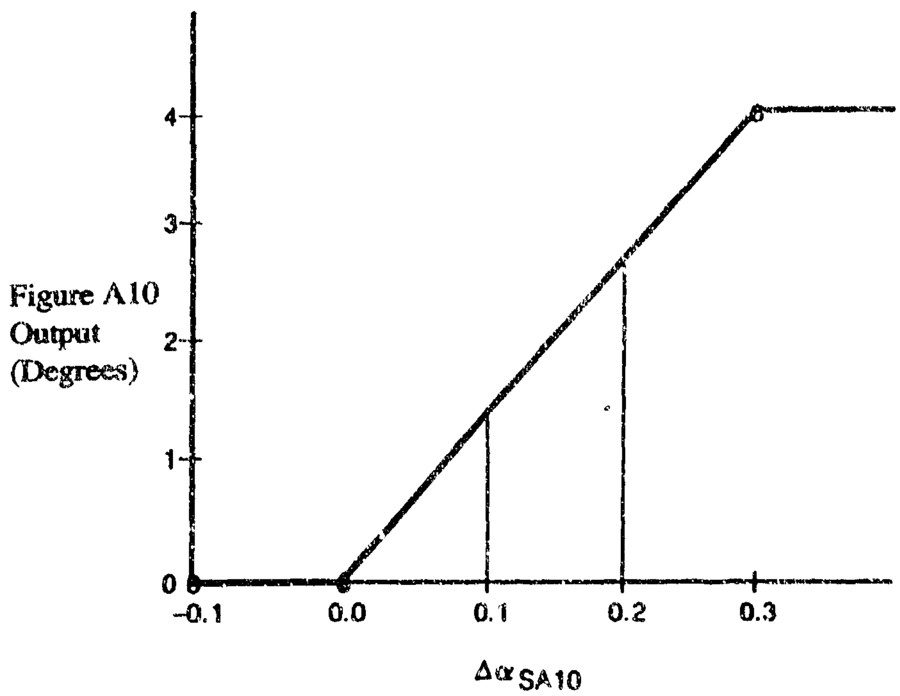


Figure 9 α_a Limit Integration Offset

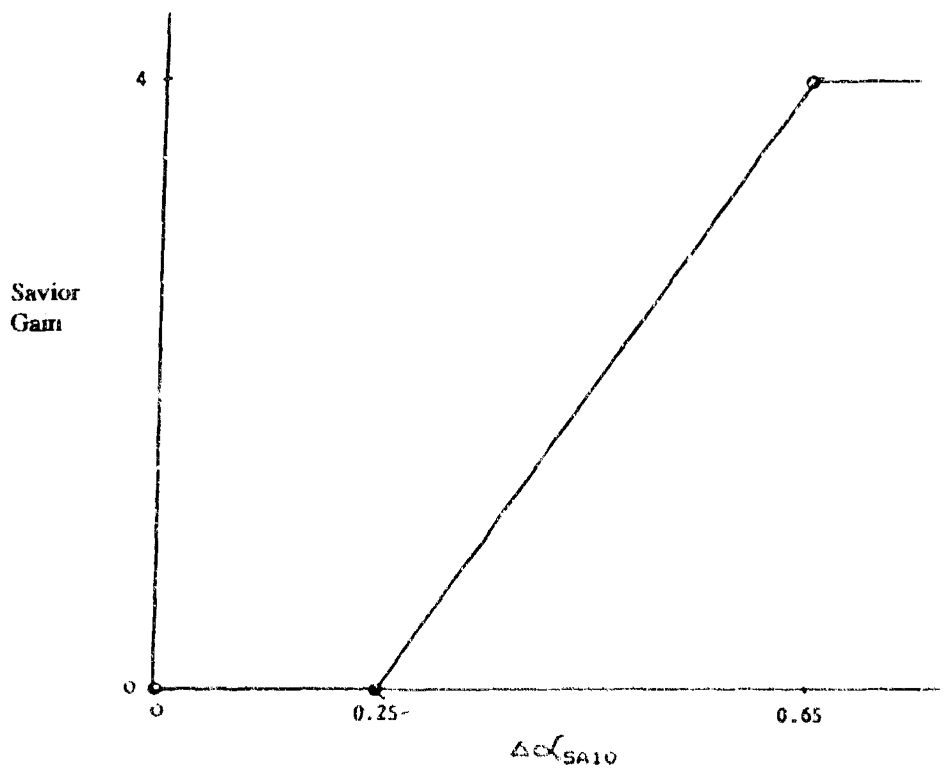


Figure 10 SIS-2/SBF Savior Path Function

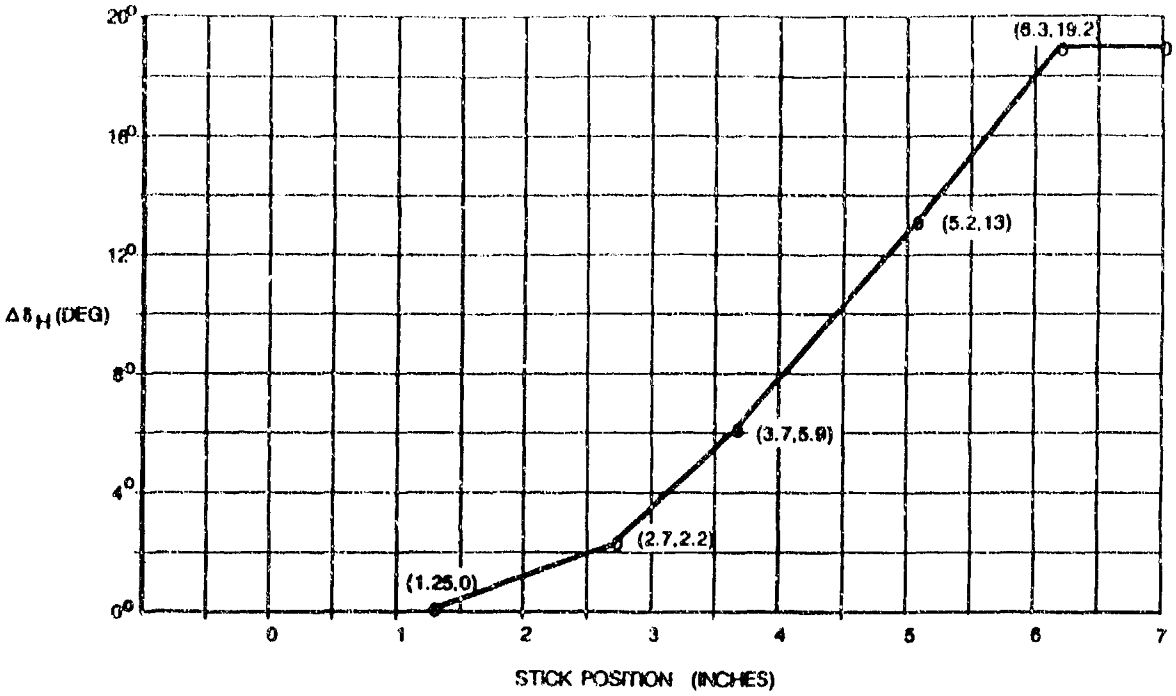


Figure 11 SIS-2/SEF Nonlinear Mechanical Cancellation

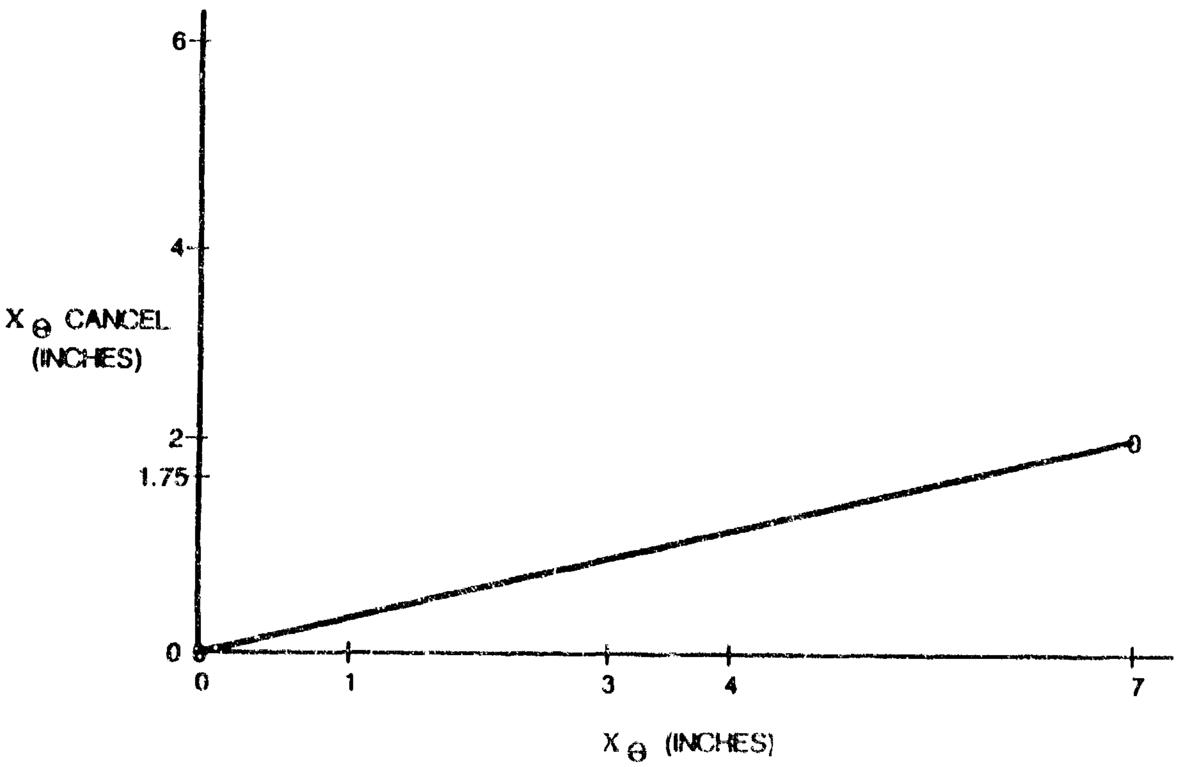


Figure 12 Electrical Cancellation

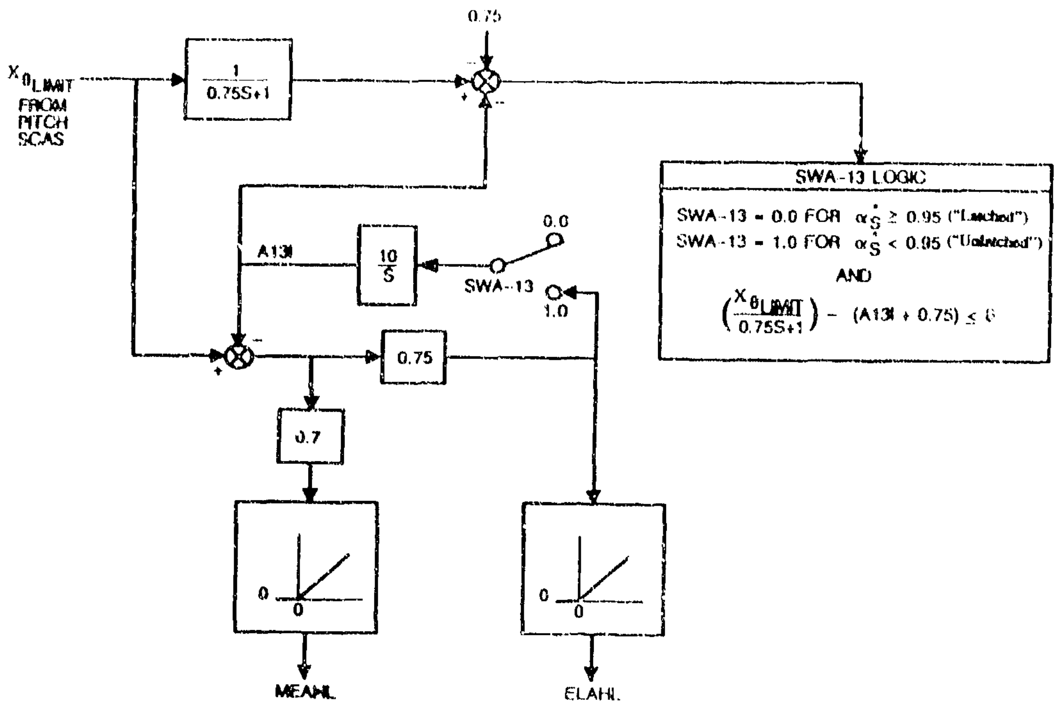


Figure 12 AOA Command Limiter

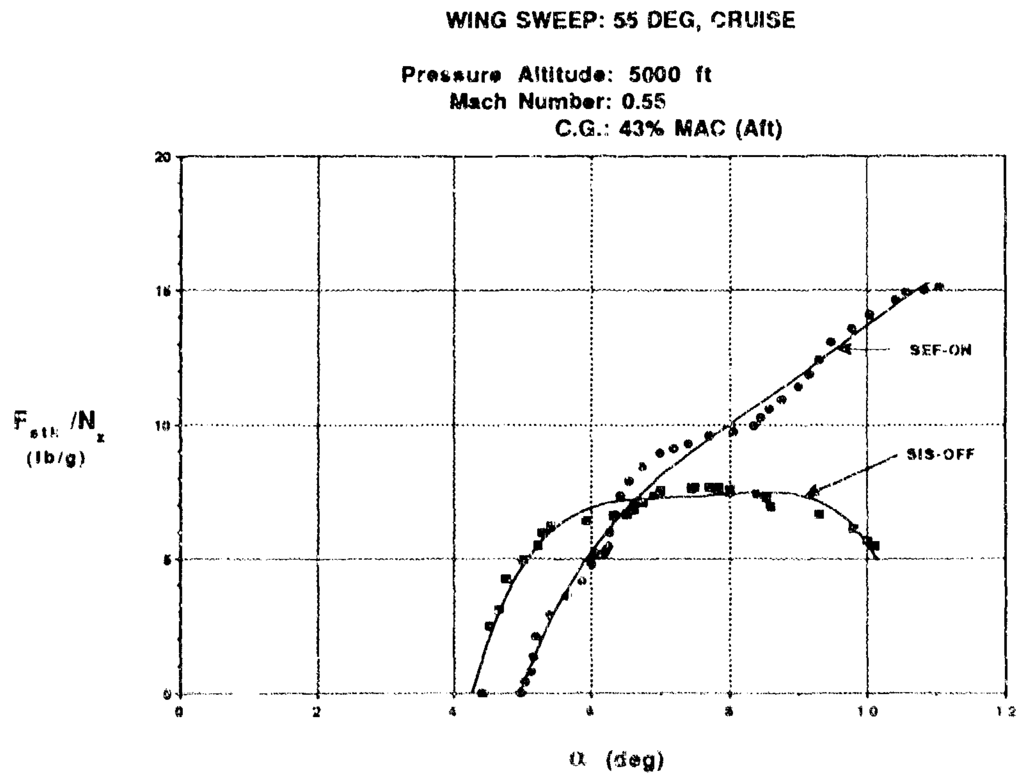
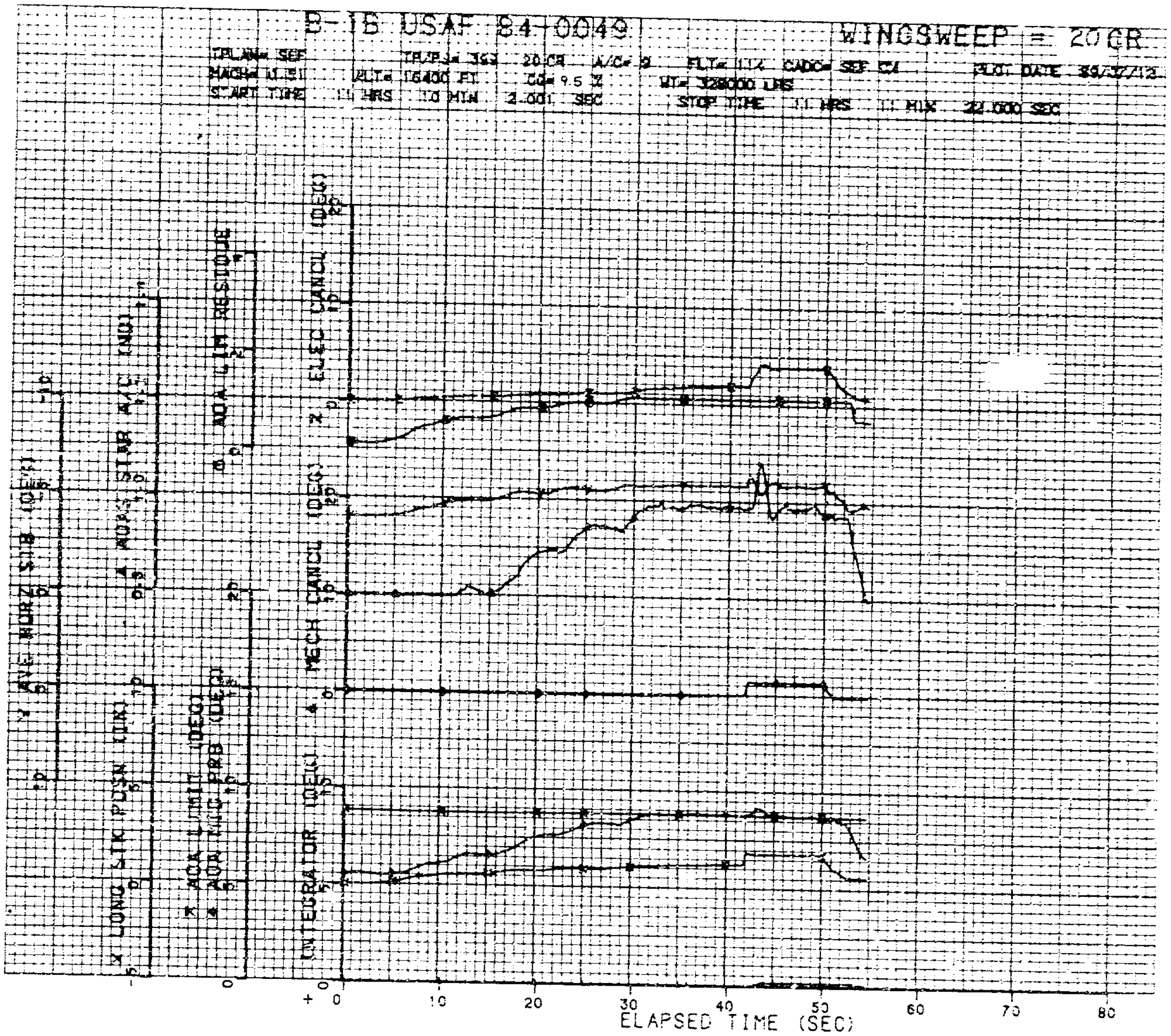
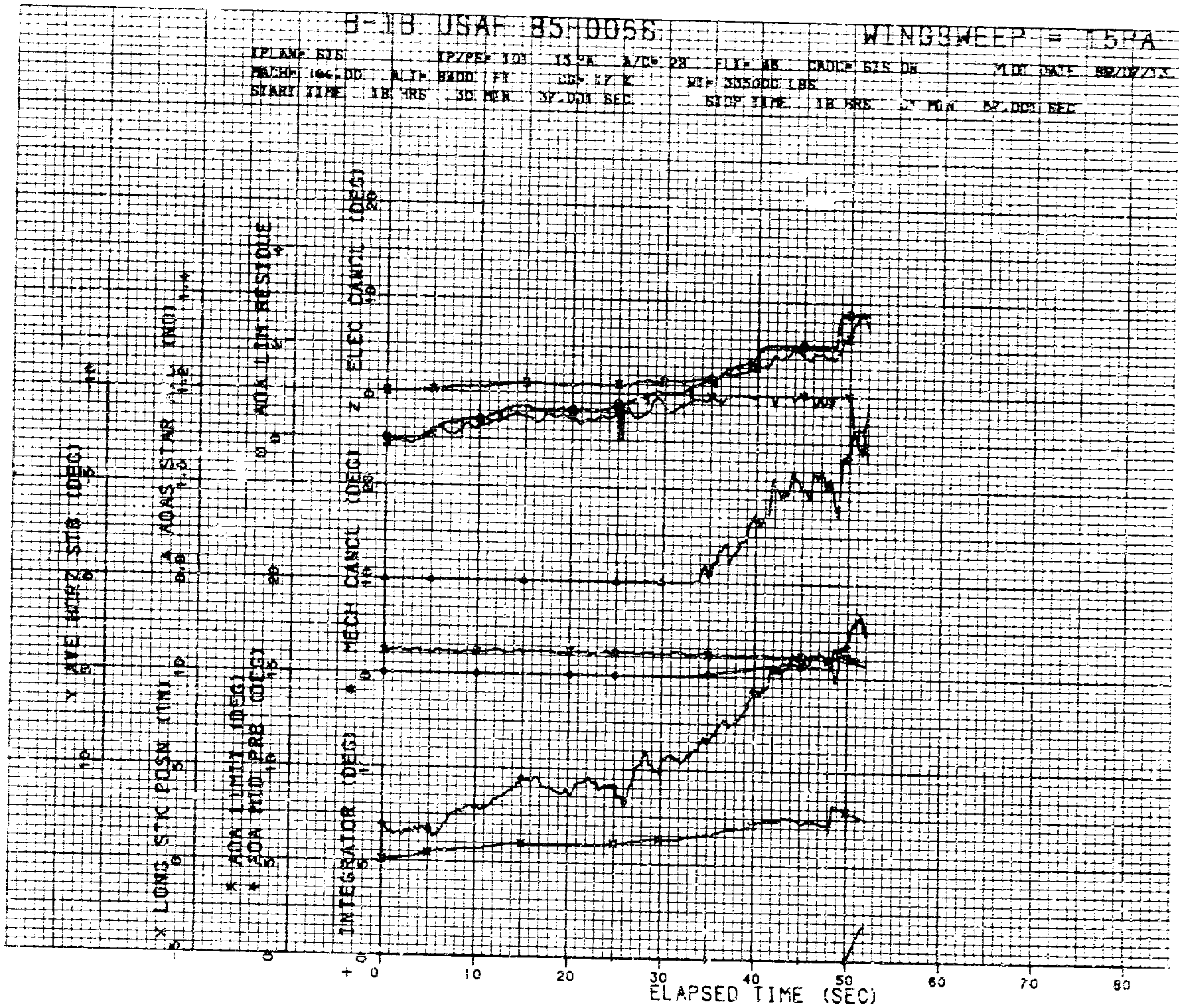


Figure 14 Windup Turn Maneuvering Performance With and Without SIS/SEF - 55CR



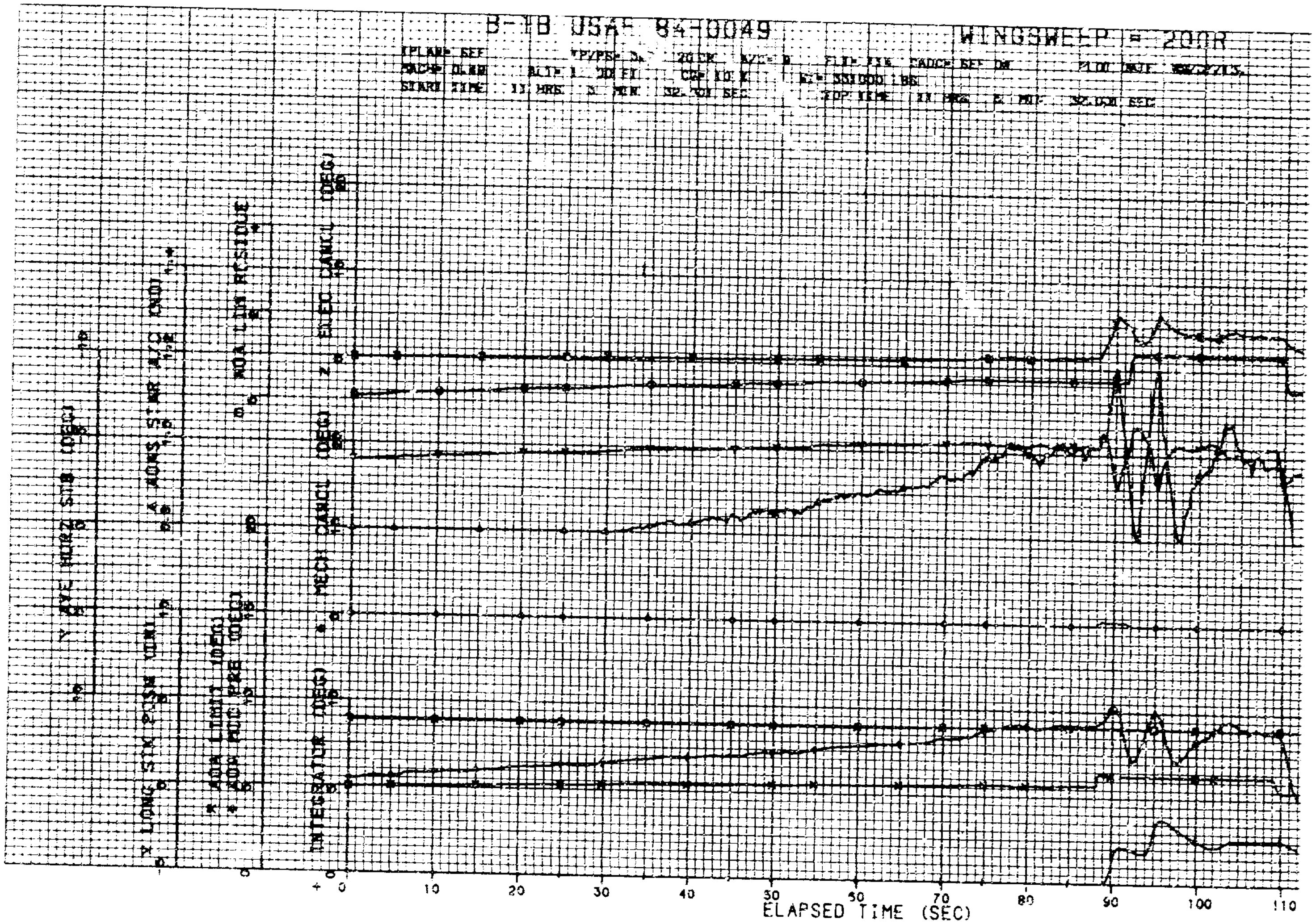
Note: Represents desired step input response.

Figure 15 Step Input - 20 CR



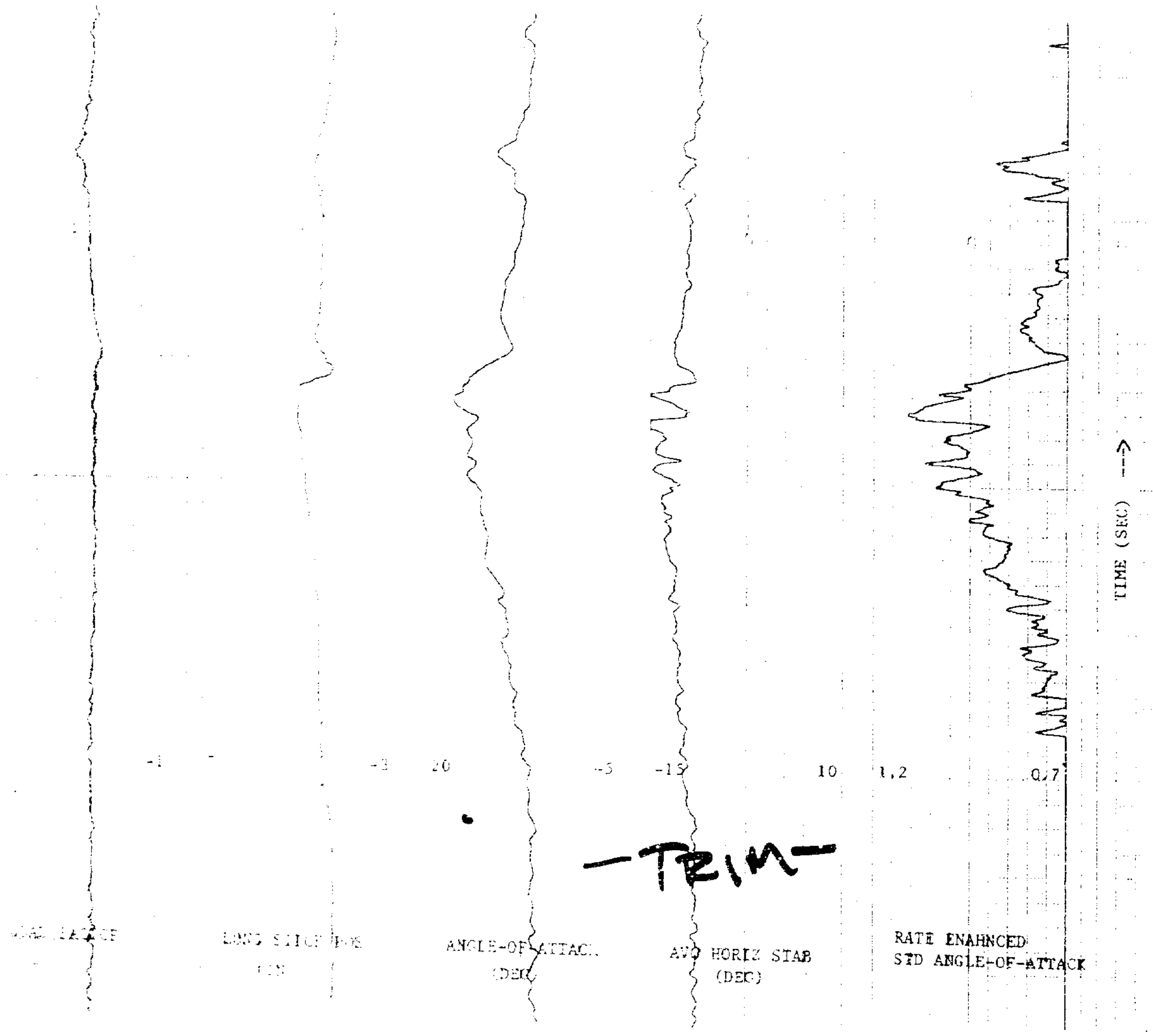
Note: Represents step input below stall warning.

Figure 16 Step Input - 15 PA



Note: Represents secondary overshoot due to unlatching and large undershoots due to high-integrator output.

Figure 17 Step Input - 20 CR



Note: Repre ents uncommanded pitchup transient due to unlatching while flying in turbulence.

Figure ... 1-g Decel - 20 PA

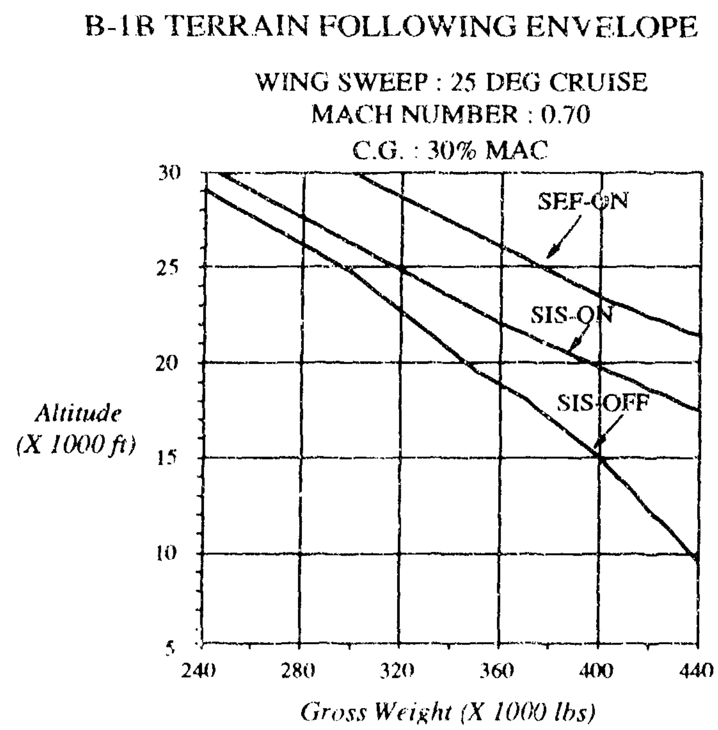


Figure 19 Aerial Refueling Benefit With and Without SIS/SEF

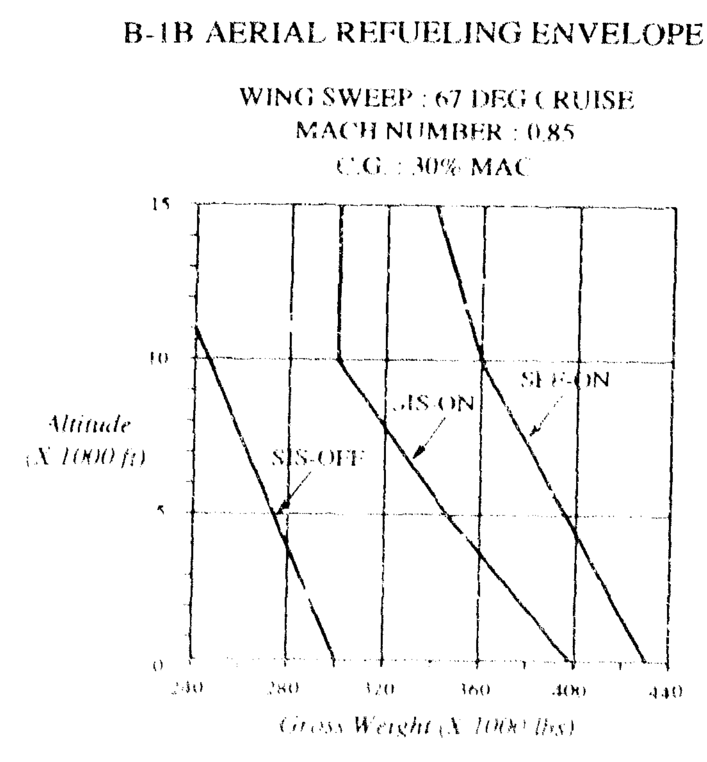


Figure 20 Terrain Following Benefit With and Without SIS/SEF

FLYING QUALITIES OF THE X-29 FORWARD SWEPT WING AIRCRAFT

by

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ABSTRACT

An overview of the X-29 Forward Swept Wing Technology Demonstrator traces its development and test path during the past 6 years. Brief descriptions of the aircraft and its flight control system provide insight for evaluating this unique vehicle. The baseline flight control system provided a starting point for safe concept evaluation and envelope expansion for the aircraft. Subsequent up-dates resulted in performance levels favorably comparable to current fighter aircraft. Efforts are described for the current expansion of the X-29's capabilities into the high angle-of-attack (AOA) regime of flight. Control law changes have permitted all axis maneuvering to 40 degrees AOA with pitch excursions to 66 degrees, thereby exploiting the full potential of the X-29 forward swept wing configuration.

INTRODUCTION

Agility, maneuverability, integration: key words used to describe the successful development of the X-29 Forward Swept Wing Technology Demonstrator.

The X-29 integrates several different technologies into one airframe as depicted in Figure 1. The aeroelastically tailored composite wing covers cause the forward swept wing to twist as it deflects, successfully delaying wing divergence. The thin supercritical airfoil, coupled with the discrete variable camber produced by the double-hinged full span flaperons, provide optimum wing performance at all flight conditions. The aircraft inherits its 35 percent static instability (time to double amplitude of about 0.15 seconds) from the close-coupled, variable incidence canard. Without it the wing-body combination is near-neutrally stable. The canard, which has an area about 20 percent of the wing area, produces lift and its downwash delays flow separation at the wing root. The three-surface pitch control--the canard, flaperon, and strake flap -- is used by the digital fly-by-wire flight control system to control an otherwise unflyable unstable vehicle. The success of the X-29 really rests with the integration of these technologies into a single synergistic configuration built for drag reduction in turning flight.

Two X-29 aircraft were designed and built. The first entered flight testing in December 1984 and concluded in December 1988, completing 242 flights and over 200 flight hours. The primary objective of Ship #1 testing was to validate, evaluate and quantify the benefits of the technologies on board, both individually and collectively.

The first 2 years of Ship #1 testing were primarily dedicated to altitude and Mach Number 1g envelope expansion. Performance testing followed the envelope expansion and was completed in December 1987. Drag reduction during maneuvering exceeded design goals by about 15 percent subsonically. Finally, Ship #1 testing provided evaluations of handling qualities, military utility and agility metrics below 20 degrees angle-of-attack.

Ship #2, which was modified to allow high AOA testing, began flying in May 1989. Its spin chute was designed to assist the pilot in regaining control in the event of a departure from controlled flight. Control surface tutorial lights mounted in the cockpit assist in this task. The flight control system software was significantly modified in order to best utilize the various surfaces in controlling the aircraft in a post-stall environment. One g envelope expansion is complete to 66 degrees AOA and 10 degrees sideslip. Accelerated entry high AOA expansion is now under way. Once the aircraft is cleared for full envelope maneuvering it will be used to demonstrate the military utility and agility of a forward swept wing vehicle operating in a high AOA regime.

AIRCRAFT AND FLIGHT CONTROL SYSTEM DESCRIPTION

Two essentially identical X-29s were designed and built by Grumman Aerospace Corporation, Bethpage, New York (Reference 1). To reduce overall program costs, the Air Force supplied several major components of the aircraft to Grumman. These included the F-5A forebody and nosegear; F-16 main gear, actuators, airframe-mounted accessory drive and emergency power unit; F-18 F404 engine; SR-71 HDP5301 flight control computers; and F-14 accelerometers and rate gyros. Use of these time-proven components also increased the reliability of the flight vehicle.

The X-29 flight control system (FCS) is a triplex digital fly-by-wire system with triplex analog back-up (as shown in Figure 2). The fail-op/fail-safe system used MIL-F-8785C and MIL-F-9490D specifications as design guides. Flying quality design goals were Level I for the primary digital mode and Level II for the analog back-up mode.

Normal aircraft operation is accomplished through the normal digital (ND) mode with its associated functional options such as automatic camber control (ACC), manual camber control (MCC), speed stability, precision approach control (PAC) and direct electrical link (DEL). ND also contains options in its gain tables for power approach (PA), up-and-away (UA), and degraded operation.

The X-29 longitudinal control law is shown in Figure 3, while the lateral/directional schematic is shown in Figure 4 (See Reference 2 for a detailed description. A more general review of the X-29 technologies and the special X-29 flight test issues are presented in Reference 3.) The normal digital mode has a pitch rate control law with gravity vector compensation, driving a discrete ACC system. This mode is gain-scheduled as a function of Mach Number and altitude and incorporates a sophisticated redundancy management system allowing fail-op/fail-safe flight. MCC is a pilot-selected, fixed flaperon sub-mode of ND used for landing. The PAC function is a pilot-selected auto throttle system. The DEL function is a ground contact control law set which is active when any landing gear weight-on-wheel relay is open. This function fades out the longitudinal forward loop integrator, allowing direct pilot control of the canards during taxi, take-off, or landing roll-out.

Gain tables for degraded operation are activated by a failure of the Attitude Heading Reference System or any 2 of the 3 angle-of-attack sensors. This function cannot be pilot-selected, nor can it be exited in flight. Degraded normal digital operation is the last option available during sensor failures prior to automatic down-moding to analog reversion.

The analog reversion (AR) mode is the back-up flight control system, designed to bring the aircraft safely back to base. The AR mode provides a highly reliable, dissimilar control mode to protect against generic digital control failures. It incorporates UA and PA functions similar to those of the ND mode. AR contains no longitudinal trim capability or pitch loop gain compensation with dynamic pressure while the aircraft is on the ground. In all other aspects, it performs like the ND control system.

The initial X-29 Ship #1 flying was limited to 0.60 Mach Number and 30,000 feet pressure altitude. As the flight envelope was expanded, the FCS evolved. Several gain and redundancy management modifications were made as a result of flight test data. The PAC and MCC modes were added to enhance the research capability of the aircraft. Addition of the Remote Augmented Vehicle (RAV) system, developed by NASA, provided the capability to pulse individual control surfaces so as to extract their effectiveness. As Ship #1 entered the military utility and agility phase of its flight test program additional changes were made. In order to enhance agility and improve handling qualities, control stick harmony was improved by reducing the longitudinal throw by about 50 percent. A further modification to the gains for both longitudinal and lateral axes was made to remove the earlier sluggishness in both pitch and roll response of the aircraft.

Late in 1987, Ship #2 was removed from storage and modifications were begun for a high AOA program. A spin chute system was added to the aircraft to assist in recovery of the aircraft from an inadvertent departure. The system was designed for pyrotechnic chute deployment and mechanical jettison. A pyrotechnic emergency jettison is also available. Cockpit instrumentation was changed to accentuate the importance of the angle-of-attack and yaw indicators by using large 6 inch meters centered on the console. Spin chute system status lights and test switches were added as well as instructional lights to assist the pilot in applying spin-recovery control inputs.

The Ship #2 flight control laws were modified to permit all-axis maneuvering to 40 degrees AOA, and pitch-only maneuvering to as high as 70 degrees AOA. Below 10 degrees, the control laws are identical to those last flown on Ship #1. Between 10 and 20 degrees, the high AOA modifications are faded in until above 20 degrees they are fully functional. The high AOA control law modifications are included in Figures 3 and 4.

The high AOA changes are fairly simple. A spin prevention logic is active above 40 degrees or below minus 25 degrees AOA with increasing yaw rate. The logic increases the authority of both the rudder peddles and lateral stick and disconnects all other lateral/directional feedbacks. Besides the spin prevention logic, an aileron-to-rudder interconnect provides for better roll coordination at high AOA. Also assisting in roll coordination is a rate-of-sideslip feedback to the rudder. Since substantial wing rock was predicted for the X-29 above 30 degrees angle-of-attack, a high gain roll rate-to-aileron feedback loop has been added to compensate for the unstable rolling moment coefficient due to roll rate.

X-29 LOW AOA TEST RESULTS

Background

The ultimate goal of the X-29 Technology Demonstrator Program was to transition new technologies to future fighter-class aircraft. The flight test program was structured to ascertain whether the suite of X-29 technologies can in fact produce performance gains through drag reduction over existing systems without sacrificing the pilot's ability to comfortably control his aircraft. The original design goal for the X-29 control law was to have this 35 percent statically unstable aircraft exhibit Level I handling qualities. For about the first 3½ years of flying (186 flights), the X-29 Ship #1 exhibited Level II handling qualities. These less-than-desirable ratings were a result of several programmatic decisions to trade design iterations and system performance for safety margin and cost/schedule savings. On early flights the control stick harmony was judged poor for a fighter aircraft, but adequate for a technology demonstrator. The flight control system gains in pitch and roll were purposely reduced to achieve added margin of safety while validating the wing structure. Again, the resulting performance was acceptable for a demonstrator but did not represent current fighter capabilities.

Flight Control System Modifications

The original control stick had a 10 inch travel in pitch and a 3.2 inch travel laterally. This unharmonious situation made lateral tracking difficult during high g maneuvers and created slow pitch response. A hardware and software change was made after flight 186 which cut the pitch throw on the stick in half while maintaining the same stick force per g. The pitch neutral point was also moved forward one inch.

Following flight 213, another flight control system change was made (Reference 4 and 5). Pitch and roll gains were increased to permit better dynamic performance from the aircraft. However, the flight g limit of 6.4 was not changed since no structural proof test has ever been conducted (thereby limiting flight to 80 percent design limit load). The results of these gain changes were a 41 per cent improvement in available maximum pitch acceleration and a 40 per cent increase in the maximum roll rate to 220 degrees per second. The results are shown in Figures 5 and 6 over the 0.4 to 0.9 Mach Number range. Note that no supersonic gain changes were made.

Flying Qualities Evaluation

The matrix of low AOA handling quality pilot evaluation tasks is depicted in Figure 7. These tasks were performed only in the normal digital, ACC flight control system mode with the up-and-away gain set. They were flown within a flight envelope of 10,000 to 25,000 (10 and 25K) feet pressure altitude and 200 to 450 Knots indicated air speed (KIAS), to a maximum 0.9 Mach Number. In all cases, the chase aircraft began either specified or random maneuvers and the X-29 pilot reacted to them.

The Cooper-Harper ratings for these tasks are presented in Figures 8a to 8d. The height of the bars indicates the range of ratings received for each task prior to flight 187. The ellipses show the average for data from flights 187 through 213, and the stars show the results for the final software flown in the military utility and agility flights after 213. The data shows an overall improvement in handling qualities from Level II to Level I.

The finger tip formation task (Figure 8a) was flown by virtually all of the pilots who flew the X-29 prior to flight 187, a total of 13. All pilots felt that the stick harmony between the longitudinal and lateral axes was poor. This resulted in an apparent sluggishness in pitch and an overcontrol tendency in the lateral direction. In general, they rated the task as medium-to-high workload. Following the control stick modification, 4 pilots repeated the task and 3 new pilots flew it for the first time. Comments referred now to good stick harmony, but perhaps too much sensitivity (gain) in roll. Pilot ratings for this task improved from Level II to Level I. The final 29 flights of Ship #1 had the increased pitch and roll response gains in the flight control system. Cooper-Harper ratings and pilot comments remained about the same. Roll response was better, although still too sensitive at elevated load factor.

The close trail formation task (Figure 8b) was performed by 11 pilots using the original control stick configuration. Again, the stick harmony was found to be a little annoying. During elevated g maneuvering, one pilot commented that he would not fly the slot position with the X-29. Several of the pilots found a small overcontrol tendency in pitch. The pilot ratings reflect borderline Level I handling qualities. The control stick modification was made and 2 pilots repeated the task. Another pilot flew it for the first time. All agreed no overcontrol tendency existed and aggressive pitch inputs could now be made. Good solid Level I ratings were given. Following the pitch/roll response flight control modification, 4 pilots reflew the task. Ratings didn't appreciably change. Several comments indicated that roll sensitivity at elevated g could be decreased.

The results for the simulated terrain following task are shown in Figure 8c. In this task the lead aircraft pilot made small (+½, -½ g) unannounced step inputs and held his input for 5-8 seconds. The job of the X-29 evaluation pilot was not to follow the lead aircraft, but to recapture the new wing reference position as quickly and accurately as possible. During the course of these evaluations, some pilots misinterpreted the original instructions and attempted to follow the lead aircraft accurately throughout the transient maneuver - an impossible task. Ten pilots participated and all had problems with the task. Many suffered small pilot-induced oscillations in pitch. "Two to 3 high frequency overshoots in pitch" was the most-used expression. It was generally a high workload task because the aircraft response lagged the longitudinal stick inputs. A solid Level II rating was assigned to this task. The same task was repeated by 3 of the pilots (and 2 new ones) following the stick modification. The overshoots still occurred but the pilots were now able to anticipate and recover more quickly. Ratings improved, but were still Level II. The biggest improvement occurred as a result of the pitch and roll gain changes. At the same time, the correct task interpretation was reinforced. Indeed, the task interpretation change was significant, for some pilots. All pilots rated the simulated terrain following task Level I with the final FCS modifications.

The final task for Ship #1 handling qualities is the air-to-air tracking task (Figure 8d). As with the finger tip formation task, all of the early pilots flew the air-to-air tracking. Three different set-ups were used: in-trail, 3g target; 90 degree heading crossing angle, 4g target; and 180 degree heading crossing angle, 4g target. Pilot ratings appeared to be independent of target set-up, although with so many variables it was somewhat difficult to interpret the results. The average scores ranked as Level II handling qualities. The lack of control stick harmony did not seem to strongly influence the pilot comments. Once the stick harmony was improved, 3 X-29 veterans and 2 new pilots flew the task. All 3 veterans found gross acquisition acceptable and fine tracking excellent. The 2 guest pilots rated the task as Level II. Following the pitch and roll response improvements to the flight control system, the same 3 veteran X-29 pilots reflew the task and found more improvement. "Good control harmony." "Nice roll response." "Pitch fine tracking was excellent." "Fine tracks as well as any aircraft I have flown!" And finally, from a guest pilot, "Fine tracks as good as our current fighters."

Summary of X-29 Low AOA Tests

The flying qualities of X-29 Ship #1 evolved during the low AOA flight tests from Level II aircraft to a solid Level I aircraft with excellent flying qualities in a variety of realistic tests. This improvement in the flying qualities was accomplished with relatively simple modifications to the FCS. The end result was an aircraft which is representative of current fighters.

X-29 HIGH AOA TEST RESULTS

Background

The X-29 configuration is novel in that it was designed from inception by Grumman to fly to high angles-of-attack. This design requirement, in concert with high levels of longitudinal instability at low AOA and subsonic speeds, defined the need for horizontal fuselage strakes at the rear of the aircraft. These strakes move the center of pressure of the aircraft behind the center of gravity at very large AOA, thereby ensuring a nose-down pitching moment to eliminate the possibility of a hung stall condition. Wind tunnel tests of the X-29 have demonstrated its ability to trim at AOA through 70 degrees. In addition, lateral control is predicted to be available to 90 degrees AOA. With this combination of low AOA instability and longitudinal and lateral control power to very high AOA, the X-29 represents a unique vehicle for investigating the application of high AOA maneuverability in future tactical aircraft.

The X-29 Ship #2 high AOA flight test program has progressed through its 5 functional test flights (June 89) and into the envelope expansion phase. To date, 50 flights have been flown with Ship #2 on the high AOA program. Once the performance envelope is fully cleared, the military utility and agility capabilities of the X-29 will be evaluated at high AOA.

Program Objectives

The objectives or goals of the high AOA flight tests are summarized in Figure 9 which also shows the X-29 stabilized at approximately 30 degrees AOA. These objectives were:

1. Define the limits of aircraft controllability. The maximum AOA at which the aircraft will respond satisfactorily to pilot inputs, both in 1g stabilized flight and during higher speed (up to 200 KIAS) maneuvering flight must be determined.
2. Maneuver in the 35-40 degrees AOA range. Can the aircraft be effectively maneuvered, particularly in the lateral-directional axes, at these angles-of-attack?
3. Pitch point up to 70 degrees AOA. Can the impressive pitch power of the X-29 be utilized to achieve these angles-of-attack without dramatic handling qualities degradations?
4. Evaluate the high angle-of-attack capability of the X-29 in the context of air-to-air maneuvers consistent with military fighter utility.
5. Finally, since this aircraft is a technology demonstrator and not a prototype for a new fighter production run, it is important to understand the flow effects within this unique configuration which produce the high AOA capabilities. In other words, it is not sufficient to just achieve good high AOA capability; it is essential that the flow mechanism which creates this capability be understood in order that the benefits can be transferred into future designs.

High AOA Test Aircraft

The high AOA test aircraft, Ship #2, is essentially the same as Ship #1 except for the following key features:

1. Modified triplex digital flight control system with special control laws for flight above 20 degrees AOA (the limit for Ship #1).
2. Modified flight test noseboom to include 3 AOA vanes to provide the necessary redundancy for the high AOA control system.
3. Spin chute for recovery from inadvertent out-of-control flight. To date, this event has been avoided by prudent test planning.

The important X-29 high AOA design features are summarized in Figure 10.

High AOA Flight Control System

Specific control law modifications for high AOA flight are given in Figures 3 and 4 while the complete details of the X-29 high AOA flight control system are summarized in Reference 2. Major features of the design are shown in Figure 11. This relatively simple design provided a rate command system in pitch with a weak AOA feedback and allowed velocity vector rolls using only lateral stick inputs. Spin prevention logic was incorporated beginning at 40 degrees AOA. Variable gain features were included in the design which allowed ±20 percent variation of selected flight control system gains in flight.

Flight Test Expectations

Extensive background studies and evaluations were made in support of the development of this unique, highly unstable aircraft. Over a period of 10 years, studies were made using a rotary balance, wind and water tunnels, special drop model tests, and free flight wind tunnels. These studies were largely done at the NASA Langley facilities and the facilities at Grumman Aircraft Corporation, who designed and built both aircraft.

From these studies, the major trends and problem areas were identified. High AOA utility of the X-29 (above 30 degrees) was expected to be influenced by, if not limited by:

1. Lack of directional control power.
2. Nose slice potential of the F-5 forebody.
3. Wing rock.
4. Engine inlets which were not specifically designed for high AOA flight.

Results from these extensive studies were also used to create the aerodynamic models required to produce the necessary ground simulations. The ground simulator was used in support of the high AOA control law development and the actual flight tests. The X-29 simulation facility was an essential part of the test program, particularly during the high AOA control law development phase. Once the tests began, the simulator remained an important element in the program despite the need for constant revision to incorporate the differences between predictions and the actual flight test measurements. The evolution of the X-29 flight test process, including the role of the ground simulator, is discussed in the Lessons Learned subsection which follows.

High AOA Flying Qualities

The X-29 performance at high AOA was excellent and generally better than expected; the flight tests were certainly not just validation of the predictions. This phase of the X-29 program was exciting, technically stimulating, and rewarding from both the engineering and test pilot viewpoint.

The results for 1g stabilized flight can best be summarized using Figure 12. The major highlights from these X-29 high AOA tests which were conducted between 40000 and 25000 (40 and 25K) feet were:

1. Good pitch control at all AOAs.
2. Controllability degraded above 43 degrees AOA, but always in a graceful manner (no departures).
3. Good lateral control up to 40 degrees AOA.
   - roll rates of approximately 35 degrees per second about the velocity vector.
   - full lateral stick rolls were well behaved and precise.
4. Wing rock didn't start until about 40 degrees AOA and then was mild (±4 degrees bank angle).
5. Directional asymmetries were evident above 43 degrees AOA: right yaw evident between 43 and 50 degrees AOA, left yaw above 50 degrees AOA.
6. Maximum stable AOA was 50 degrees.
7. Pitch excursions were performed from a stabilized 40 degrees AOA up to as high as 66 degrees AOA without departures (left yaw was not controllable, but graceful at forward c.g.'s).
8. Engine response was excellent throughout the high AOA test program.

The pitch transient from a stabilized point at 40 degrees AOA to 66 degrees AOA is shown in Figure 13. The impressive pitch power and linearity of the response are evident in the figure which represents the response at the most forward c.g. About 1/3 of the available forward stick was used to initiate recovery. It is noteworthy that the canard momentarily saturated on its 60 degree nose down stop during recovery with no ill effects. Subsequent tests to 55 degrees AOA at a more aft c.g. were not as linear and well behaved, but still there were no controllability problems.

Tests were also completed at 160 and 200 KIAS up to 35 degrees AOA which resulted in a maximum load factor of about 3.5g's. Full lateral stick 360 degree rolls were flown with good response and control. The flight control design produced near-perfect velocity vector rolls at these AOAs.

Lessons Learned

In the case of the X-29, the differences between predictions and the flight test results were gratifyingly generally in the positive direction. The aircraft flew better than expected at high AOAs. As previously noted, the major trends in the X-29 characteristics at high AOA and the potential problem areas were properly identified by the extensive background studies. The details of the X-29 characteristics were brought out during the flight tests. In some cases the differences were significant; for example, the wing rock of the real aircraft was less severe and its onset delayed compared with the predictions.

Out of this X-29 high AOA program experience, which can best be described as a discovery process, several lessons can be drawn. There are many potential reasons for the differences between the actual and predicted aircraft characteristics. The complete story will not be available until the flow effects on this unique configuration are better understood, hopefully through flow visualization and pressure studies. A partial list of the reasons for the differences would include:

1. Flow effects of the forward swept wing and canard interaction are not fully understood.
2. Important effects of the unsteady forebody vortex flow are not fully understood.
3. Rudder control power was higher at high AOA than predicted.
4. Dihedral effect was lower than predicted.
5. Lateral control power was higher than predicted.
6. Roll damping was more stable than predicted.
7. The effects of the high gain FCS are not fully understood.

In conventional flight regimes (low AOA) with a low-gain flight control system, the observed differences in control derivatives would not be significant. However, in the sensitive high AOA arena with a high-gain flight control system operating, differences between predictions and actual aircraft characteristics which are within previously accepted data tolerances can now be significant. For example, the X-29 lateral control derivative was higher than predicted by about 20 percent and the roll damping was more stable than predicted. These differences, in combination with the very high lateral control system feedback gains, produced a significant delay in wing rock onset and reduction in magnitude.

From the first tests above 20 degrees AOA which showed, at the time, apparently alarming differences from the ground simulator, the test process and role of the simulator in the test program changed. The initial concerns with these differences produced a desire to stop flying long enough to understand "the problem". Flight testing did, however, continue with a revised, more pragmatic test approach. Efforts continued in an attempt to understand the differences, which represented a difficult problem since no new prediction data were available. The new flight test approach began by admitting that the flight tests were not just validation and that the aircraft was itself one of the available research tools. Surmounting that obstacle, the next step was to proceed with a series of tests to increase AOA prudently. As part of this process, baseline observations were solicited from the pilots: does the aircraft roll in the direction commanded? by the stick? by the rudder? is the dihedral effect positive? is the aircraft directionally stable?, etc. In this way, important fundamental stability questions were directly addressed.

As the flight test results were obtained, every effort was made to update the simulator using a variety of parameter identification data. The simulator, in addition to its usefulness for test preparation, was also used to evaluate worst-case scenarios. However, the results at the next higher increment in AOA always produced new discoveries. This less than elegant flight test process was, in fact, both effective and educational and allowed a safe expansion of the high AOA envelope.

Summary of X-29 High Angle-of-Attack Tests

The X-29 high AOA research program was clearly a discovery process which included many important elements. Analysis, wind tunnels, rotary balances, and the final step, flight test, were all essential to this exciting learning process. Simply stated, the X-29 flies very well at high AOA.

Future Plans

The X-29 high AOA program is not yet complete. Future plans include:

1. Flight control system improvements. These efforts are centered on increasing the velocity vector roll rates and reducing the high frequency lateral actuator noise during higher speed, high AOA maneuvers. These high frequency inputs caused several actuator miscompares during the tests.
2. Complete the military utility and agility evaluations to understand the potential of applying the X-29's high AOA maneuvering capabilities during air-to-air engagements.
3. If possible, inclusion of flow visualization capability and pressure measurement instrumentation on the aircraft. These additions are required to understand fully the complex flow interactions on the aircraft.
4. Finally, the control of the important forebody vortices would significantly increase the X-29 capability at high AOA. Various studies are underway to determine the best method of achieving the necessary forebody vortex control.

CONCLUDING REMARKS

Flight tests of the unique X-29 design have been performed at high and low angles-of-attack using both test aircraft. The drag reduction design goals of the X-29 were demonstrated at low angles-of-attack (less than 20 degrees) using Ship #1, while the high angle-of attack capability of the forward swept wing design was explored using Ship #2. The important observations about the X-29 flying qualities in both angle-of-attack regimes are as follows:

1. The X-29 Ship #1 tests clearly demonstrated the viability of flying a highly unstable forward swept wing aircraft using a 3-surface digital flight control system.
2. Excellent flying qualities (Level 1) were achieved for operations-oriented tasks during the Ship #1 testing up to 20 degrees AOA through a series of relatively simple control system modifications.
3. Excellent high AOA capability and flying qualities were demonstrated during the recent flight tests using Ship #2.

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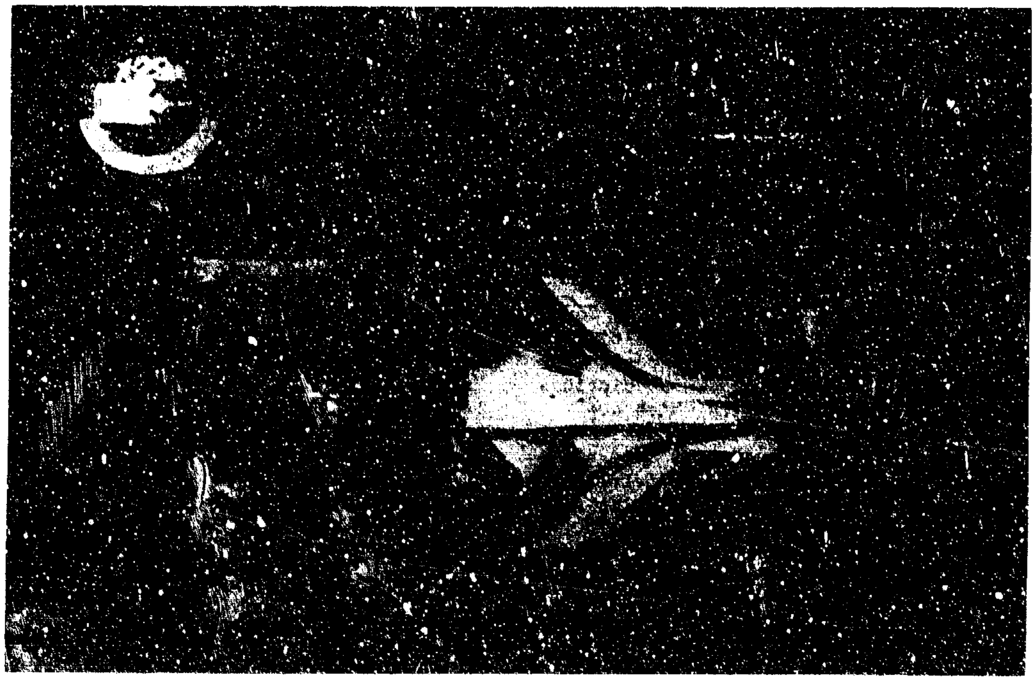


FIGURE 1. X-29 ADVANCED TECHNOLOGIES

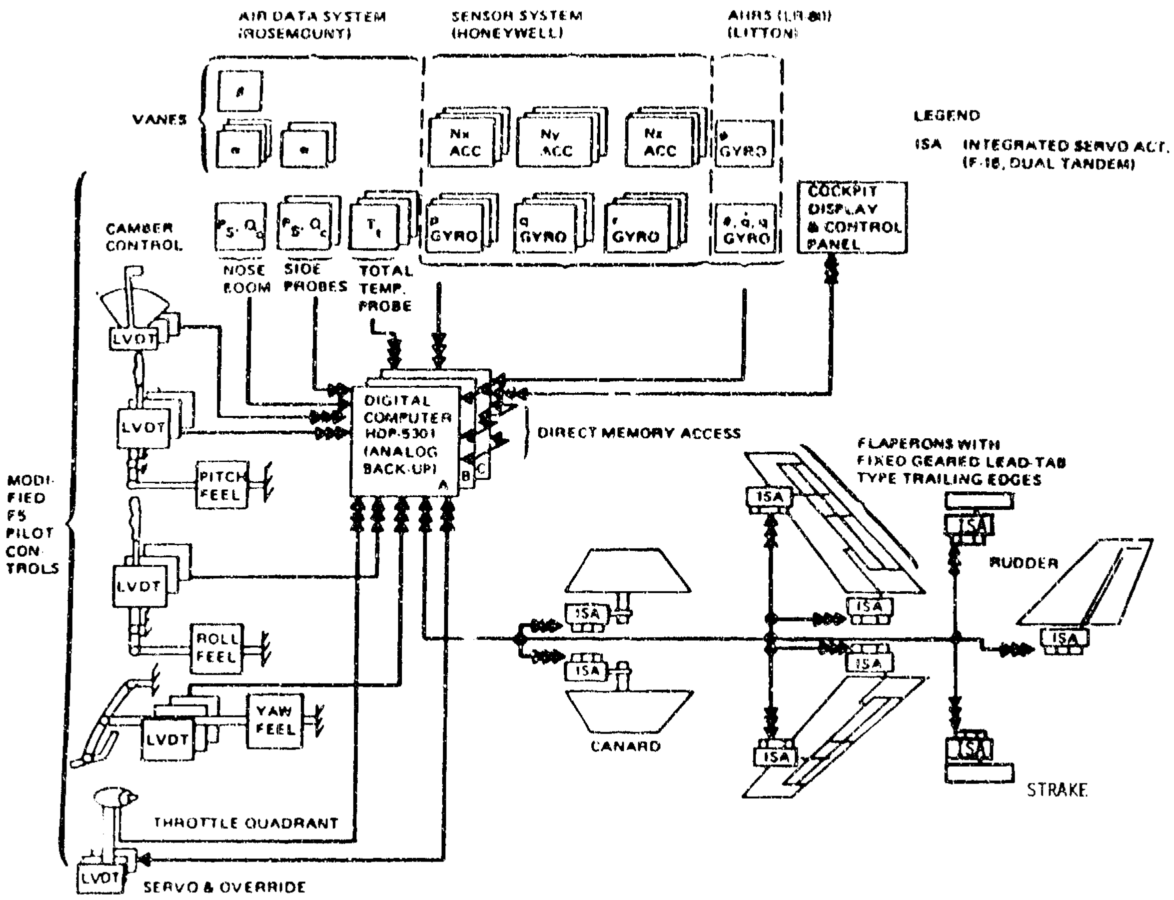


FIGURE 2. X-29 FLIGHT CONTROL SYSTEM SCHEMATIC

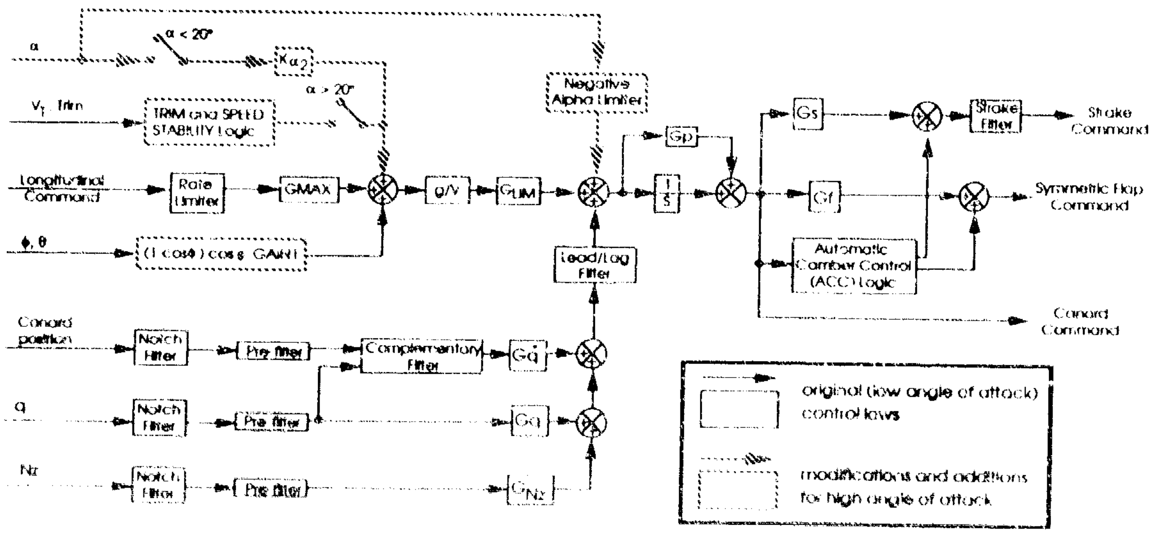


FIGURE 3. X-29 LONGITUDINAL CONTROL LAW DIAGRAM

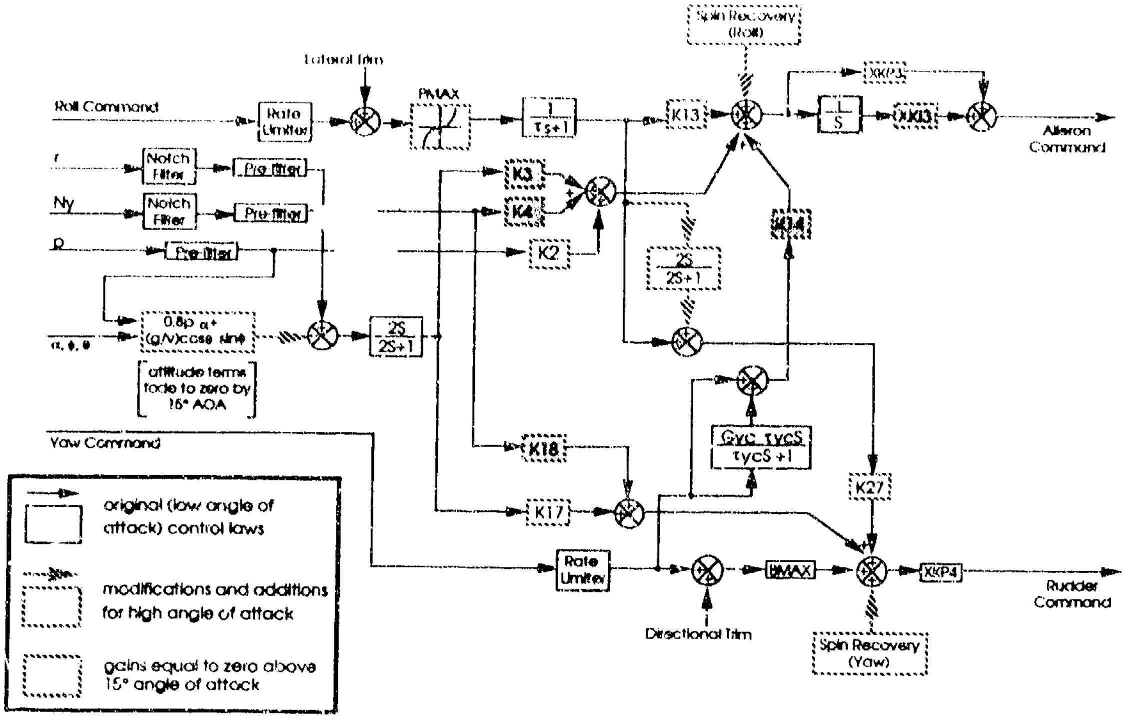


FIGURE 4. X-29 LATERAL/DIRECTIONAL CONTROL LAW DIAGRAM

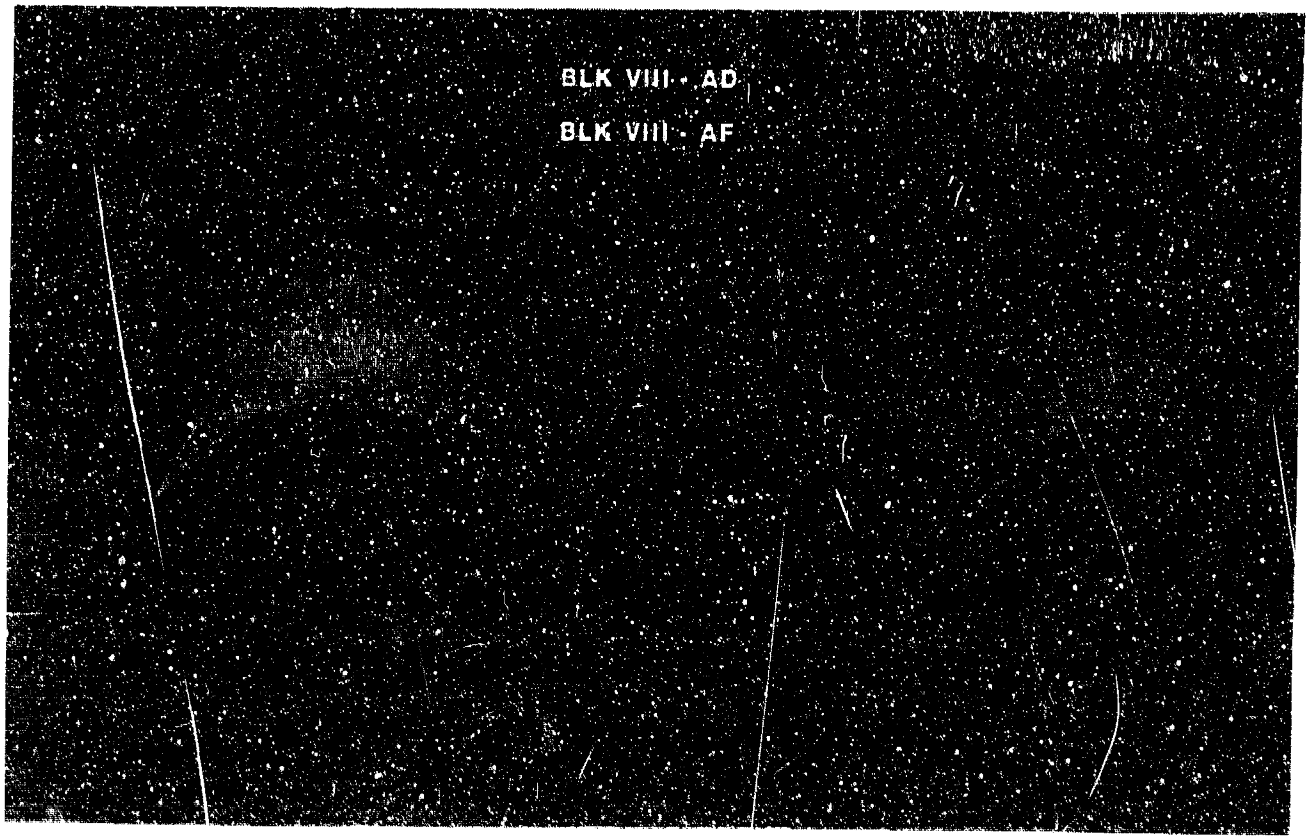


FIGURE 5. X-29 LONGITUDINAL PERFORMANCE

FIGURE 6. X-29 LATERAL PERFORMANCE

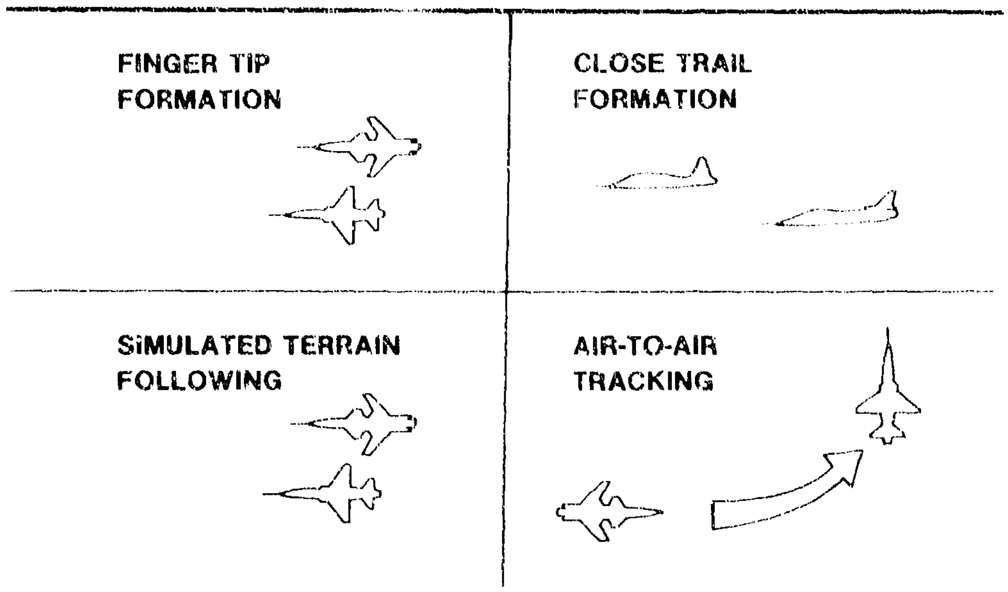


FIGURE 7. X-29-1 PILOT EVALUATION TASKS

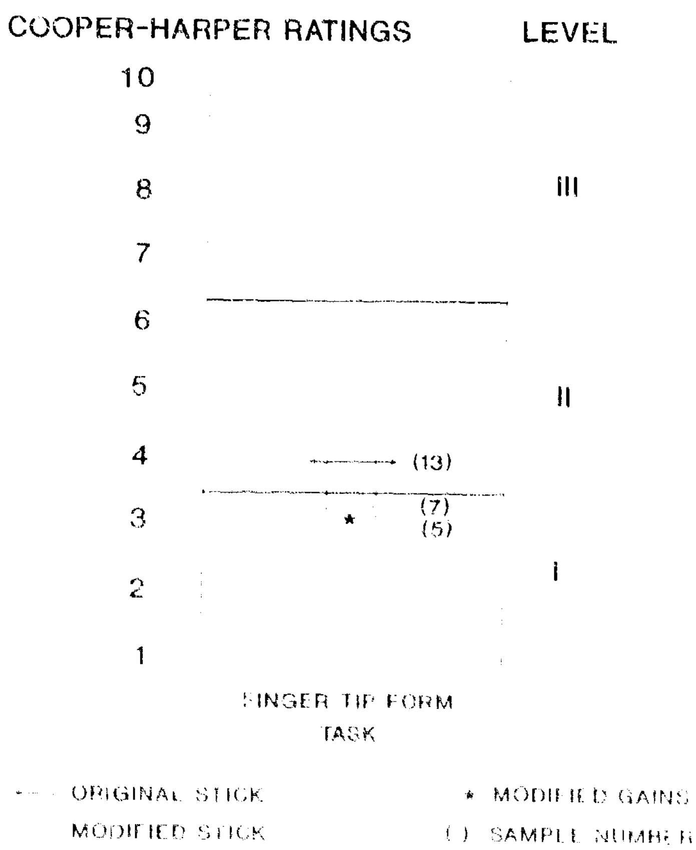
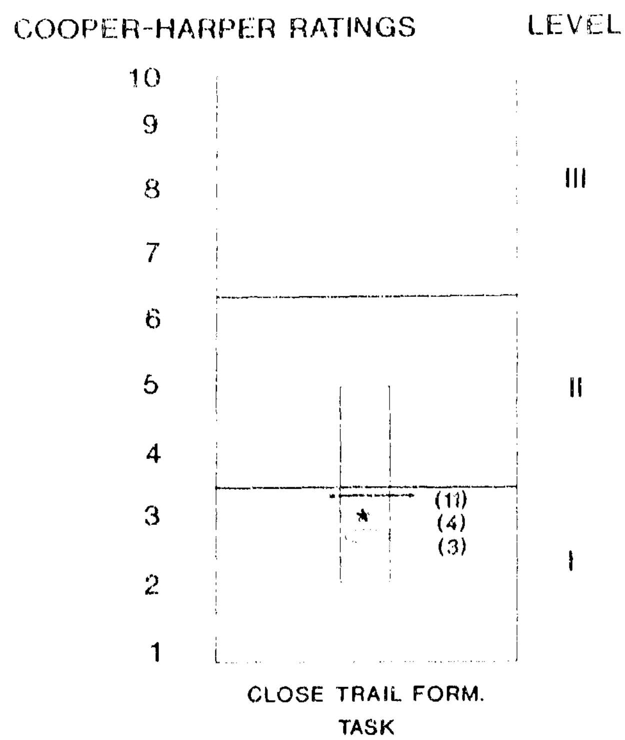
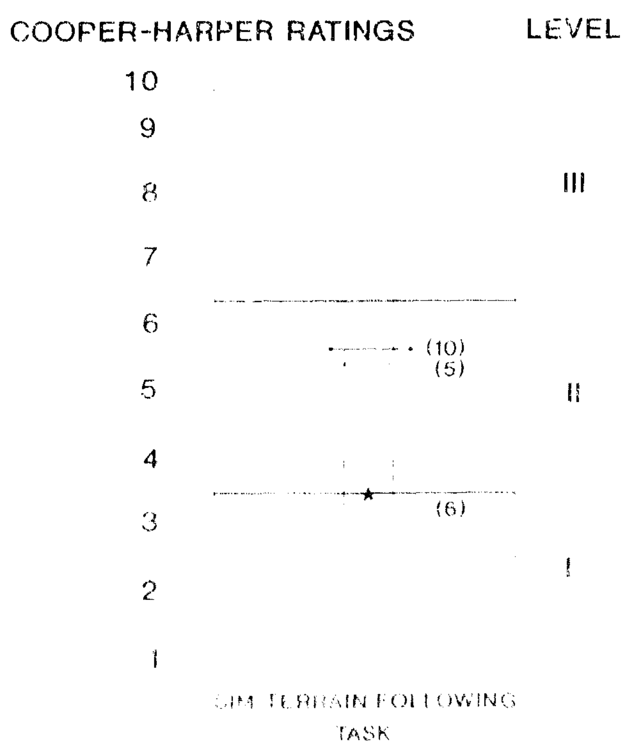


FIGURE 8a. X-29-1 HANDLING QUALITIES



ORIGINAL STICK
MODIFIED STICK
★ MODIFIED GAINS
( ) SAMPLE NUMBER

FIGURE 8b. X-29-1 HANDLING QUALITIES



ORIGINAL STICK
MODIFIED STICK
★ MODIFIED GAINS
( ) SAMPLE NUMBER

FIGURE 8c. X-29-1 HANDLING QUALITIES

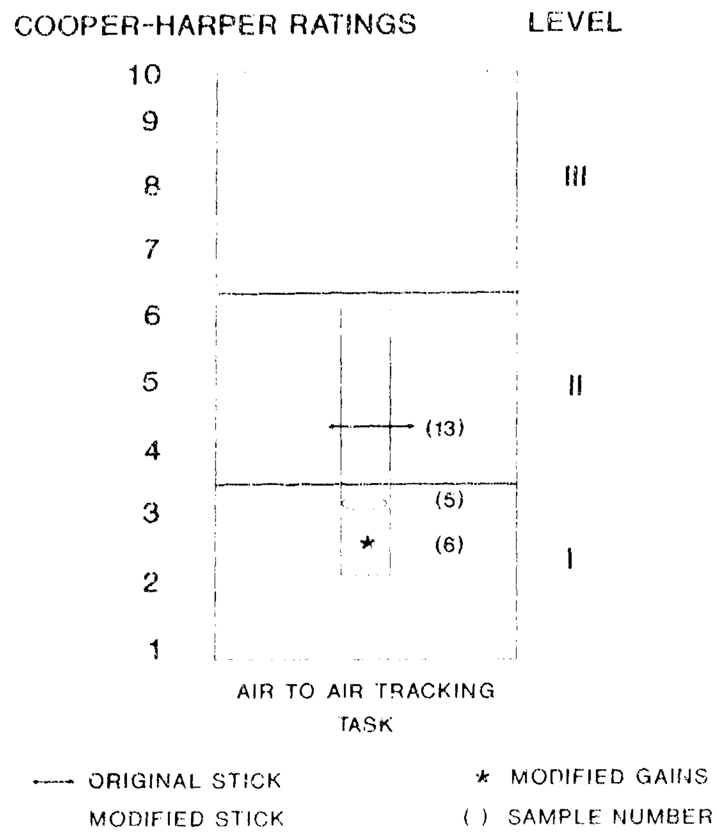


FIGURE 8d. X-29-1 HANDLING QUALITIES

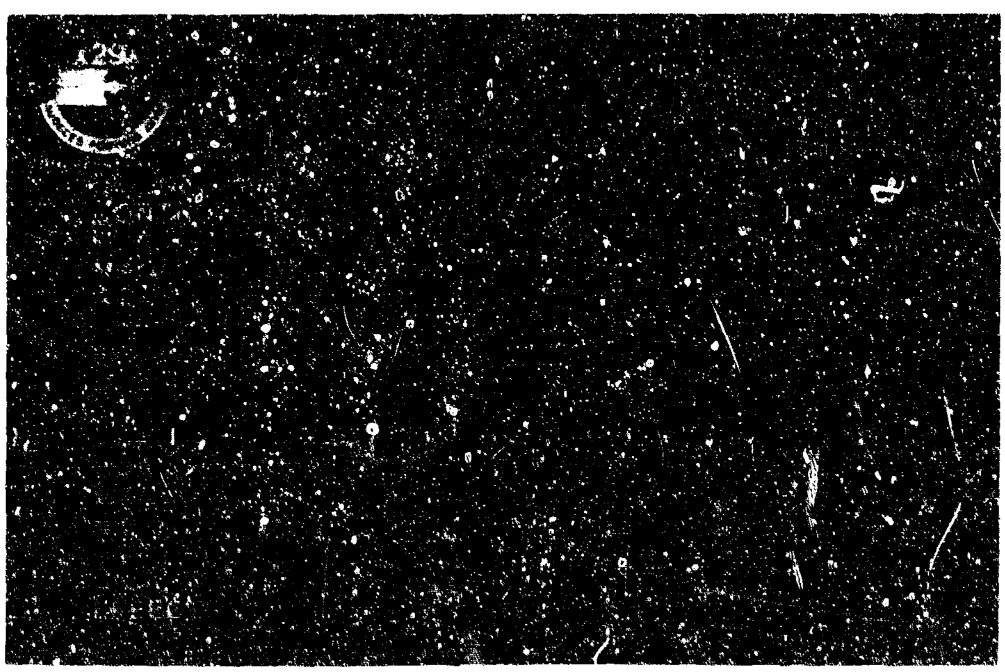


FIGURE 9. HIGH AOA PROGRAM OBJECTIVES

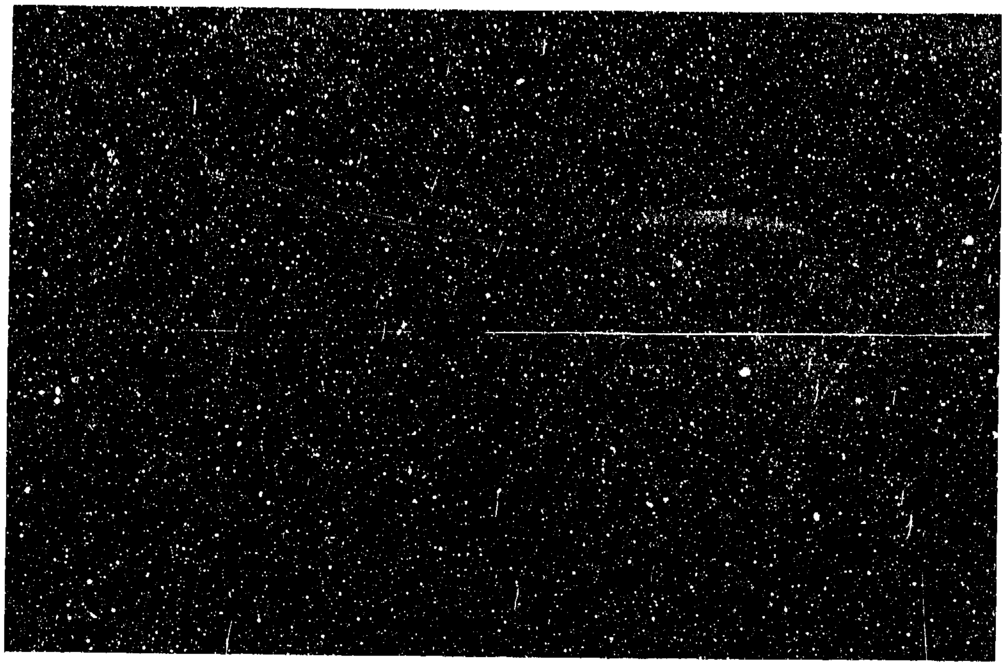


FIGURE 10. HIGH AOA DESIGN FEATURES

FIGURE 11. HIGH AOA FLIGHT CONTROL SYSTEM

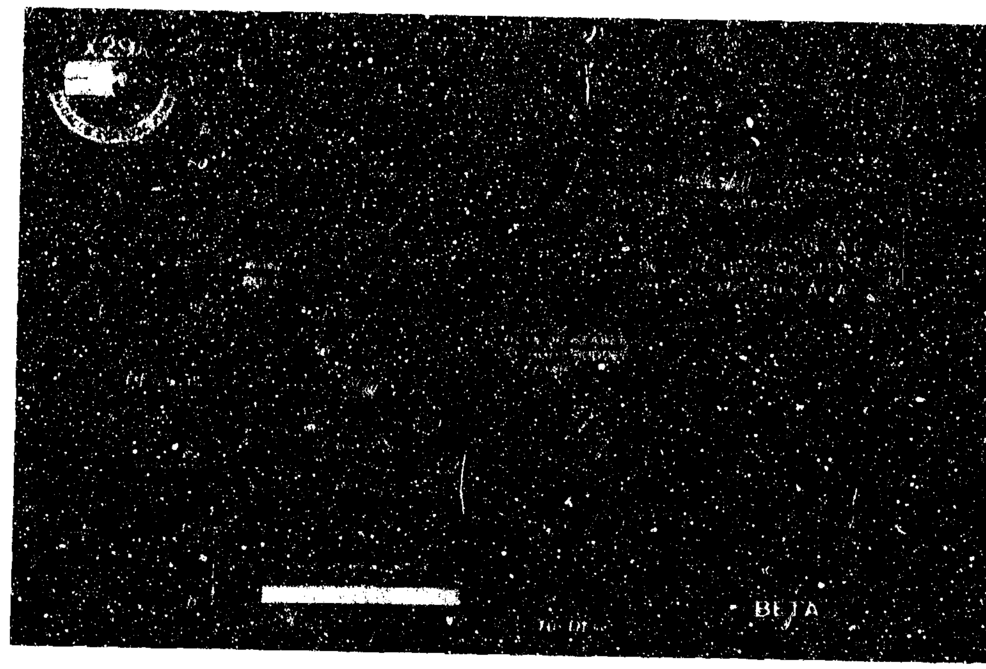


FIGURE 12. HIGH AOA FLYING QUALITIES (1G)

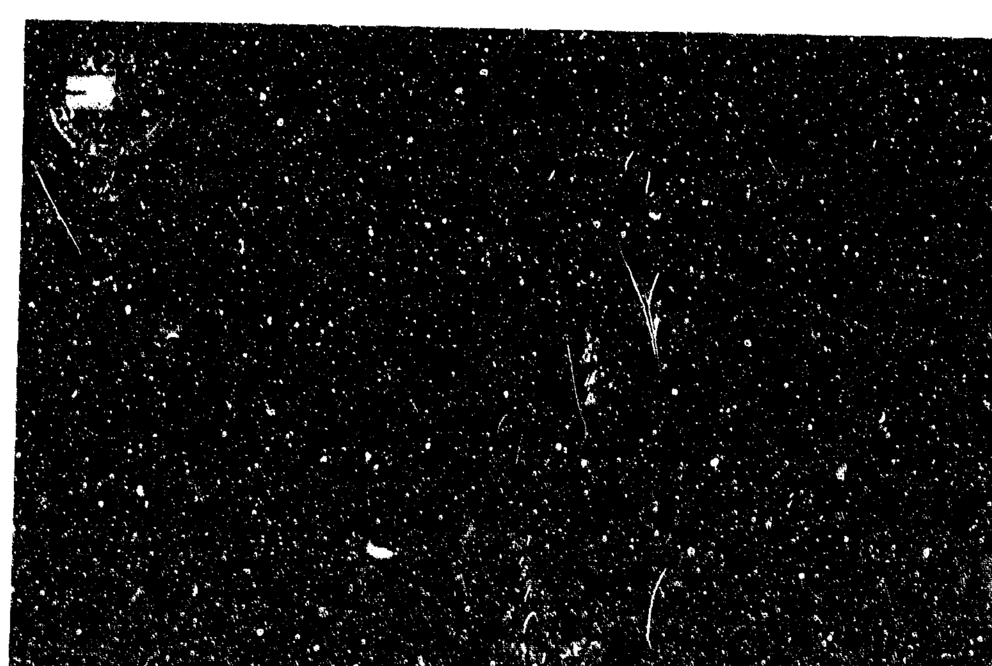


FIGURE 13. HIGH AOA PITCH TRANSIENT

Handling Qualities Evaluation for Highly Augmented Helicopters

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SUMMARY

High standards of mission performance are required for the next generation helicopters. For military and civil use, this aspect will pervade the design of helicopters and will direct the efforts to integrate control systems and cockpit technologies. The challenging question for flight control research is to specify a guideline for the development answering what is required. A facility for pilot-in-the-loop studies is needed with the capability to implement sophisticated control systems and to vary the control system parameters. This paper discusses essential aspects to evaluate in flight the performance of helicopters with high authority control systems.

Starting with describing new operational demands like high agility and precise tracking ability, the derivation of flight test tasks being well suited for the use in handling qualities investigations, is considered. In particular, the relevant DLR activities are addressed. To contribute to the establishment of a generic and credible data base for handling qualities, DLR is concentrating on the realization and utilization of the helicopter airborne simulator ATTHeS (**A**dvanced **T**echnology **T**esting **He**licopter **S**ystem). The explicit model following control system which is designed for ATTHeS is briefly presented. This paper reviews the potential of the simulation system which is illustrated by the overall system performance identified from flight test data. Due to the implemented explicit model following systems, the in-flight simulation facility is provided with the capability of a flexible and broad variation of stability and control characteristics. Finally, results of a bandwidth phase delay study are presented and the influence of coupling on handling qualities evaluation is discussed.

1. INTRODUCTION

High standards of mission performance are required for the next generation helicopters. For military and civil use this aspect will pervade the design of helicopter systems, and will direct the efforts to integrate control systems and cockpit technologies. The overall objective is to improve the potential of helicopter utilization. For example, the military operational demands include flying with high aggression close to the ground in bad weather conditions with low visibility or at night. A civil EMS (emergency medical service) mission of the future will require to land with high precision in a confined and unconstrained area in bad weather conditions, when a high probability of accidents exists, especially. The realization of all these operational demands will lead to a continuously increasing of the pilot workload. With the ultimate objective to operate helicopters under such operational conditions and with such high gain piloting tasks, there is the need to develop and to integrate technologies which reduce drastically the pilot's workload and tailor the flying characteristics to enable a satisfactory task performance with an acceptable level of pilot's workload.

The application of active control technology (ACT) inheres the potential to change necessarily the way of developing and flying a helicopter system. The implementation of high authority control system has emerged as an important tool to tailor the flying qualities of a flight vehicle to specific demands of a mission. Only the application of sophisticated digital flight control systems can yield the level of augmentation and flexibility for mission oriented flying qualities. The lessons learned from fixed wing application, especially, show that such capable high authority and high bandwidth control systems can only satisfactorily be designed when all integrated elements are considered. As the helicopter is a fundamentally different air vehicle, many of the justifications for the fixed wing application cannot be easily read across. The helicopter has its own set of problems in its design and operation for which ACT can offer attractive aspects of solution.

The challenge for flight control research is twofolded: (1) to define what is required and then (2) to design what is required. The first topic deals with the definition of guide lines for the design of high authority control systems. The control system performance evaluation is the evaluation of the integrated system and all elements, which interfere with the piloting task, will influence the pilot's evaluation of the control system acceptance. While sensors, actuators, pilot's controllers, and information system will feature in the overall system performance, it is the control law that determines primarily the flying qualities. The control law aims to assist the pilot and to achieve level 1 flying qualities throughout the operational envelope. The second topic deals with the necessary level of sophistication for the design of the control system. An individual balance has to be found between the technological possibility, the operational necessity and the originated costs for the realization.

The military handling qualities specification has been currently revised considering more advanced rotorcrafts with advanced control systems integrated [1]. Data of many studies have been included to define the updated criteria. Nevertheless, data gaps have been identified which have to be filled to verify proposed criteria or to define alternate criteria. Test programs have been conducted and will be conducted at DLR to contribute to credible handling qualities data. Two flight test vehicle serve for flying quality investigations at DLR: (1) an operational BO 105 helicopter with a standard measurement equipment and (2) the in-flight simulator ATTHeS (Advanced Technology Testing Helicopter System) based on a BO 105. *Figure 1* summarizes the areas of use for the both testbeds. The operational helicopter is mainly used to develop flight test techniques and to define the flight test tasks which represent the mission demands. The in-flight simulator ATTHeS is endowed with the capability of variable controllability and stability. ATTHeS is used as the main DLR testbed to establish data bases for definition of handling qualities requirements for advanced rotorcraft systems. In addition, the simulation system is used as a testbed for technology demonstration and control law development and evaluation.

This paper highlights the activities at DLR in the field of handling qualities investigations for future helicopter systems, with particular reference to

- the operational demands and the derivation of flight test tasks and procedures,
- the in-flight simulator ATTHeS developed for sophisticated handling qualities testing, and
- some results of DLR studies contributing to a handling qualities data base.

2. OPERATIONAL DEMANDS

The user of a helicopter seems to be not so much interested in a technical solution which can meet his operational demands. He primarily asks for the demonstration of the mission performance with an acceptable workload for the pilot. The designer of a helicopter and of the subsystems needs the substantial and complete requirements which can be used as a design guide in the phases of development. In addition, the established requirements are necessary for certifying a helicopter system. It is the task of the handling qualities engineer to transfer the operational demands in a technical terminology which can be understood by the designer and can be the basis for the communication between the customer, the certification administration, and the manufacturer.

The approach to define flying qualities criteria consists of three main steps: (1) the derivation of the flight test tasks which are representing the operational demands, (2) the establishment of a data base which can be used for the definition of the handling qualities criteria, and (3) the verification of the implemented handling quality characteristics. This paper concentrates on the first two topics. The third topic is reflecting the techniques of system identification which are covered in more detail in [2]. Nevertheless, the importance to verify the overall system characteristics shall be emphasized because the pilots are relating their evaluations to the real and not to the commanded vehicle characteristics.

A classical transport mission of a helicopter under visual meteorological conditions can be characterized in adequate acceptance with the parameters speed, altitude, and load factor. The mission can be described in a flight envelope and the requirement is to operate the helicopter safely within the envelope boundaries without reaching helicopter limitations. This classical approach to characterize a mission cannot indicate the demands of some special military and civil missions of today. Flying close to the ground and near the obstacles, tracking a target in a maneuvering flight, and landing within an EMS mission in an unconstrained and confined area in an adverse weather situation result in the need to describe the extremely high demands with some more sophisticated parameters.

In any discussion about the nature of today's and future helicopter operation the terms

- high maneuverability,
- precise tracking ability, and
- high agility

are used to illustrate the demands. Reviewing the literature, some definitions for these terms are given but an accepted standard definition cannot be found. A summary of the different measures for agility of fixed wings is given in [3]. The purpose of this paper is not to offer some new individual definitions. Referring to the topics of this paper, a clarification of these attributes shall be attempted to allow a meaningful and systematic shaping of related handling qualities requirements. These definitions are close to those given by Lappos in [4]. Also in previous papers, definitions of maneuverability and agility were discussed [5, 6]. A good description of the relations is given in [7] which points out two general properties of an aircraft which support its ability to maneuver: "One is related to the degree to which the aircraft can be maneuvered, and the other is related to the rapidity and precision with which the aircraft can be maneuvered."

Maneuverability is a function of the first characteristic. It describes the ability and limitation of the helicopter to change the flight path vector. With this understanding, maneuverability is an open loop parameter which quantifies the ability only of the helicopter. Quantities of maneuverability can be the maximum rate of climb, turn rate, and load factor.

In contrast to maneuverability, tracking preciseness is a parameter describing the performance of the closed loop pilot helicopter system. Typical design areas to influence the preciseness include the feed back control system and the display technology. An example for a maneuver including both demands maneuverability and tracking preciseness is shown in *Figure 2*. The bank to target maneuver includes the two phases: (1) the turn to target phase with a performance that is determinated by the maneuverability and (2) the target tracking phase.

Agility is the ability to change rapidly and precisely the maneuver state. A definition of agility can be put forward: Agility is proportional to the inverse of time for the transition from one maneuver to another. Agility is mainly a function of the helicopter control response and, consequently, a pilot in the loop capability. To investigate agility aspects is therefore a typical domain of handling qualities engineers. Referring to the bank to target maneuver, the time between the turn to target phase and the tracking phase describes the agility of the tested system. *Figure 3* illustrates the tracking error measured in flight. Although the testbed features good maneuverability the agility is in need of improvement. The reason for the low agility is not the on-axis response characteristics which have been evaluated as satisfactory, but the highly coupled response of the testbed which reduces the preciseness to change the maneuver state. The implementation of an adequate control system which reduces the level of coupling will facilitate the piloting task and, consequently, increase the agility.

3. FLIGHT TEST PROCEDURE ASPECTS

As previously mentioned, the approach to define handling qualities criteria includes in a first step the establishment of the test procedure which is most suited for collecting the experimental handling qualities data. The flight test task has to be defined in strong correlation with the mission or with a specific mission phase, and the piloting task has to represent the demands of the mission phase. The definition of the flying qualities levels is dependent on the various levels of piloting tasks which can be described as required levels of precision or of aggression. With this understanding of the brief framework of flight test tasks for handling qualities investigations this attribute to be representative for the mission demands must be underlined. *Figure 4* illustrates, for example, the derivation of slalom tasks from NOE (nap of the earth) mission phases. The slalom task addresses the demands on the roll axis primarily. The power spectra of the roll rate are used to correlate the demands of the operational maneuvers and the piloting task for the handling qualities investigation. *Figure 5* reviews the roll axis flight test tasks which have been used at DLR. These tasks cover the flight maneuvers of precision hover, nap of the earth, and air tracking.

In contest to the request to define representative tasks, any flight test task should fulfill the requirement to be reproducible. This means, the task should be flown by different test pilots with the same understanding of the desired task performance. A handling qualities experiment is usually conducted by systematically changing specific vehicle characteristic parameters and determining the respective pilot evaluations for those vehicle parameters. One problem, which should be eliminated or at least should be considered, is any unintentional variation in the task performance following a change in the vehicle characteristics. An approach to ensure the reproducibility is to simplify the task to a reduced number of axis. This allows the test pilot to concentrate his evaluation on the changed vehicle characteristics. An on-line quicklook to examine the achieved task performance in the test is a helpful tool for the engineer. For the slalom task a score factor was computed which indicates the averaged deviation of the flown ground track from a idealized ground track (see *Figure 6*). In addition to the examination of the achieved task performance, the score factor gives an indication of the pilot's learning curve. If parameters of the vehicle response characteristics are changed, the pilot needs some time to become familiarized with the new vehicle configuration and to adapt his control strategy to the configuration and the task. The pilot ratings and comments should be related to the test runs when the pilot has achieved a well adapted control strategy. This avoids a possible misinterpretation of the pilot evaluations in correlation to the implemented vehicle characteristics.

A third aspect which has to be concerned is primarily related to the investigation of level 2 or 3 handling qualities. To conduct handling qualities tests in a realistic operational environment with a vehicle having a reduced level of handling qualities incorporates the aspect of safety. The objectives of the definition of handling qualities levels have to include the examination of degraded handling qualities. To avoid any risk, these test can be performed in a ground simulator, but any ground simulation experiment should be verified in flight. In order to fulfill these needs, in-flight simulation is the ultimate assessment technique providing high realism, flexibility, and credibility. The utilization of an in-flight simulator ensures flight safety also when simulating a helicopter with degraded handling qualities.

The discussed attributes can be summarized as the R^3 requirements on a test procedure for handling qualities investigations:

- Definition of piloting tasks which are *representative* for the corresponding operational mission phases.
- Definition of piloting tasks which are *reproducible* in tests related to the incorporated pilots and the varied system configurations.
- Application of a *low risk* test procedure concerning the flight safety.

4. IN-FLIGHT SIMULATOR ATTHeS

The DLR has developed the helicopter in-flight simulator ATTHeS(Advanced Technology Testing Helicopter System). The ATTHeS simulator is based on the helicopter BO 105-S3 (*Figure 7*) which provides a nonredundant fly-by-wire for the main rotor and fly-by-light control system for the tail rotor. The basic research helicopter corresponds in all essential components to the serial BO 105 helicopter. Only the control system for the evaluation pilot has been modified. The modified system requires a two man crew (evaluation and safety pilot) when the system is flown from the evaluation pilot in the simulation mode. The safety pilot is provided with the standard mechanical link to the rotor controls. The evaluation pilot's control are electrical/optical linked to the helicopter control actuators. A simulation computer is integrated in the evaluation pilot's control link which offers to implement high authority control systems. The helicopter can be flown in the fly-by-wire disengaged basic helicopter mode, where the safety pilot has exclusively the control, and in the fly-by-wire simulator mode, where the evaluation pilot has full authority to control the testbed. The fly-by-wire mode can be switched off by both pilots and can be disengaged by the safety pilot by overriding the actuator inputs with specified control forces. In addition, an automatic disengagement system is installed using defined limitations of the hub and lag-bending moments. When the testbed is flown from the evaluation pilot s seat, the safety pilot has to monitor the piloting task with his hands on the controllers. This assignment of the safety pilot is significantly facilitated by a mechanical feedback of the actuator inputs to the safety pilot's controllers. A schematic diagram of the control system is shown in *Figure 8*.

Up to now the testbed was used at the Institute for Flight Mechanics of DLR covering the objectives

- to develop and examine high authority control systems,
- to realize an explicit model following control system for in-flight simulation application, and
- to utilize the in-flight simulation system for handling qualities studies.

The inherent high maneuverability and response bandwidth of the basic helicopter is an excellent precondition for realizing a high bandwidth in-flight simulator and for a high potential tool which can be used in the design process of future helicopter control system technology and in the establishment of credible rotorcraft flying qualities data, covering the agility and preciseness demands of future missions. Since 1982 the testbed is operated by the DLR. The realization of an in-flight simulator was started 1985 by designing a model following control system which specifically meets the objectives of in-flight simulation purposes [8,9]:

- high flexibility for variation of a broad spectrum of stability and control characteristics,
- good initial response characteristics for application in agile and precise maneuvering, and
- acceptable mid and long term response.

The most promising and challenging method of control system design is to force the basic helicopter to respond on the pilot's inputs as a commanded model. In principle, two concepts for model following control system (MFCS) can be distinguished as skeletonized in *Figure 9*. In an implicit MFCS, the control inputs to the host vehicle are formed from the vehicle response (x), the pilot input (u_p) and the controller. The controller can be composed of a feedback and a feedforward. The commanded model states (x_m) appear only in the performance criterion for the design of the overall system. The command model is implied in the controller which is designed to force the host vehicle to behave like the commanded model ($x = x_m$). Any variation in the commanded model needs a modified design of the control system which opposes to the required flexibility of the simulation system. An implicit model following system is well suited for the application in operational helicopters where only a small number of commanded model configurations is required.

In-flight simulation requests the use of explicit model following design. The commanded model response (x_m) is calculated explicitly from the pilot inputs (u_p) and is fed into the controller. The controller is not depending on the state and control matrix of the commanded model. To achieve a fast model following a explicit MFCS is most composed of a feedforward and a feedback network. The feedforward works as a compensator of the host helicopter dynamics and, ideally, the feedforward is the inversion of the host dynamics. The feedforward controller is calculated from a model of the host helicopter (state matrix A and control matrix B). A flight vehicle state feedback is implied to minimize the influences of noise and of feedforward inaccuracies and to reduce the tendency of long term drift in the model following. Consequently, an explicit model following control system is being developed for the ATTHeS in-flight simulator [10,11]. Greater details will be required, however, to reflect the depth needed to describe the integrated design of the ATTHeS MFCS. Besides the control laws, the elements of computing, actuating and sensoring influence essentially the overall performance.

The quality of the long term model following of ATTHeS is demonstrated in *Figure 10*. The commanded model for this example has been a decoupled rate response system. No tendency of drifts in the attitude signals, especially, can be recognized. The off-axis response in the roll attitude show an satisfactory level of decoupling. Compared to the remaining coupling in the ATTHeS, the basic BO 105 helicopter has a coupling ratio of nearly one in roll due to pitch.

In designing the ATTHeS MFCS, the feedforward system was emphasized to achieve an only slightly reduced quality of the initial response characteristics compared with the basic BO 105 helicopter. The calculation of the feedforward is based on an extended 8 DOF model of the BO 105 which has been identified from flight test data [12]. The extended model describes more accurately the short term response of the BO 105 in the roll and pitch axis which is characterized by a coupling of the main rotor tip path plane and the fuselage dynamics. The dynamic feedforward control is an exact inversion of the state space formulation of the basic helicopter model. The implementation of a well defined feedforward controller reduces the efforts necessary to design the feedback loops. In the ATTHeS approach, a classical network of proportional and integral controller loops are applied.

Figure 11 illustrates the effective time delays for the overall simulation system in the pitch and roll axis. The time delays of ATTHeS are assessed in comparison to first order rate command models in both axis. In addition, the effective time delays of other helicopter technology demonstrators or simulators, equipped with a high authority digital control systems, are compared in the diagram [13,14]. The main elements contributing to the overall time delays are specified. The effective time delays for the ATTHeS are about 110ms for the roll axis and about 150 ms for the pitch axis. The higher value in the pitch axis results from the slower response of the basic helicopter in this axis due to a higher moment of inertia. The computational time for the frame rate and refreshing rate contributes about 40ms. The pilot input shaping needs about 10 ms. This small value is only valid for a center stick as integrated in the ATTHeS. A more sophisticated ministick needs a more sophisticated data conditioning technique which yields higher values for the effective time delays as it is illustrated in the ADOCS result.

For an evaluation of the simulator bandwidth capability, the phase delay and bandwidth criteria, defined in the updated military handling qualities specification, can be quoted. *Figure 12* shows the roll axis bandwidth and phase delay of ATTHeS and illustrates the potential of the simulation system. Comparing the overall simulation system with the basic BO 105, a small reduction of about 1 rad/sec in the bandwidth and a small increase in the phase delay have been accepted. Nevertheless, the obtained short term response characteristics meet satisfactorily the military requirements and guarantee a broad capability to cover the expected spectrum of flight dynamics for future rotorcraft systems. Corresponding to the bandwidth assessments, the *Figure 13 and 14* show the frequency response characteristics of the basic BO 105 and the in- flight simulator ATTHeS. Eigenvalues characterizing the roll response of the BO 105 are:[1])

- dutch roll dynamics [0.32,2.86]
- coupled roll / rotor tip path plane dynamics [0.45,13.14]
- lead lag / air resonance $\frac{[0.045,14.94]}{[0.021,14.96]}$.

The frequency response of the simulation system shows a good matching of the commanded response up to the bandwidth frequency. The mismatch in the higher frequencies is resulting from the additional time delays and the lower relative stability of the closed loop roll response. Increasing gains in the integral controller especially reduce severely the relative stability of the closed loop system [15]. This underlines the demand to design a satisfactory feedforward which allows a low gain feedback system. For ATTHeS, the roll response eigenvalue is changed by the feedback gains to [0.23,12.90].

5. ATTHES UTILIZATION FOR HANDLING QUALITY STUDIES

The updated military handling qualities specification for military rotorcraft has been published in 1987 after five years of preparation. The specification is based on data and experiences from previous development programs and from recent theoretical and experimental studies [16]. The structure of the specification and the defined criteria have taken into account the new mission demands and the integration of modern cockpit and control system technologies. It is not surprising that the data available from previous programs and from recently performed studies cannot be adequate to verify all the criteria. Many topics have not been addressed up to now, data gaps are identified, other data are not credible or the spectrum of the variation for some parameters is not sufficient. A continuous activity is required to increase the data base, in general, and to actualize the requirements by considering new missions and advanced technology integration. In complement, data for the establishment of criteria regarding the specific civil operational aspects have to be generated.

As mentioned before, the generation of generic and credible handling qualities data insists on systematic pilot in-the-loop studies using general purpose test facilities with adequate characteristics. DLR operates the ATTHeS to cover these objectives. A test program, specifically planned to contribute to the data for the updated specification, was a study of the roll response behaviour required in the NOE slalom flight [17]. In these tests the roll control sensitivity and the roll damping had been varied. These test were recently conducted again with a slightly modified slalom task. Tracking phases have been integrated into the slalom task to address more the short term response characteristic in the piloting task and, consequently, in the pilot ratings and comments. With a first order rate response commanded in the pitch and roll axis, the parameters time delay, damping coefficient, and control sensitivity have been varied harmoniously in both axis. The damping coefficient is nearly equivalent with the bandwidth of the

[1]) Notation $[\zeta, \omega_0]$ implies $s^2 + 2\zeta\omega_0 s + \omega_0^2$

system. The time delay parameter influences both the phase delay and the bandwidth. *Figure 15* shows the test data together with the pilot ratings. This verification of the criteria boundary defined in the updated military specification argues for a change of the slope in the level boundaries. For low phase delays a shifting of the levels to higher bandwidth values can be recommended. Additionally, the test data indicate a generally different boundary slope for high phase delays. The level one boundary can be drawn to have a limited phase delay of about 0.15 to 0.2 sec for high bandwidth. This boundary slope would also be in a good agreement with the corresponding requirement in the military fixed wing specification. A broader variation of bandwidth and phase delay is of interest to support this statement. The variation of bandwidth phase delay configurations is limited using a commanded first order rate response model (see the mapping of the rate command configuration in figure 15). A test program is planned realizing attitude command model which allow a broader variation of bandwidth with higher phase delays.

Another aspect was considered in the tests. *Figure 16* demonstrates the influence of control sensitivity on the bandwidth evaluation. This parameter is not directly taken into account in the updated handling qualities specification. Only it is referred to the need to adapt the control sensitivity. From the DLR data, the requirement can be derived that the ratio of control sensitivity and bandwidth shall be nearly constant. For a first order rate system this ratio is quite well assessed with the control power measure.

A third topic which shall be discussed as an example is the definition of decoupling requirements. In the ADS 33 the pitch-to-roll and roll-to-pitch coupling requirement during aggressive maneuvering is defined in the time domain. The ratio of the peak off-axis response to the on-axis response is required for level one to a limit of 25 % within the first 4 sec. *Figure 17* shows the examination for the ATTHeS. With a fully decoupled command model the remaining coupling of the overall system is clear below the level one boundary for moderate and aggressive maneuvering. But for small amplitude pilot inputs the ATTHeS coupling level comes more closer to the required limits. Especially the coupling time history for the small control input points to an effect of augmentation systems which is not taken into account in the existing requirements. It can be distinguished between two types of coupling which should be also considered in the requirements. One examination of an augmentation system is the level of decoupling in the initial response. The short term decoupling will be performed mainly by a feedforward corresponding to the control coupling of an unaugmented helicopter. The feedback system is responsible for the quality of the mid and long term decoupling. Indeed, also pilots react upon the types of coupling with different control strategies. The control coupling (short term) is controlled by use of a crossfeed whereas the pilot controls the state coupling (mid and long term) like a feedback system. *Figure 18* illustrates the dependencies of the coupling of an unaugmented helicopter in the frequency domain. The frequency response of the roll rate due to pitch inputs indicates clearly, that a low frequency and high frequency coupling contribute to the coupling response.

6. CONCLUSIONS

In this paper the effects of new operational demands on the development and on the evaluation of highly augmented helicopters have been discussed. Key parameters describing the increasing requirements on

- maneuverability,
- tracking preciseness, and
- agility

have been associated with the definition of representative and reproducible flight test tasks and with the establishment of low risk test procedures for handling qualities investigations.

An in-flight simulator is an effective tool for generating generic and credible handling qualities data. DLR has developed the ATTHeS simulator which yields variable stability and control capability with a high level of flexibility. The measured performance of the implemented explicit model following control system illustrates the good potential of ATTHeS to simulate high bandwidth helicopter systems in agile and precise maneuvering.

Exemplary results of handling qualities studies indicate the need to verify some previously defined criteria. The bandwidth phase delay requirements for the initial response of the roll axis should be modified in the slope of the level boundaries. For the format and the measures of a coupling criteria it is recommended to consider a differentiation between the initial and the mid/long term response characteristics.

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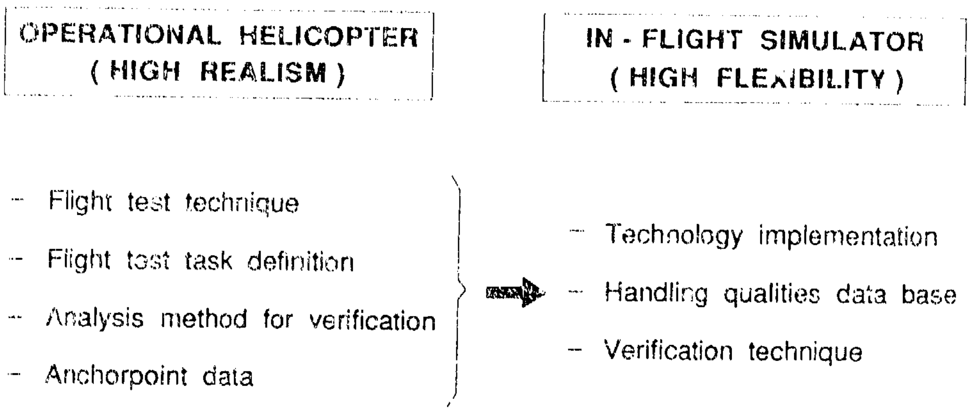


Figure 1. Use of testbeds

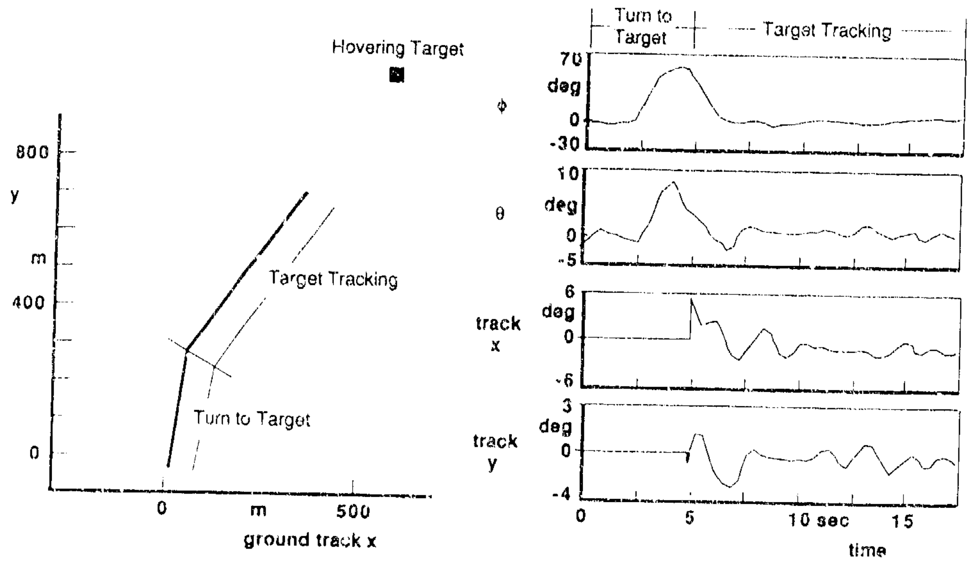


Figure 2. Bank to target maneuver

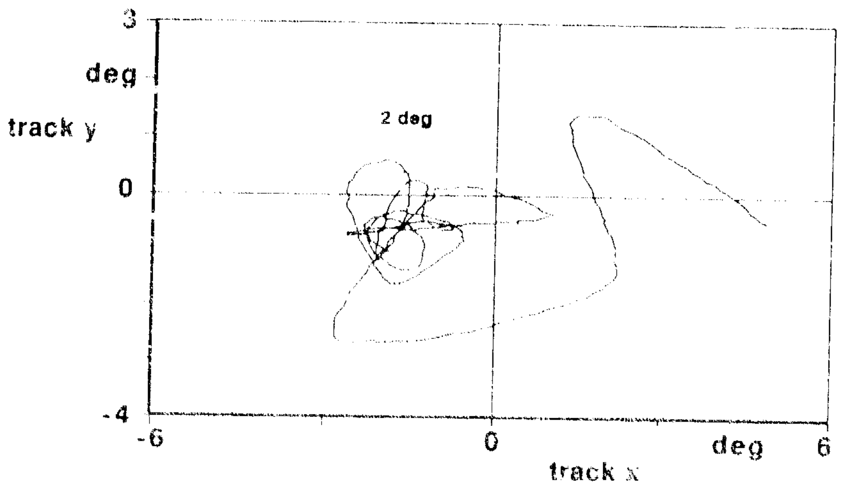


Figure 3. Tracking error crossplot

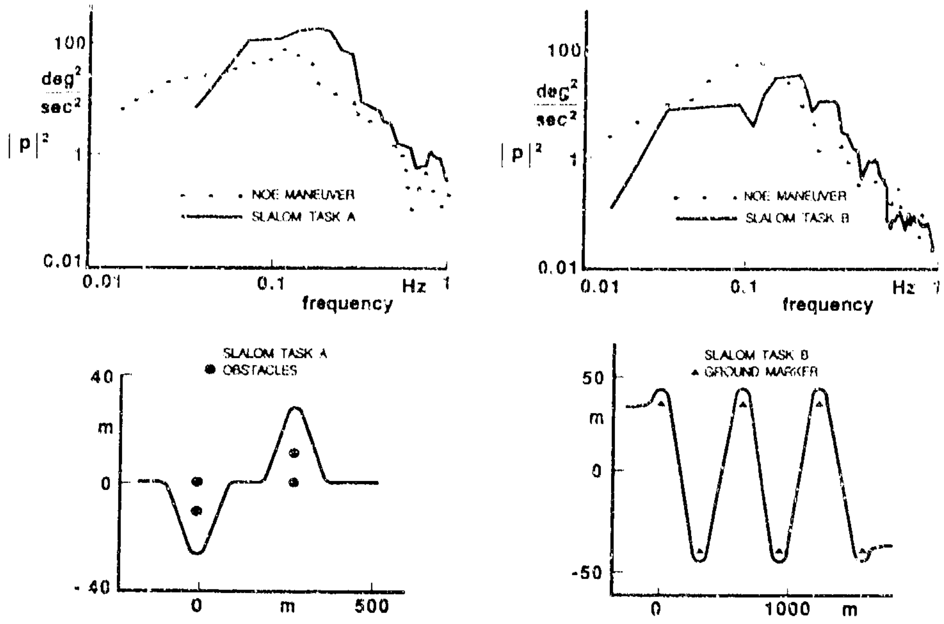


Figure 4. Slalom task representing NOE

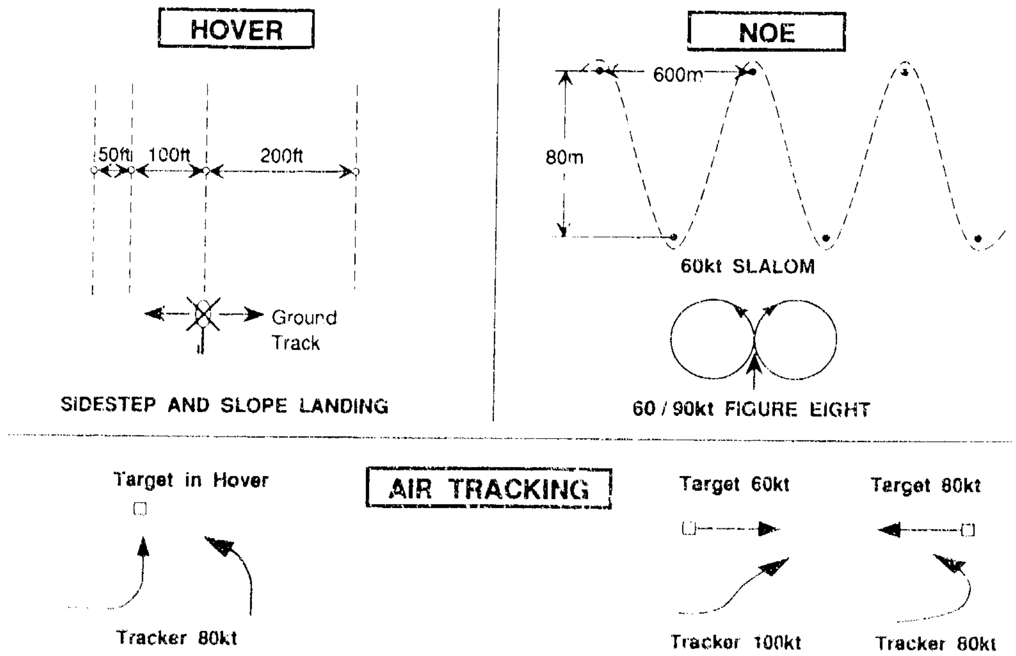


Figure 5. Roll axis flight tasks

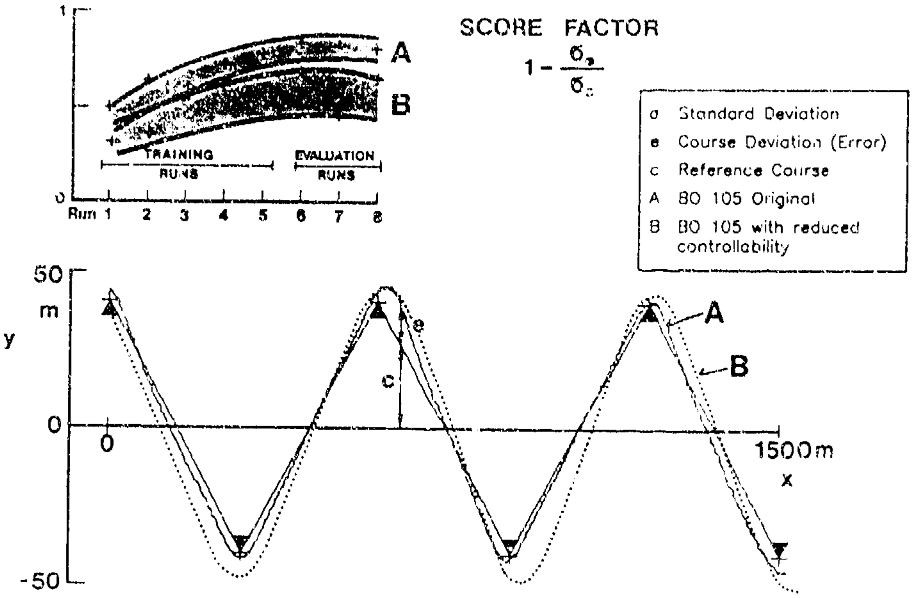


Figure 6. Slalom task quicklook

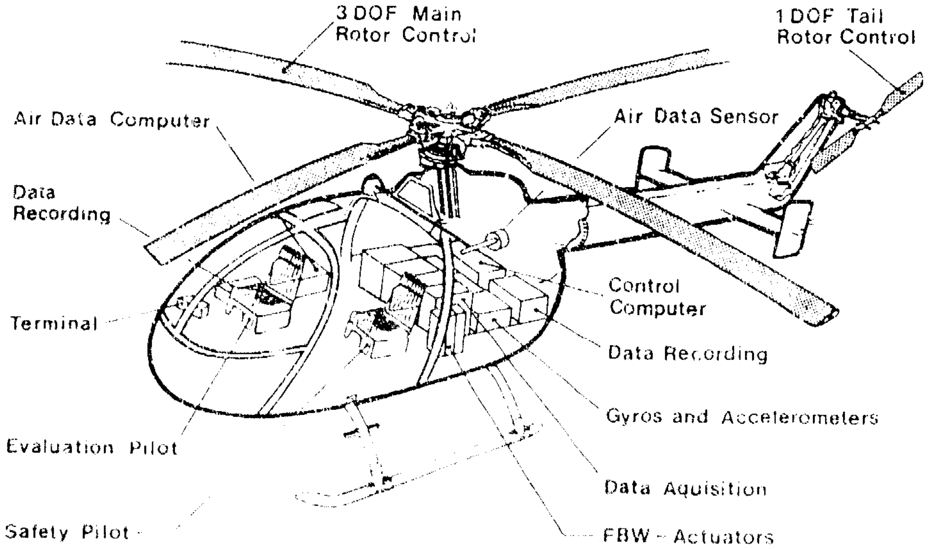


Figure 7. ATTHeS airborne simulator

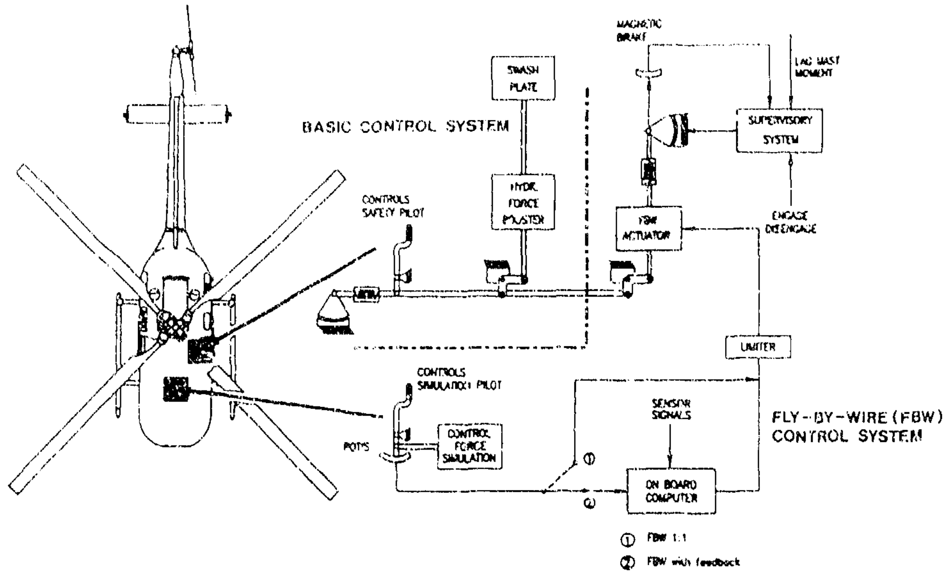


Figure 8. BO 103 S3 control system modification

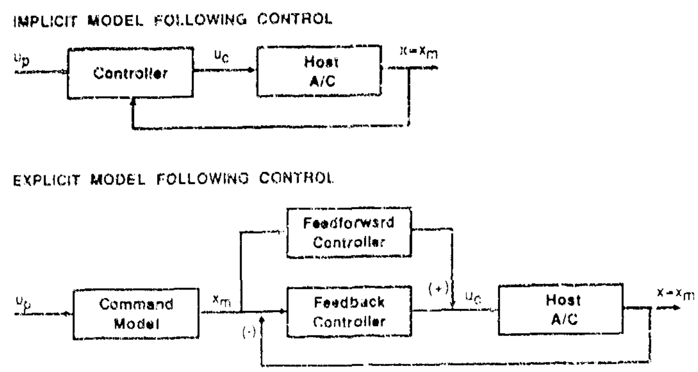


Figure 9. Concepts of model following

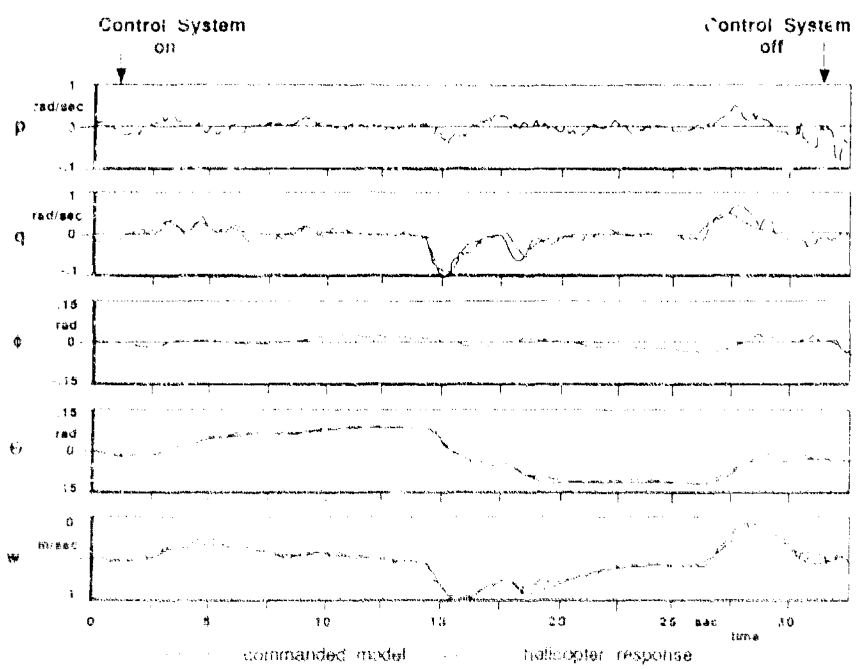


Figure 10. ATTHeS long term model following

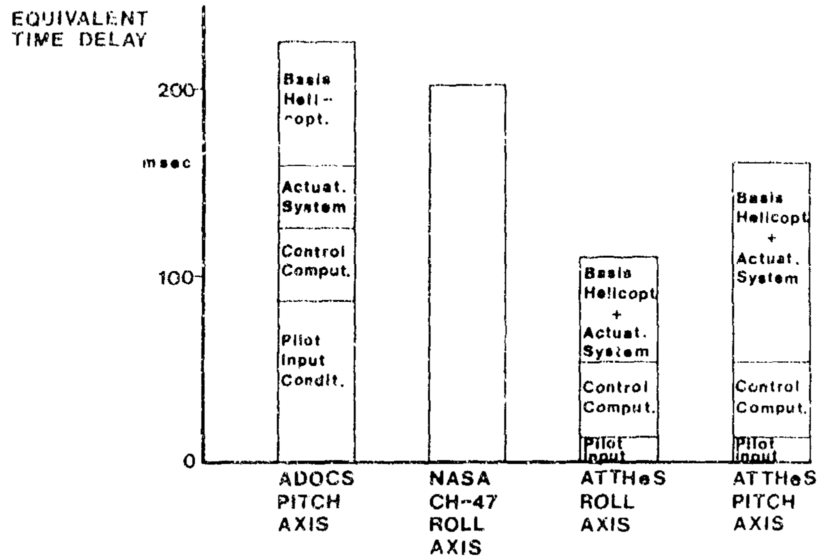


Figure 11. Effective time delays

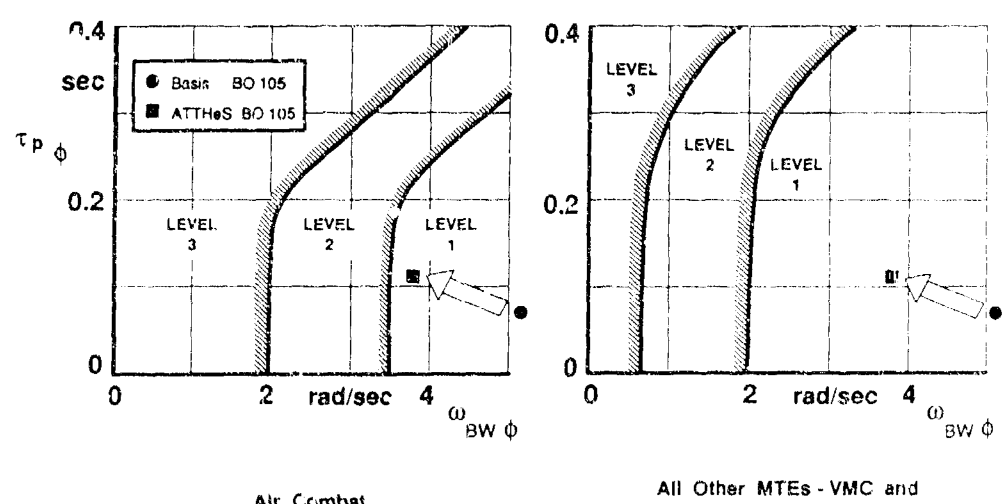


Figure 12. ATTHeS compared with bandwidth requirement

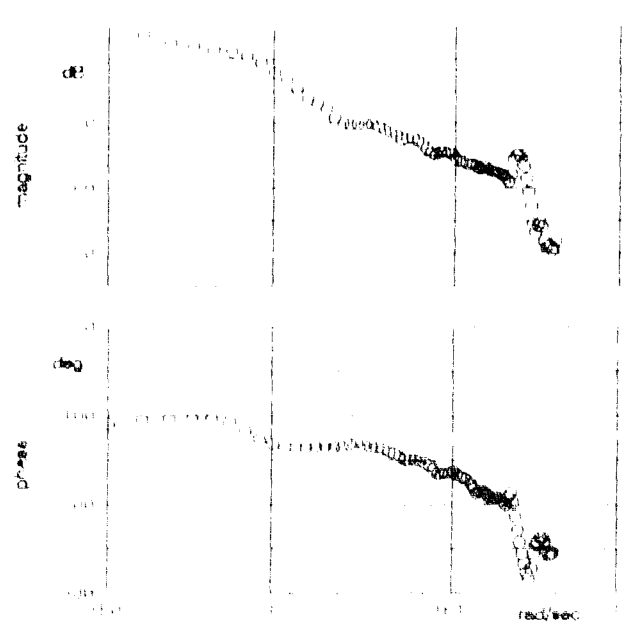


Figure 13. Frequency response $\left(\frac{\Phi}{\delta_y}\right)$ of basic BO 105

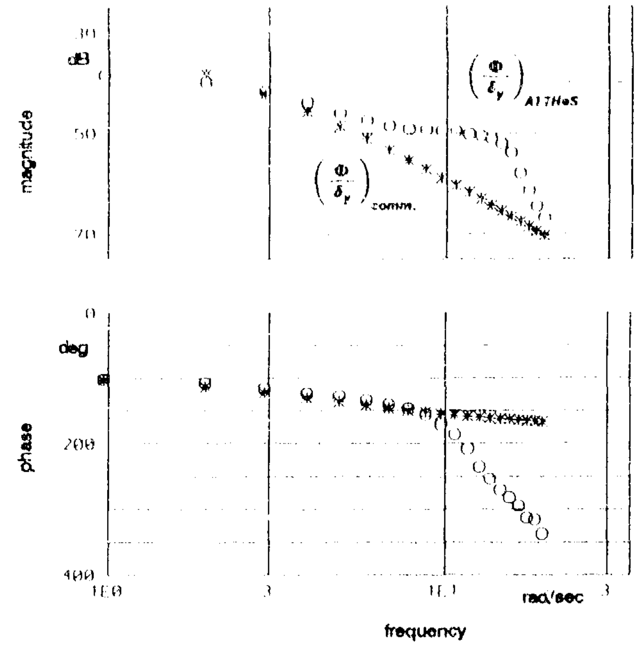


Figure 14. Frequency response of the overall system compared with the command model

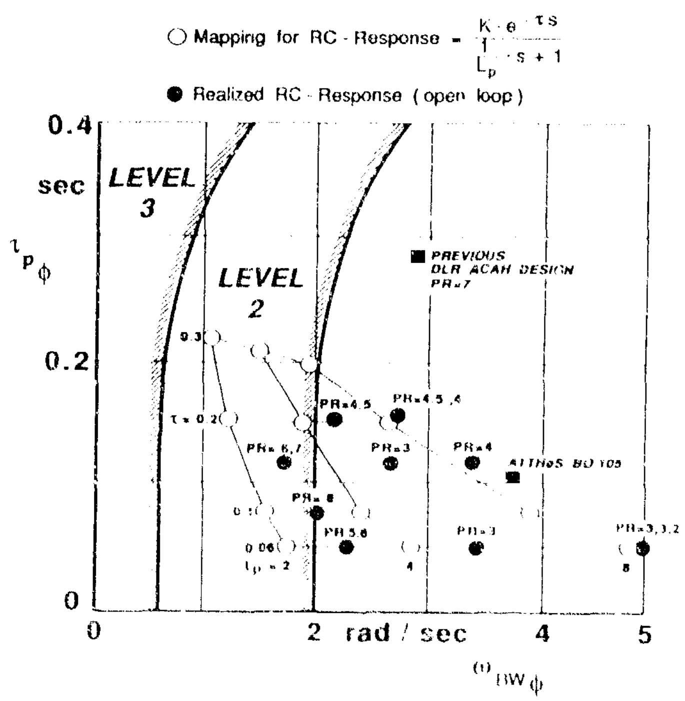
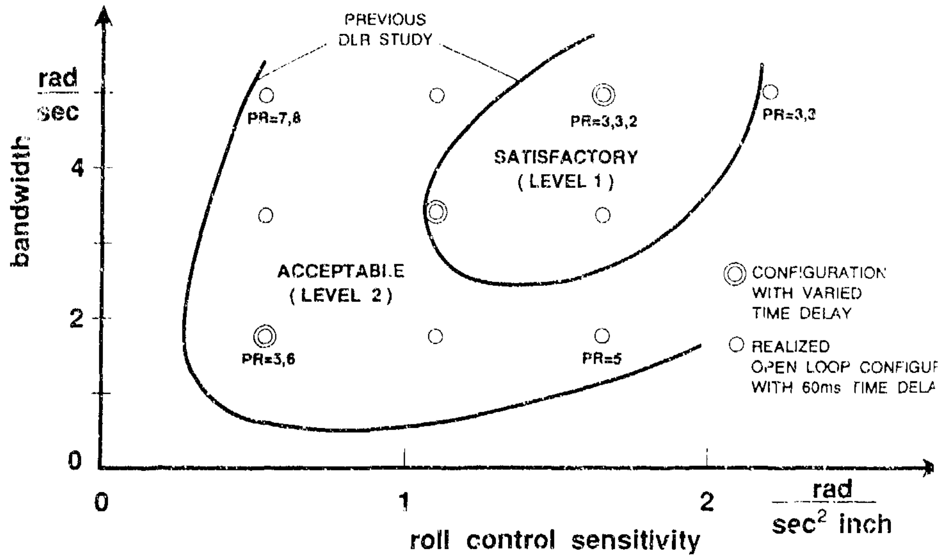


Figure 15. Verification data for roll bandwidth requirements

Figure 16.
Influence of
control sensitivity



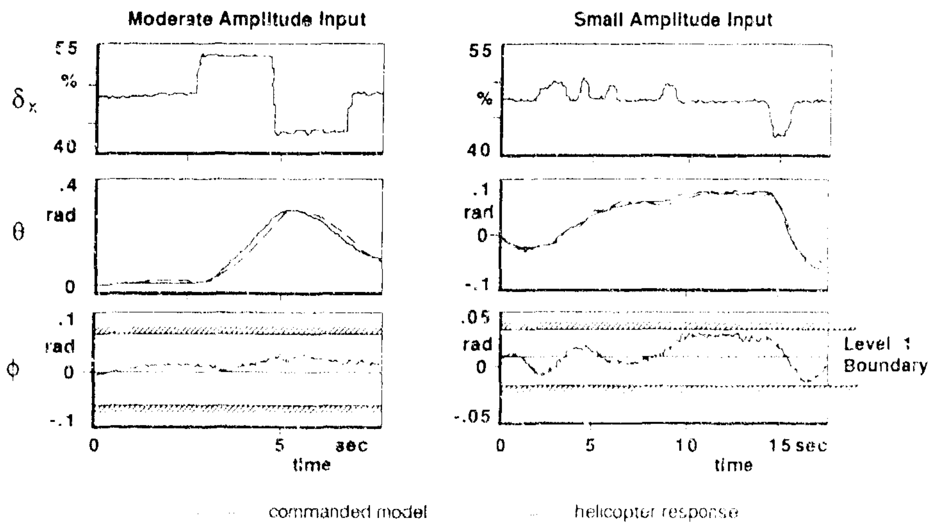


Figure 17. ATTHeS compared with coupling Requirement

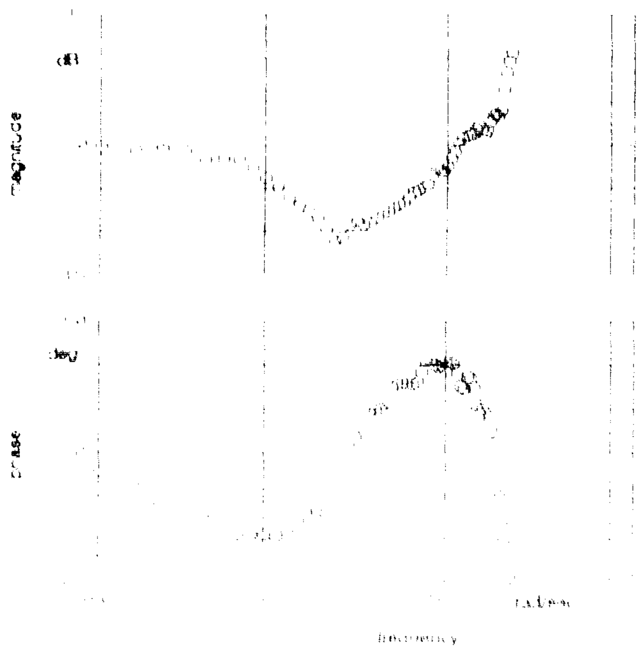


Figure 18.
BO 105 frequency
response of roll rate
due to pitch inputs
$\left(\frac{p}{q}\right)$ of basic BO 105

Agility: A Rational Development of Fundamental Metrics and their Relationship to Flying Qualities

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SUMMARY

The results of the first phase of a three year agility program are presented. In large measure, the work accomplished to date and reported in this paper has produced a highly viable approach for developing a rational concept of agility and, more importantly, for relating agility to the flight dynamics, maneuvering performance and to the design of aircraft. The flight mechanics of a rigid aircraft in three-space maneuvering flight are examined with respect to total velocity, acceleration and the time-rate-of-change of acceleration; the latter being assumed to correspond most directly with agility. In particular, the terms of the expanded "agility vector" are interpreted with regard to their potential for providing a rational basis for the evaluation of any given set of agility metrics and for suggesting, directly, a new set of metrics. A potential form of agility is offered for which a readily accceptable relationship is traced to both flying qualities and maneuvering performance. Finally, the remainder of the program is outlined to carry the work toward the development of a practical set of design guidelines for agility.

SYMBOLS

$\dot{a}_A$	Axial Agility	ft/sec^3
$\dot{a}_C$	Curvature Agility	ft/sec^3
$\dot{a}_T$	Torsional Agility	ft/sec^3
$\dot{a}_x$	x_B Agility Component	ft/sec^3
$\dot{a}_y$	y_B Agility Component	ft/sec^3
$\dot{a}_z$	z_B Agility Component	ft/sec^3
$\bar{b}$	Unit Bi-Normal Vector	–
$\dot{D}_{x,y,z}$	Drag Rate Components	lb/sec
$\dot{F}_{x,\ y,\ z}$	Force Rate Functions	lbs/sec
$\kappa = 1/r$	Curvature	1/ft
L	Lift/Rolling Moment	lbs or ft-lbs
ω	Turn Rate	rad/sec
$\dot{L}_{x,y,z}$	Lift Rate Components	lb/sec
$\bar{n}$	Unit Normal Vector	–
$\bar{R}$	Position Vector	ft
r	Radius of Curvature	ft
s	Flight Path Arc Length	ft
ρ	Radius of Torsion	ft
$\bar{t}$	Unit Tangent Vector	–
$\tau = 1/\rho$	Torsion	1/ft
$\dot{T}_{x,y,z}$	Thrust Rate Components	lb/sec
$\dot{Y}_{x,y,z}$	Side Force Rate Comps.	lb/sec

INTRODUCTION

It is difficult to point precisely to the origins of the recent wave of interest in agility but they are important to consider if we are to understand the reasons for seeking aircraft design metrics beyond those currently available to us. The combat paradox of "The Fighter Airplane That Could" is well known in the combat conscious community, i.e., the situation in which an aircraft of 'medium performance' frequently gains an advancage over its superior opponent (such observations having accounted for the differences in tactics and pilot skill levels). In many of these cases a strict comparison of the respective maneuvering performance charts of the two aircraft yield no surprising discoveries of EM areas where the medium performance aircraft might have a distinct advantage and so, the speculation of superior "agility" has often been suggested as a possible explanation for the apparent (?) anomaly. Another motivation for understanding and applying the concepts of agility stems from the trends in fighter aircraft designs. Today we are faced with the real possibility of designs which incorporate thrust vectoring as a means of providing useful control power well beyond normal aerodynamic limits, i.e., the X-31A. Such aircraft offer maneuvering capabilities which are unique and largely unmeasureable by ordinary maneuvering performance methods. Thus, once again, the speculation of agility arises as a potential means of providing us with the engineering design control required to predictably realize given levels of this "new" maneuvering capability. All of the above plus the awareness of the potential for direct force control and our growing understanding of high angle-of-attack aerodynamics, leads us quite naturally to, among other places, agility as a means of rationally comprehending and measuring both the new and unique as well as the normal maneuvering characteristics of the next generation of fighter aircraft which may emerge from today's developing technology base. It will be useful to establish a 'reference agility concept' early in this paper in order to provide a reasonably clear and receptive communicative environment for the reader. For this we will have to, at the very least, address the issues of definition.

There are almost as many definitions of agility as there are investigators in the field. This, of course, has made it difficult to compare the results of one investigator with those of another. The problem is more than just lack of standardization: it is worse...it is one of disagreement on the most fundamental level. There simply is very little agreement on what agility is! Despite the general acceptance of the broad definition that propounds that agility is the rate of change of maneuver state, specific interpretations depart widely from this straight forward concept. Proposed agility metrics vary considerably from Kalyisti's Point-and-Shoot (DT) parameter[1] through Skow's dPs/dt, $t_{d\theta}$, T.R./$t_{d\theta=90°}$[2], to Herbst's agility vector components, i.e., $\ddot{V}$, $(\dot{V}\omega + \dot{\omega}V)$, $(\ddot{X} - \dot{X}\sin\gamma - \dot{\gamma}\dot{X}\cos\gamma)$[3]. We find that contemporary metrics tend to fall into one of three categories: 1 those which closely resemble flying qualities design criteria, 2. those which are derivative forms of energy-maneuverability performance parameters and 3. those which are based on the differential geometry properties of the flight path. An important fourth category, a combination of the first two, is very prevelant. The work described in this paper summarily dismisses the first two categories on the basis that, although agility may be related to flying qualities and/or maneuvering performance, it is believed to be a uniqely independant flight dynamics characteristic and that is what is sought in this work. Figure 1 provides a perspective which one can use to differentiate between flying qualities, maneuvering performance and agility. It will be noted that, in addition to segregating flying qualities, maneuvering performance and agility, aircraft 'pointing' has been unambiguously assumed to be a flying qualities characteristic...and nothing more; although its relationship to agility is recognized along with other, equally important, flying qualities characteristics.

The point of view which will be presented is that agility is:

1. real, and not just another transformation of long existing engineering concepts,
2. unique, in that it is impossible to completely define and apply it through our present state of knowledge of either flying qualities and/or maneuvering performance, and
3. can be explicitly developed into manageable engineering terms which are relatable to 'proximity dynamic characteristics', such as flying qualities, etc. and eventually, to aircraft design.

AGILITY: A DEFINITION

Having indicated the wide disparity existing with regard to the definitions and conceptualization of agility, it becomes obligatory to offer yet another definition:

Agility Is A Property Which Characterizes The Time-Rate-Of-Change Of Maneuver State (Acceleration State) And Addresses, Exclusively, The Translation Of A Moving Body In 3-Space.

In this paper, the above definition is regarded literally, with all equations and their development being consistent with the definition. A set of corollary statements (assumptions) accompanies the definition:

* **Agility Is A Characteristic Exhibited By All Bodies In Motion**
* **"Fundamental" Metrics Apply Equally To All Bodies** (Controlled Or Uncontrolled)
* **Agility Sensibly Exists In Any And All Spatial Dimensions** (1, 2, 3, ...n)
* **Agility Is A Long-Term Property** (Non-Steady-States And Multiple Maneuvers)

One further qualification is required. Although the above definition and "groundrules" refer to a generalized rigid body, in this paper agility will be developed specifically for an aircraft. The basic equations and the operations performed on them are, of course, universal and can be applied to any rigid body in motion.

THE AGILITY VECTOR

The rate of change of maneuver state logically translates into the rate of change of aircraft acceleration and, in turn, to a possible rational development of agility. This approach reduces to the derivation, expansion, interpretation and application of $\dot{\bar{a}}$, i.e., the agility vector; its components, terms, etc. The agility vector will be developed in two different axis systems. One will be, strictly speaking, a differential geometry approach and will be derived in the Frenet, or flight path system while the other will be derived from the Newtonian system, i.e., from the consideration of $\bar{F}=m\bar{a}$ and $\dot{\bar{F}}=m\dot{\bar{a}}$. With the Frenet approach, it must be emphasized that only the flight path geometry of a point-mass aircraft is considered. Although some few investigators have confined their studies to the Frenet system exclusively, the Newtonian development is very revealing and offers considerable insight into the aircraft design characteristics which may affect agility directly (through the development of $\dot{\bar{F}}$). Both forms collapse into the same total agility vector (with proper coordinate transformations). The Frenet vector is the simpler of the two and allows us to see relationships to maneuvering performance more readily. The Newtonian vector and its associated $\dot{\bar{F}}$, permits us to more readily associate flying qualities characteristics and aircraft design implications with its components and terms.

The Frenet Development

Figure 2 presents a small segment of a general maneuvering flight path wherein no assumption of steady state is implied, i.e., the entire maneuvering state is assumed to be continually varying with time. The position of the rigid body along the flight path is defined by the position vector, $\bar{R}(t)$.

The familiar equations for total velocity and acceleration follow:

$$\overline{V} = \dot{\overline{R}} = \dot{s}\overline{t} \qquad (1)$$

$$\overline{a} = \ddot{\overline{R}} = \ddot{s}\overline{t} + \dot{s}^2\kappa\overline{n} \qquad (2)$$

Figure 2 shows that $\overline{V}$ is coincident with the $\overline{t}$ unit vector and that $\overline{a}$ lies in a plane defined by the $\overline{t}$ and $\overline{n}$ unit vectors. The agility vector is obtained by differentiating equation (2) with respect to time,

$$\dot{\overline{a}} = \dddot{\overline{R}} = \dddot{s}\,\overline{t} + \ddot{s}\; d\overline{t}/dt + (2\dot{s}\ddot{s}/r - \dot{s}^2\dot{r}/r^2)\overline{n} + \dot{s}^2/r\; d\overline{n}/dt. \qquad (3)$$

Substituting for $d\overline{t}/dt = \dot{s}/r\overline{n}$ and $d\overline{n}/dt = -\dot{s}\kappa t + \dot{s}\tau b$, we get

$$\dot{\overline{a}} = (\dddot{s} - \dot{s}^3/r^2)\overline{t} + (3\dot{s}\ddot{s}/r - \dot{s}^2\dot{r}/r^2)\overline{n} + (\dot{s}^3/r\sigma)\overline{b}. \qquad (4)$$

Equation (4) is the flight path axes or Frenet version of the agility vector and can be seen to consist now of three components directed along the $\overline{t}$, $\overline{n}$ and $\overline{b}$ unit vectors, respectively. Figure 2 shows this clearly. If we use curvature (κ) and torsion (τ) instead of their reciprocal equivalents we get what may be a more familiar version(4) of equation (4), i.e.,

$$\dot{\overline{a}} = \dddot{\overline{R}} = (\dddot{s}-\dot{s}^3\kappa^2)t + (3\dot{s}\ddot{s}\kappa + \dot{s}^2\dot{\kappa})\overline{n} + \dot{s}^3\kappa\tau\overline{b}. \qquad (5a)$$

$$\dot{\overline{a}} = \dot{a}_A\,\overline{t} + \dot{a}_C\,\overline{n} + \dot{a}_T\,\overline{b} \qquad (5b)$$

Referring to either equation (4) or (5), the $\overline{t}$, $\overline{n}$ and $\overline{b}$ components are interpreted as the <u>axial</u>, <u>curvature</u> and <u>torsional</u> agility components, respectively. Each of the three components can be thought of as representing three distinctly different translation path types along the differential arc segment "AB" in Figure 3. Figure 3 provides a vizualization of the Frenet components. The illustration is interpretively useful in that one can begin to get a "feel" for the make-up of an elemental translation in three-space, for the roles of each of the terms within the components and lastly, for the types of control inputs/responses the aircraft would have to experience in order to effect a given translation or portion thereof.

The Newtonian Development

If we look at agility, i.e., the time rate of change of the translational acceleration of the aircraft, in a Newtonian or body-axis system of equations, we can begin to see not only an expanded new form of the agility vector but also the forces (aerodynamic, thrust, etc.) upon which agility depends. This, in turn, may provide us with a means through which we can ultimately derive tangible aircraft design control over agility. The development is interesting and revealing.

Beginning with

$$\overline{F} = m\overline{a} \qquad (6)$$

and taking the time derivative of both sides,

$$\dot{\overline{F}} = m\dot{\overline{a}}, \qquad (7)$$

we immediately get the fundamental relationship that agility depends primarily on the <u>transient</u> behavior of all of the forces acting upon the aircraft. Agility does not necessarily, if at all, depend on the quasi-steady state of the forces as we normally treat them in solving equation (6) as we typically do to obtain response solutions to given control inputs. Futhermore, the relationship explicitly stated by equation (7) is valid whether the forces are linearly or non-linearly represented. It can be seen from the outset that the above concept can be very effective in unraveling the "puzzle" of agility. The elusiveness of the understanding of agility lies in the realization that the phenomenon is generated through what is probably the short-lived but highly influential transient behavior of the forces. Dwelling on this for a moment, we can well understand how difficult it is for a pilot to be expected to evaluate a dynamic characteristic which he cannot possibly be able to directly detect. All we can expect is that the pilot will be able to comment on the longer term effectiveness of this seemingly intangible characteristic and not so much on agility per se. That one observation tells us that the problem is not simple and also that the implications on past and future flight simulation or flight test experiments is vast. Let us take a closer look at equation (7) and develop it to a reasonably interpretable form.

If we perform the implied time derivation on equation (7) we obtain

$$\dot{F}_X\overline{i} + \dot{F}_Y\overline{j} + \dot{F}_Z\overline{k} = m(\dot{a}_X\overline{i} + \dot{a}_Y\overline{j} + \dot{a}_Z\overline{k}), \qquad (8)$$

where the components of the rates of change of forces and the components of the new form of the agility vector are:

$$\dot{F}_X = \dot{L}_X + \dot{D}_X + \dot{Y}_X + \dot{T}_X + \ldots - mg(\cos\theta)q \qquad (9a)$$

$$\dot{F}_Y = \dot{L}_Y + \dot{D}_Y + \dot{Y}_Y + \dot{T}_Y + \ldots + mg[(\cos\theta\cos\phi)p - (\sin\phi\sin\theta)q] \qquad (9b)$$

$$\dot{F}_Z = \dot{L}_Z + \dot{D}_Z + \dot{Y}_Z + \dot{T}_Z + \ldots - mg[(\cos\theta\sin\phi)p - (\cos\phi\sin\theta)q] \qquad (9c)$$

$$\dot{a}_X = [\ddot{u} + 2q\dot{w} - 2r\dot{v} - u(q^2 + r^2) - p(rw - qv) + w\dot{q} - v\dot{r}] \qquad (10a)$$

$$\dot{a}_Y = [\ddot{v} + 2r\dot{u} - 2p\dot{w} - v(p^2 + r^2) - q(pu - rw) + u\dot{r} - w\dot{p}] \qquad (10b)$$

$$\dot{a}_Z = [\ddot{w} + 2p\dot{v} - 2q\dot{u} - w(q^2 + p^2) - r(qv - pu) + v\dot{p} - u\dot{q}]. \qquad (10c)$$

Equations (9) explicitly provide us with the time rate functions of lift, drag, sideforce, thrust, etc. as well as the effective time rate of change of the weight vector direction. Equations (10) present the body axis components of the agility vector and can be seen to contain the axial, curvature and torsional components of the Frenet system in a much more distributed and less comprehensible form. The single axial Frenet component term, $\ddot{s}$, for instance, is seen to be contained in the x, y and z components of $\ddot{u}$, $\ddot{v}$ and $\ddot{w}$, respectively. Equations (9) and (10) are offered principally for the benefit of the left side, i.e., the force rate terms. Closer examination of all components and terms of the left side could eventually provide a usefully different interpretation of the agility vector. Certainly this is the case when attempting to interpret the roles of the familiar aircraft states of $\dot{p}$, $\dot{q}$ and $\dot{r}$ and p, q and r throughout aircraft responses to controls applied during a set of maneuvers being examined for agility. So, both equations (10) and (9) can be of value in agility investigations.

Conspicuous in their absense from the equations given above, are the moment equations, L, M and N. Adhering to the definition already given which associates agility exclusively with translational dynamics, how an aircraft rotates, or even if the aircraft has sufficient control power to rotate is of no direct relevance in describing "agility". How the lift, drag, etc. vary throughout aircraft rotations is, of course, considerably relevant because these forces directly influence agility; but we do not have to know L(t), M(t) and N(t) per se. It is not necessary to compound the confusion which agility normally presents by introducing the moment equations. The flying qualities of the aircraft can well handle the moment generation aspects related to agility or to any other specific dynamic characteristic of the aircraft. That is why, although contrary to popular notions, an aircraft with a thrust vectoring system capable of providing impressive contol power especially in the absense of aerodynamic control...is not more agile because it can rotate (yaw/roll) faster, or at all, but because the main thrust direction can be altered much more rapidly and therefore the three-space translation characteristics of the aircraft are enhanced. This is hardly a semantic arguement. It is a logical conseqence of the ordering of the "agility" and "flying qualities" of the aircraft in such a way as to provide us with unambiguous design control, free of the conflicts inherent in most contemporary concepts of agility.

EXPLORATION OF $\dot{F}_{x,y,z}$

A further look at the force rate coefficients in equations (9) is in order. We will be looking mainly at the effect of $\dot{L}_z$ and $\dot{L}_y$ in the simplified form of L and $L\sin\dot{\phi}$. In order to do this let us examine the manner in which an aircraft gets from point A to point B (say in the same plane) through a curved flight path achieved by first rolling in at A and subsequently rolling out again so that the aircraft is once again in a wings-level state at B. The transient character of the lateral force build-up ($L\sin\phi$), first to accelerate the aircraft toward B and then the lateral force attenuation to decelerate the aircraft in order avoid overshooting B, is of paramount importance in determining the agility inherent in this maneuver. Obviously, throughout the maneuver, the flying qualities are of equal importance in providing the control power, damping, etc. for acceptable response and precise control. For the moment, however, let us just concentrate on what happens to the tilted lift vector in order to obtain the required lateral translations.

Figure 4 presents the constructed time history of this particular maneuver in terms of δ_a, $\Delta\alpha$, ΔL, Φ, $L\sin\Phi$ and $\dot{L}\sin\Phi$. The traces are "broken" at each of the two relatively long duration, steady-state (constant Φ) turns. Also shown in the figure is a top-down view of the 'planar' maneuver, with a number key which corresponds to significant events on the time traces. A small amount of back-stick has been assumed in order to increase angle-of-attack and thereby attempt to compensate for inherent altitude losses. Although the traces are not absolutely faithful to an actual maneuver they will serve adequately, in this synthetic case, to demonstrate the nature of 'lateral agility'.

A close look at $L\sin\Phi$ and $\dot{L}\sin\Phi$ reveals the manner in which the lateral side force and force rate due to $\Phi(t)$ are developing throughout the maneuver. The force rate term, in particular, can be thought of as being the dominant term driving $\dot{a}_y$ (from equation (10b)). Readily apparant is the fact that the $\dot{L}\sin\Phi$ trace achieves a maximum value at the point of inflection of the turn maneuver. One can justifiably say that lateral agility is maximum at this point, which intuitively feels very comfortable; although the equations suggest it as well. **The traces also show us how we may effect design control over agility.** If all the state parameters were computed from an actual set of aero data and examined, along with the agility terms from either equations (5) or equations (10) . far more complete analysis of this maneuver would be possible and would certainly yield effective correlations between L, D, T, α, Φ,....(t) and the agility components and terms. A complete dynamical analysis, including a 6 D.O.F. set of equations with fully coupled terms and non-linear aero is planned to begin soon in this study for the above maneuver and several other families of selected maneuvers. The relative ease of examining the relationship of flying qualities to agility is apparant from both the equations and figure 4

RELATIONSHIP TO FLYING QUALITIES

Although the sensitivity of specific flying qualities criteria/parameters has not yet been performed, figure 4 graphically suggests that all time-domain-based criteria can be readily and directly correlated with agility. In the case cited above, for instance, if we were to look at a complete set of roll performance traces (p, Φ, etc.) we could draw direct relationships between the parameters of roll dynamics and agility. Further, **we could re-shape roll responses in order to optimize agility, whether through aerodynamic design or through control law tailoring.** Thus, we have the beginning of practical design control for agility. It was evident even during the construction of figure 4 that changes in $\alpha(t)$ and $L(t)$ could dramatically affect $L\sin\Phi$, i.e., agility. In a similar manner and to a far better degree of effectiveness, one could begin with a full set of dynamic equations, generate families of maneuvers and, using this data base, compute the agility terms, the flying qualities and the maneuvering performance and be able to relate any one part of the analysis to any other part and to the whole as well. This is a degree of design and analysis freedom that the "agility community" has not had to date.

RELATIONSHIP TO MANEUVERING PERFORMANCE

The agility vector, Equation (5a), can be manipulated and rearranged in several revealing ways. If, for instance, we recognize that $\dot{s}/r = V/r = \omega$ and divide equation (5) by g, we obtain the following:

$$\dot{\bar{a}}/g = (\ddot{V}/g - \omega n_z)\,\bar{t} + (3\omega n_x - \omega^2 r^2 \dot{\kappa}/g)\,\bar{n} + (\omega V^2/g)\tau\,\bar{b} \qquad (11)$$

The first term of the normal component, $\bar{n}$, i.e., $3\omega n_x$, can alternatively be expressed as

$$3P_s/r \text{ or } 3\kappa P_{sk}, \qquad (12)$$

where P_{sk} = "kinetic specific excess power", i.e., $P_s - \dot{h}$.
Mulliplying through again by g to eliminate the awkward $\dot{\bar{a}}/g$, we obtain

$$\dot{\bar{a}} = (\ddot{V} - \omega n_z g)\,\bar{t} + (3\omega n_x g - \omega^2 r^2 \dot{\kappa})\,\bar{n} + (\omega V^2 \tau)\,\bar{b}. \qquad (13)$$

Now we have an expression for the total agility vector which is directly related to all of the terms commonly used in describing maneuvering performance, i.e., V (airspeed or Mach no.), ω (turn rate), n_z (normal load factor) and, for all intents and purposes, P_s (specific excess power). In addition, the axial load factor, n_x, appears in the curvature component and even the torsional component begins to look a little friendlier. In fact, the entire agility vector is friendlier and more easily interpretable.

CONCLUSIONS

This work, to date, has produced a rational and well structured model for agility. It has been consistent throughout and has neither supplemented existing ambiguities in contemporary agility concepts or introduced any new ambiguities. Futhermore, this agility model has been shown to be easily and directly related to both flying qualities and maneuvering performance and, quite naturally, to aircraft design. As such, the model is expected to serve well in continuing on with the investigation of aircraft agility and, further, offers the promise of doing so in an engineeringly practical manner.

Subsequent phases of this study will include the examination of a wide variety of existing and newly generated off-line, manned simulation and flight test data bases for an equally wide range of fighter aircraft types. In addition, existing, contemporary "agility metrics" will be re-examined in terms of this agility model. These relationships with both flying qualities and maneuvering performance, incompletely developed in this paper, will be expanded considerably and, in turn, a more unified set of "flight dynamics design criteria" will be sought to replace the fractured and minimally adequate flying qualities design criteria currently in use for high angle-of-attack flight conditions, i.e., for up-and-away combat environments.

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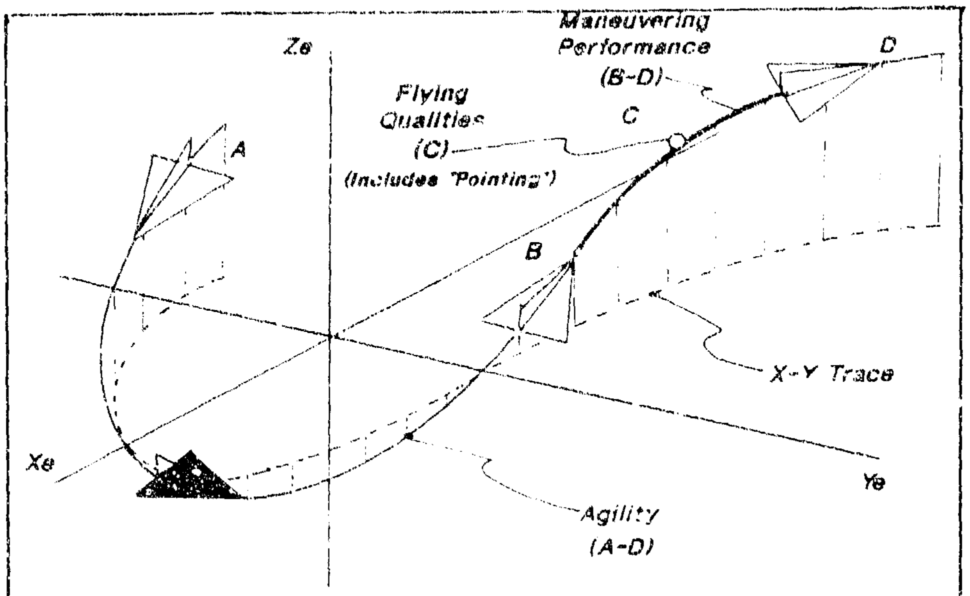

Figure 1. Time/Spatial Relationship Between Flying Qualities, Maneuvering Performance and Agility

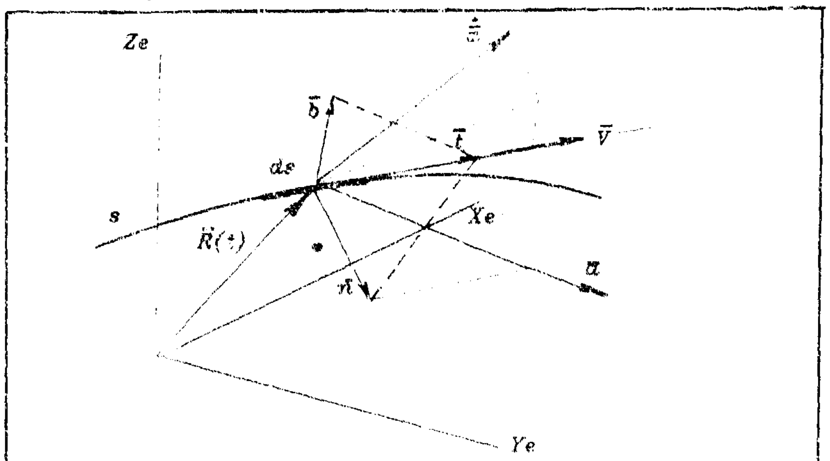

Figure 2. $\bar{V}$ and $\dot{\bar{A}}$ in the Frenet Axis System, Showing the Moving Trihedron, $\bar{t}$, $\bar{n}$, $\bar{b}$

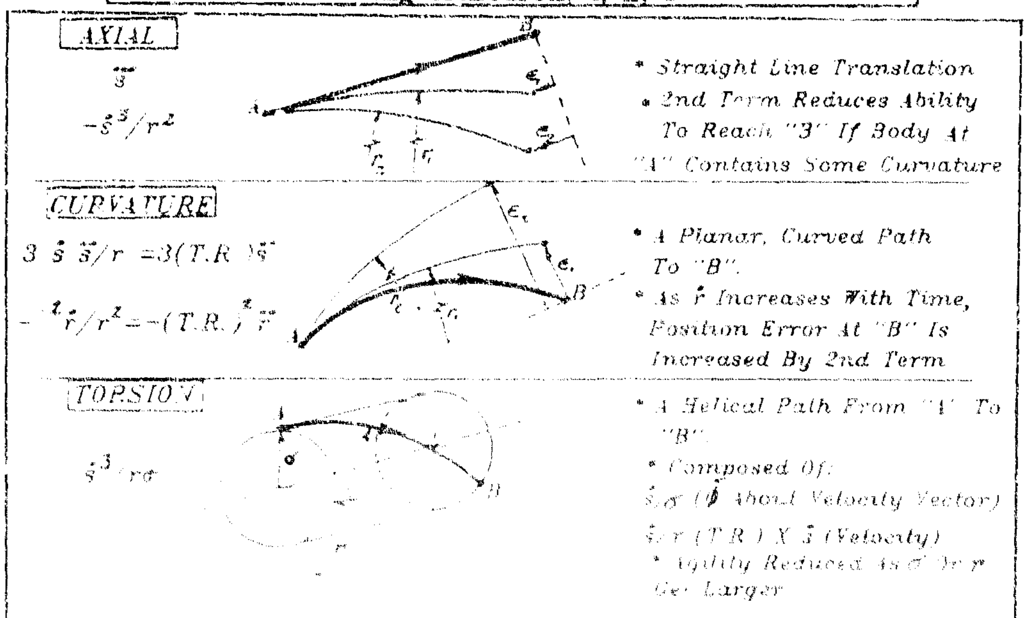

Figure 3. Visualizing The Frenet Components

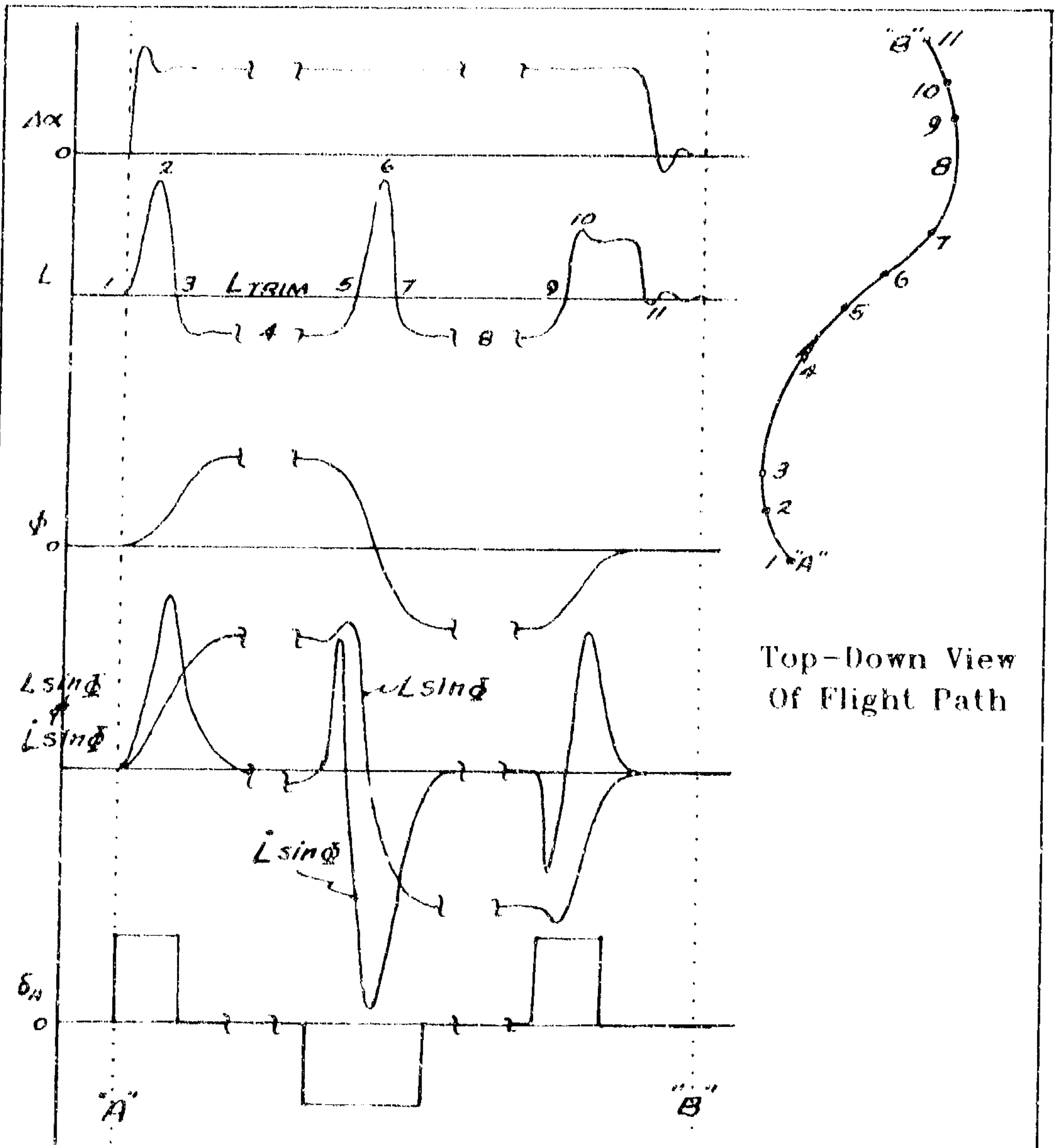

Figure 4. Constructed Time Traces Of A Turn Reversal Maneuver, Showing Lateral Force And Force-Rate Development

A REVIEW OF HIGH ANGLE OF ATTACK REQUIREMENTS FOR COMBAT AIRCRAFT

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SUMMARY

Design of an aircraft for use at high angles of attack can have major implications on the configuration which is chosen. The objective of this paper is to review the implications of designing for high angle of attack on configuration. This naturally leads onto consideration of agility and the criteria which could be used in the early design stages to ensure an aircraft is adequately agile. A number of questions are raised which cannot, as yet, be answered on a general basis. Some questions will not be answered until further research is put in hand and the results of existing experiments are made available.

1 INTRODUCTION

The objective of this paper is to review the implications of designing for high angle of attack with particular emphasis that this has on the basic choice of aircraft configuration.

Good handling qualities are the result of combining good basic aerodynamic characteristics with a robust, tolerant flight control system. When an aircraft is 'agile' and required to use high angle of attack in combat manoeuvring in order to gain tactical advantage over an opponent, these interdependencies become particularly evident.

The overall choice of configuration is achieved by a blend of a number of often conflicting requirements. This paper will examine configuration aspects which relate specifically to handling qualities and high angle of attack manoeuvring. However, the paper proposes that in relation to 'agility' the examination must go further then just aerodynamics and handling qualities.

It is suggested that a better understanding of what is meant by 'agility' can be gained from consideration of the total weapon system capability, in which the aircraft manoeuvre capability is just one factor. 'Fighting qualities' might be a better term to use.

A number of questions are raised, which cannot be properly answered on a general basis. Answers must be decided on an individual project basis. Indeed, some questions will not be answered until further research is put in hand and the results of existing research are made available.

2 HIGH ANGLE OF ATTACK - WHY ARE WE INTERESTED AND WHAT DO WE MEAN?

Interest in high angle of attack (AoA) has always been associated with defining the limits on aircraft manoeuvrability. Historically, this was typically associated with investigating the wing stall and associated phenomena and any higher angle of attack work related to the spin and recovery behaviour.

However, over the last twenty five years, aerodynamic and control systems technology have progressed such that, for many aircraft, wing stall is not a limit under many conditions and maximum lift occurs at significantly higher AoA.

All aircraft are required to fly at maximum usable lift, or within a margin thereof, at some point during their operation. The term, high AoA, will mean different things depending on the design role of the aircraft, but it will generally be associated with the AoA regime around maximum lift and above.

Combat aircraft, and combat trainers, are frequently called upon to operate at maximum lift, or close to it. Figure 1 illustrates a typical combat scenario, starting at a moderate speed and altitude. If the initial engagement does not result in a kill and the opponents do decide to mix it, the fight will inevitably degrade to low speed, maximum lift conditions as the opponents seek for the firing opportunities.

This plot was generated many years ago, but it captures the essential features which remain true for this type of one-on-one engagement.

Obviously the better an aircraft's control around maximum lift, then the less likely a mistake in handling the aircraft is to lead to the opponent winning by default. Such losses were all too common in the years between 1940 and 1970, with advantage changing from one side to the other as technology developed.

Figures 2 and 3 illustrate the typical features which limit the conventional manoeuvrability of an aircraft. The overall constraints are provided by design strength limits and maximum usable lift. Figure 2 indicates how this relates to turn rate, or the ability to turn the aircraft's velocity vector, without rolling the aircraft. Usable lift equalling maximum lift provides the other upper constraint and use of higher AoA may increase normal force, but reduce the turn rate. Thrust provides a limit on the ability to sustain the turn, due to the increasing induced drag.

Figure 3 looks at the type of phenomena which could limit usable lift. Some configurations may become uncontrolled or exhibit unacceptable handling qualities before maximum lift is achieved. Good aerodynamic and flight control system design should ensure that this is not the case, but it may cause compromise of other aspects of the aircraft configuration design. As examples, a large fin which provides high AoA directional stability may compromise aircraft signature, or nose shape, which also influences directional stability and yaw damping (reference 1), can be influenced by radar installation and performance requirements.

As noted earlier, high AoA will mean different things depending on the role the aircraft is being designed for. Figure 4 categorises the various types, indicating what would be considered as high AoA for each. On a transport design, high AoA would be likely to embrace the wing stall i.e. anywhere between 15^{o} and perhaps 30^{o}, depending on the sophistication of the wing high lift devices. Equally, a combat aircraft designer would now define it as the region beyond maximum lift, i.e. from 25^{o} upwards, and this would embrace post stall manoeuvre, loss of control or spinning as appropriate.

Technology developments in aerodynamic design and in design of flight control systems have enabled substantial increases in the maximum AoA that can be contemplated without loss of control, as shown in figure 5, and described in reference 2. Only a few years ago, the high AoA message that could be given to any pilot was 'the aircraft spins, so don't!', as even if it recovered quickly, the altitude loss and spatial disorientation could still result in an accident. (And deliberate spinning ceased to be a viable combat manoeuvre very many years ago, if it ever was!)

Now, a totally different scene is emerging. Aircraft can be designed which are protected from loss of control; some may not lose a lot of altitude in high AoA, but it really depends on the duration of the manoeuvre as to how much energy is lost, either kinetic or potential.

3 AIRCRAFT CONFIGURATION DRIVERS

It is self evident that the major influence on an aircraft's configuration is the role for which it is designed.

Let us briefly look at how the configuration is affected by the choice of role and related requirements.

Firstly, the role will dictate payload, range, performance and manoeuvrability requirements. It will now almost certainly include requirements relating to aircraft detectability, i.e. signature, covering radar, infra-red, optical and possibly acoustic signatures. It will also dictate any supersonic requirements.

Resulting from these requirements, the engineers will decide on wing planform, wing area and wing sweep. They will also decide on whether the aircraft should have a tailplane, canard surface, both or be tailless. This will also relate to the stability levels which can be assumed, as these are dictated by consideration of trim, performance and control system gain capability. The level of complexity required in the FCS and associated electrical and hydraulic supplies is determined by whether the basic aircraft is stable or unstable, the need for carefree handling and the AoA range to be covered. Signature will play a part in dictating control positioning and general 'shape'. A choice will be made regarding weapon payload, whether it should be internal or external, which again affects size, performance and signature. In turn, these affect the choice of engine and engine cycle. Finally, at least as far as we are concerned within this paper, the manoeuvrability requirements, in combination with sizing and planform, dictates the high AoA use of the aircraft.

In deciding the AoA the aircraft must fly to, this again is a parameter which, because of the need to retain good control characteristics, will impact on wing planform, aircraft shape and layout, control sizing and chosen stability levels.

As can be seen, there are a large number of aspects that the design engineers must consider in the formative stage, and a significant number of the considerations will impact directly on the behaviour of the aircraft at high angle of attack. Figure 6 summarises the various aspects that have to be considered.

4 CONFIGURATION AND RELATION TO HANDLING QUALITIES AND AGILITY

If we look more closely at how basic configuration design relates to handling qualities, then the AoA the aircraft is required to achieve to fulfil its objectives has a significant influence.

In setting about designing an 'agile' aircraft, which is what the user says he wants, and in this context one might distinguish the user from the customer, it is important to recognise the relation of handling qualities, aerodynamics and the flight control system (FCS).

For example as shown in figure 7, the configuration layout determines the aerodynamics, in terms of stability, control power and the aerodynamic damping. The degree of non-linearity in all of these can also be determined by the configuration, but the major term at a given Mach number is AoA.

It is then the tasks of the FCS designers, flight mechanicists and aerodynamicists to ensure that the handling qualities and safety requirements are achieved. This may result in some degree of reconfiguration.

Simple criteria (references 3-6) can assist with critical configuration evaluation, based upon the basic aerodynamics. Whilst being useful as guides, they fall short of accuracy too often, as noted in reference 7, and progressively more complex criteria have been developed to try and account for the additional terms, see references 8 and 7 for example.

But what handling qualities should we design to? Various sets of criteria exist which one might start with, for example references 9 to 11, see figure 8. In addition, other criteria can be used, such as those relating to response shaping and flight path control, see references 12-16. These are probably more relevant when high levels of augmentation are required. Then the trick is to provide crisp response with high levels of damping. Underlying it all, though, is a very basic message; the FCS is only as good as the aerodynamics and can only make up to a limited extent for poor design, either in the configuration and its associated aerodynamics or in the FCS itself. The procedures followed at BAe are illustrated in figure 9.

Once the basic handling is set out, it is essential that the large perturbation or gross manoeuvre aspects are examined. The non-linear effects of aerodynamics, control system and actuation systems must be investigated, particularly if the aircraft is to be agile and 'carefree', or 'care-less' in its handling. A distinction of the latter two terms is important. The former implies the pilot can do anything, the latter anything reasonable. The distinction could be worthy of a complete meeting on its own!

In addition it is necessary to ensure that satisfactory safety levels are achieved under all manoeuvring conditions. In simple terms, this means that potential loss of the aircraft must be avoided in all reasonably probable system failure states.

5 AGILITY

Almost inevitably, discussions relating to carefree handling and high AoA tend towards the subject of combat aircraft agility. A dictionary defines agility as 'quickness of motion, nimbleness and readiness'. Obviously, the essential element is time, in particular the time to achieve a given change of state of the aircraft.

A definition of what we mean by agility, as applied to a combat aircraft, is given in figure 10. The part played by handling qualities and conventional manoeuvrability are fairly clear. They relate to changing of the aircraft's velocity vector and the ability to control the changes. Most of the basic current handling qualities requirements identify the sort of behaviour to aim for, but only for comparatively low incidence conditions.

In recent years, a number of criteria have been put forward which are intended to aid the configuration design assessment, see references 2-8 and 20. Whilst all have their uses and may be a reasonable guide for configuration design work, furthur work remains to provide a set of universally accepted criteria. A clear understanding of the limitations of any method will be essential for successful application of the criteria.

The major difficulty lies in deciding what level of performance is required at elevated AoA, away from the conditions close to level flight, particularly about the roll axis. The prime objective, certainly within the United Kingdom, has been to ensure that rolling motions are co-ordinated, i.e. that sideslip is minimised, over the AoA range for which rolling is to be performed. This means that the aircraft is constrained to roll about the velocity vector, the wind axis, by appropriate design of the FCS.

However, it is essential to recognise the consequences of doing this, particularly on inertially slender configurations typical of high performance combat aircraft which exaggerate the inertial coupling effects. Figure 11 shows the sort of roll performance that might be sought if the maximum AoA is limited. This leaves a roll capability at either maximum normal acceleration and at maximum lift. A constant roll performance could be specified, but would have significant implications on both aircraft structure and the size of the pitch control. Figure 12 illustrates the latter aspect.

When rolling at high AoA, a nose up pitching moment is generated via inertial coupling and the pitch control has to be able to overcome this term in order to allow control of AoA in the roll. Failure to achieve this, by inadequate pitch control power or excessive roll rate will lead to a pitch up and departure. The concept of pitch recovery margin is well known and becomes particularly significant on aircraft configurations that are longitudinally unstable. A simple expression can be used to evaluate the pitch acceleration term for a co-ordinated wind axis roll, viz

$$\dot{q} = 1/2\, p^2 \sin 2\alpha$$

where p is the aircraft body axis roll rate and α is aircraft angle of attack.

Along with these criteria for roll rate as functions of AoA and normal acceleration, roll acceleration needs to be addressed. It is important that this be adequately high; good acceleration should imply good control effect rather than low damping in order to ensure the roll can decelerate as well. Naturally, this could size the roll control and/or actuation system. However, too high a value is also to be avoided. The pilot's head is usually situated above the roll axis and high roll acceleration can cause extreme discomfort. For any configuration, the maximum acceptable level of roll acceleration can be set by consideration of the geometry relating the pilot's head position to the roll axis.

6 FUNCTIONAL AGILITY OR FIGHTING QUALITIES

All of the foregoing comments on agility relate to the more conventional flight dynamics aspects. Rapid change of state is clearly important, but what other aspects of a combat aircraft should be considered?

A number of systems in addition to the FCS, play a part in providing the pilot with an agile, capable aircraft. Some have major influence on the choice of configuration. Figure 13 summarises the essential elements.

There is the engine to consider, in terms of response characteristics to throttle movement and airframe compatibility and close integration with the FCS is likely to be required to achieve optimum performance. The intake must provide adequate supply and quality of air and minimise stall or surge likelihood, even whilst the aircraft engages in extreme manoeuvring. This could lead to choice of inlet position and define any variable geometry requirements. Similarly, the nozzle has to be integrated with the afterbody, and this may feature thrust vectoring or reversing controlled by the FCS. Along with all of this, both have major impact on signature.

The avionic system plays a major role. This needs sensors and processing to identify targets, determine priorities and present information simply and clearly to the pilot such that he can concentrate on tactics in the battle. A high level of integration is required to achieve this after all, does the pilot really care which sensor told him of the threat to him? Perhaps he would prefer a system to say 'that is the threat you had not seen, and that bang was me launching a weapon at it'.

Finally there is weapon release. It is of little use to be able to point at the target, or close enough to come within the seekers field of view, if you cannot maintain the view long enough to acquire the target and launch a missile. With a gun it may be possible, but how do you launch a missile at very high AoA?

If all the systems and capabilities match, you have a formidable weapon system.

7 DESIGN FOR GOOD HIGH AoA HANDLING

Several very basic design criteria can be used to choose an aircraft configuration such that it has good high angle of attack characteristics. After all, the FCS can only make good deficiencies in the aerodynamics to a point, and if the design reaches or exceeds that point, the results can be disastrous.

For most combat aircraft, the wing is designed with sustained turn rates, gust response and minimised supersonic drag (where applicable). In addition, stealth is now a major consideration. This leads to compromise of sweep, area and span. But there are limits to be observed which relate to pitch-up, as shown in figure 14 (reference 17). Strictly, the criteria applies to the wing and body only, but it is indicative of possible problems even for tailed aircraft. Obviously for canard or tailless configurations it is significant. It is also clear that camber and twist can modify the situation.

Fin sizing, and indeed fin positioning, are crucial for good high AoA handling. Supersonic designs usually have adequate fin area and yaw control power at high AoA and the sizing criteria might be associated with rolling the aircraft at supersonic speed. For subsonic aircraft, the high AoA regime is likely to be the sizing criteria, which can make the design task more complex.

Control surface sizing is crucial to high AoA handling. Again, sizing for supersonic high speed flight ensures adequate actuation power and physical size may be dictated, for example by providing adequate roll control for crosswind landing. Use of all moving controls, particularly tails is very beneficial. For trailing edge controls, it is necessary to observe maximum angles and chord sizes, as effectiveness reduces with increasing either too much. Figure 15 shows the sort of trailing edge deflection that might result if you wish to trim and control using a trailing edge flap which also stabilises the aircraft in pitch. Maximum angle is dictated approximately by the point at which control power is halved.

Forebody shaping has been the subject of numerous studies in recent years, particularly in relation to high AoA (see references 1, 18, and 19 for example). It is possible to generate shapes which give stabilising directional effects, but at the risk of producing propelling moments in the event of loss of control and subsequent spin. Chined forebodies give benefits by controlling vortex separation points. They offer the possibility of additional control in yaw at high AoA, by varying the separation point in a controlled manner geometrically, or by blowing or sucking to alter vortex strengths. This latter effect can be applied to any nose shaping. Chines are also extremely effective for reducing the nose contribution to signature.

8 THE BALANCE TO BE ACHIEVED IN CONFIGURATION SELECTION.

It is clear that an overall perspective of the total weapon system capability required is essential when deciding the configuration and, in particular, the need to operate the aircraft at or significantly beyond the maximum lift point.

If the perception is that use of the aircraft at AoA significantly above maximum lift is required then it is essential that the aspects set out in figure 16 are addressed. Optimisation of the configuration may be made more difficult as an additional requirement has been added. Studies will be required to determine whether this would lead to penalties in airframe or engine mass, complexity of aerodynamics or FCS design, reduced performance over the rest of the flight envelope or increase the overall design cycle timescale and cost.

By comparison, if a configuration is chosen which utilises maximum lift, albeit with 'carefree' handling, then the engineering compromise may be simpler, as indicated in figure 17. A configuration more optimum for conventional performance may result, and this could certainly benefit such aspects as weapon release. The design cycle time and cost may be less, due to constraining the vehicle to a flight regime which is perhaps more readily predictable.

9 THE QUESTIONS TO BE ANSWERED

The fundamental question to be answered is shown in figure 18 and relates to whether or not AoA should be limited. In the end, the question will reduce to, 'What is it that matters?' Is it turn rate? Is it turn radius? Is it the ability to control the aircraft such that the weapon system can lock on to a target and fire a weapon?

The answer should be the last of these, but that then leads to how do we ensure that capability in the design stage with minimum development? The answer is not really available as yet. It may depend on particular project requirements or upon how the designers choose to interpret such requirements.

It could be considered that if rapid weapon pointing is really what we are seeking, then should we adopt 'alternative technology' and return to aircraft concepts featuring turrets.

The personal view of the authors is that the usefulness of PST remains to be demonstrated, particularly if it forces a compromise in configuration which affects other performance or agility parameters or results in unjustifiable increase in complexity or cost. It is likely to require significant development of weapon system avionics, weapon release aerodynamics and weapon launcher systems in order to accrue the full advantage.

No doubt the X-31 will address some of these questions. The answers are awaited with interest.

10 CONCLUSIONS

Whilst a definitive set of conclusions cannot be provided, indeed it would be presumptuous to do so given the state of the art, a summary of where we stand can be made.

1. Current design techniques can result in highly manoeuvrable aircraft which are limited only by the structural constraints on the airframe and the boundaries beyond which the FCS would no longer control the aircraft.
2. To produce an agile aircraft with exceptional fighting qualities, then it is essential not only to consider the handling, but also the performance of the overall weapon system. This includes the engine, avionics systems, cockpit displays and weapon release systems.
3. The usefulness and cost effectiveness of extreme post-stall manoeuvres remain to be proved and the experiments currently in hand will prove useful in helping designers choose the way ahead. Hopefully, they will tell us whether or not we need to consider developing the weapon release aspects in order to take full advantage of the capability.
4. Until the results of such research are completed, then effort on production aircraft should not be aimed at PST capability, although that could be acceptable fall-out from the basic design.

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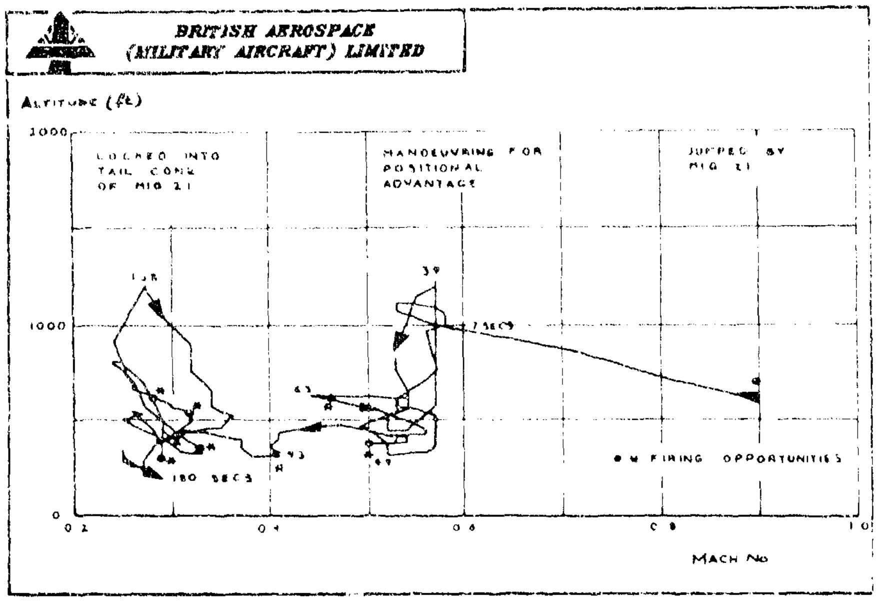
DOG-FIGHT SPEED/HEIGHT PLOT

FIGURE 1



AIRCRAFT TURN RATES

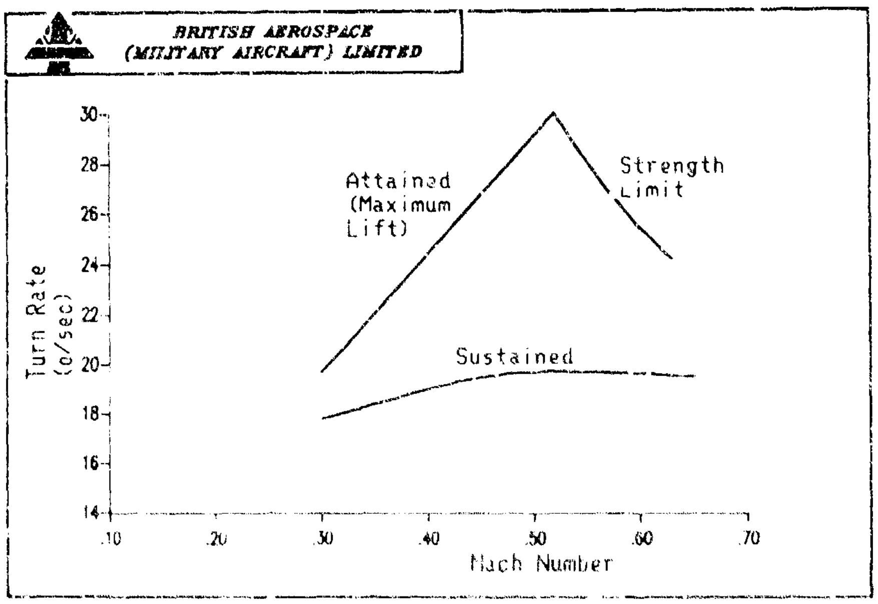
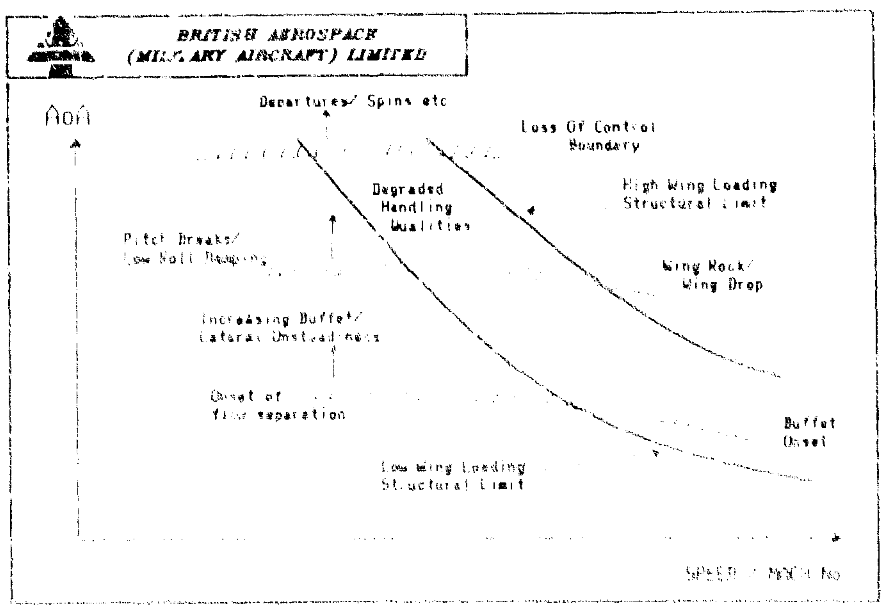


FIGURE 2

MANOEUVRE LIMITING PHENOMENA

FIGURE 3



WHAT DO WE MEAN BY 'HIGH ANGLE OF ATTACK' ?

BRITISH AEROSPACE (MILITARY AIRCRAFT) LIMITED

AIRCRAFT TYPE CATEGORY	MAXIMUM AoA LIMIT	DEFINITION OF HIGH AoA
TRANSPORT/ BOMBER	WING STALL	AoA > WING STALL
TRAINER	> WING STALL	<= SPINNING AoA >= WING STALL
MULTI-ROLE COMBAT	DEPARTURE AoA/ CL MAX	AoA >> WING STALL
MULTI-ROLE COMBAT (PST)	UNLIMITED	AoA >> CL MAX

FIGURE 4

COMBAT AIRCRAFT TECHNOLOGY V TIME

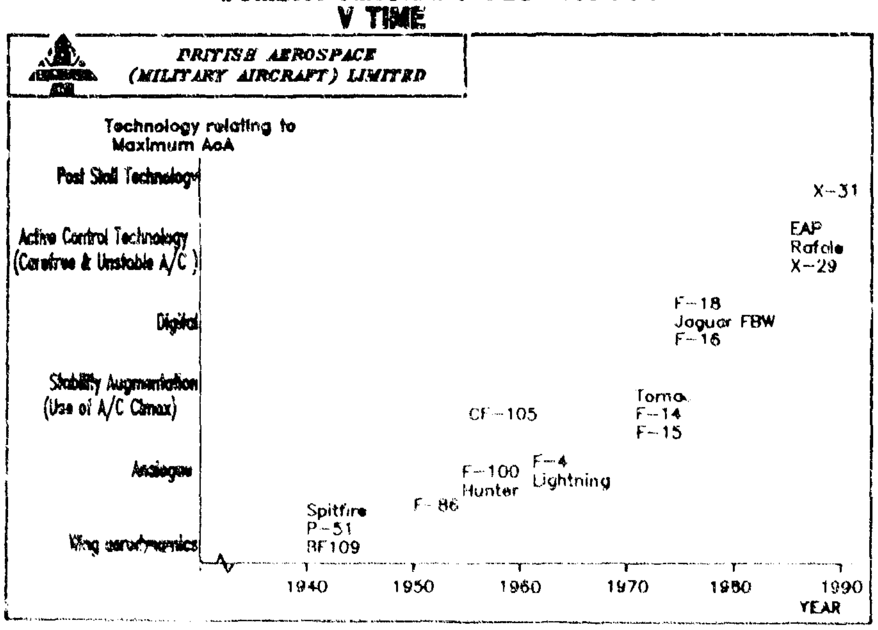


FIGURE 5

FACTORS AFFECTING AIRCRAFT CONFIGURATION

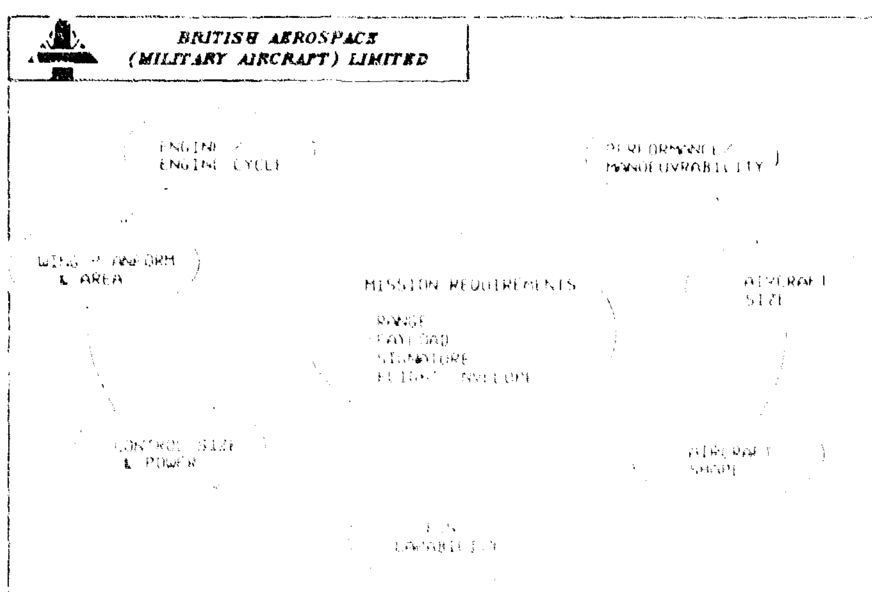


FIGURE 6

CONFIGURATION & RELATION TO HANDLING QUALITIES REQUIREMENTS & AGILITY

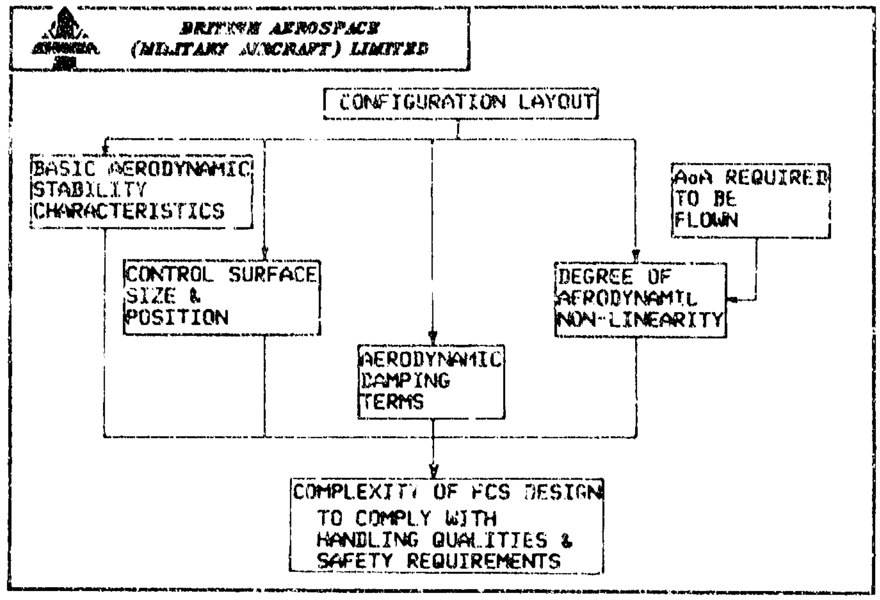


FIGURE 7

STABILITY & CONTROL & HANDLING QUALITIES FOR HIGH AoA DESIGN

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

STABILITY & CONTROL CRITERIA

PITCH RECOVERY MARGIN
- POSITIVE RECOVERY MARGIN REQUIRED IN ALL MANOEUVRES

$Cn\beta$ DYNAMIC
- POSITIVE VALUES INDICATE DUTCH ROLL MODE REMAINS OSCILLATORY

LATERAL CONTROL DEPARTURE PARAMETER
- POSITIVE VALUES INDICATE NO ADVERSE YAW DUE TO ROLL CONTROL

FIGURE 8

STABILITY & CONTROL & HANDLING QUALITIES FOR HIGH AoA DESIGN

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

HANDLING QUALITIES METHODS

CHARACTERISTIC EQUATION ROOT ANALYSIS
-EXAMINE MIGRATION OF AIRCRAFT & FCS ROOTS WITH INCREASING AoA & ENSURE MODES ARE STABLE /DAMPED OR NOT DIVERGENT

SYSTEMS ROBUSTNESS
-EXAMINE STABILITY MARGINS & SENSITIVITY TO TOLERANCES, ENSURE ADEQUATE GAIN / PHASE MARGINS ARE MAINTAINED

LARGE PERTURBATION MANOEUVRES
-6 DoF ANALYSIS TO CONFIRM HANDLING PRECISION & FREEDOM FROM SATURATION

FLIGHT SIMULATION
-PILOT IN THE LOOP ASSESSMENT

FAILURE ASSESSMENT
-ENSURE LOSS OF CONTROL IS AVOIDED AND THE AIRCRAFT IS RECOVERABLE IN ALL REASONABLY PROBABLE FAILURE STATES

FIGURE 9

DEFINITION OF AGILITY

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

THE ABILITY TO CHANGE AND CONTROL AN AIRCRAFTS VELOCITY VECTOR, IN MAGNITUDE AND DIRECTION, SUCH THAT A WEAPON FIRING SOLUTION IS ACHIEVED IN MINIMUM TIME

AGILITY CRITERIA

: TURN RATE
: TIME TO PITCH & STOP
: TIME TO BANK & STOP
: TORSIONAL AGILITY
: SPECIFIC EXCESS POWER
: POWER RATE

: WHAT ELSE ?

FIGURE 10

SCHEMATIC OF TYPICAL ROLL PERFORMANCE REQUIRED OF AN AGILE AIRCRAFT AS A FUNCTION OF AoA & NORMAL ACCELERATION

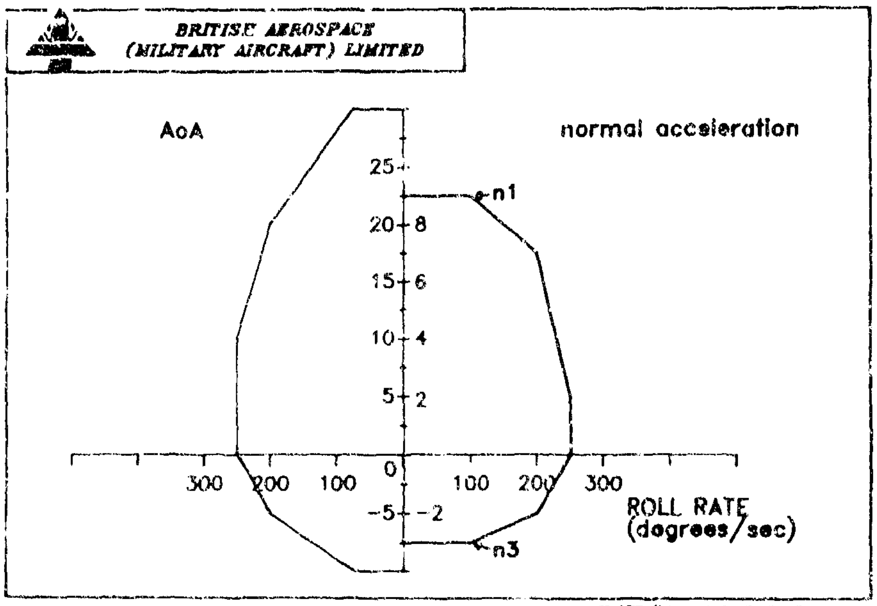


FIGURE 11

PITCH RECOVERY MARGIN FOR INERTIALLY COUPLED MANOEUVRES

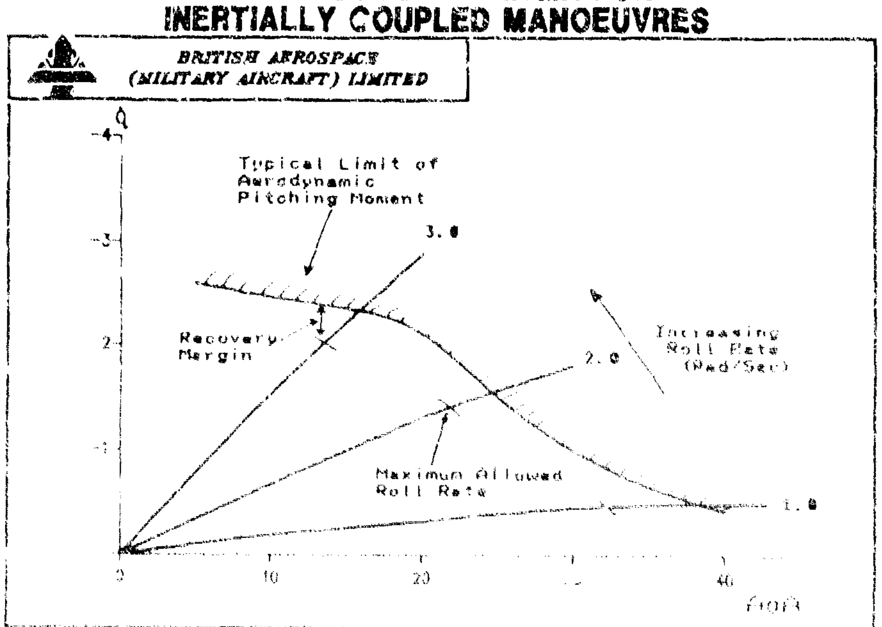


FIGURE 12

AGILITY - OTHER FACTORS INVOLVED

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

ENGINE RESPONSE CHARACTERISTICS
- INLET DESIGN
- NOZZLE/AFTERBODY INTEGRATION
- FCS/ ENGINE INTEGRATION

AVIONIC SYSTEM
- SENSOR PERFORMANCE
- TARGET DISCRIMINATION
- TARGET AQUISITION
- INTEGRATED SYSTEM

WEAPON RELEASE
- LAUNCH AT ANY FLIGHT CONDITION, AS DEMANDED

COCKPIT DISPLAYS
- INFORMATION TO ENABLE PILOT TO APPRECIATE & EVALUATE HIS TACTICAL SCENARIO

FIGURE 13

PITCH - UP BOUNDARIES

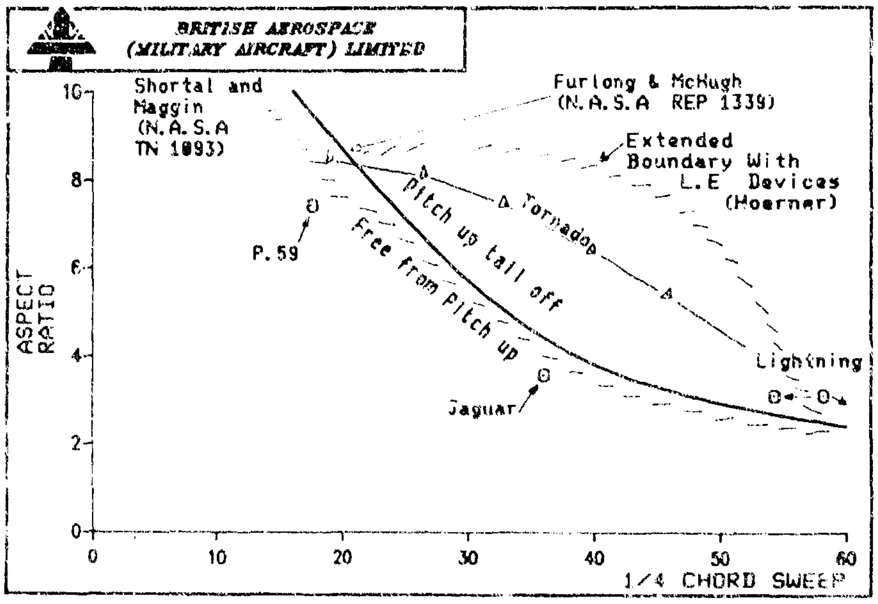


FIGURE 14

MAXIMUM ALLOWABLE FLAP ANGLE ALLOWING FOR FCS CONTROL AND STABILITY REQUIREMENTS

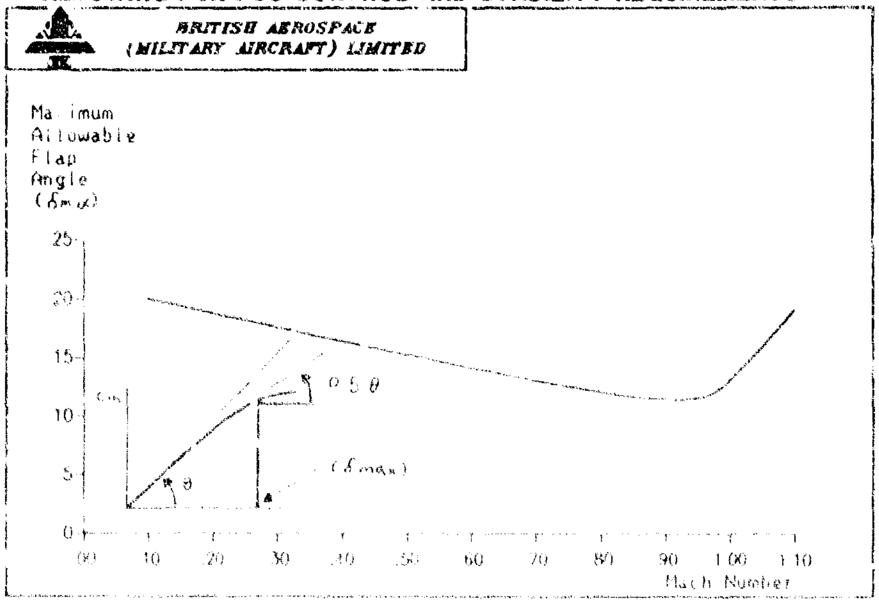


FIGURE 15

USE OF AoA CLOSE TO $C_{L\,max}$

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

- : FREEDOM TO CHOSE CONFIGURATION FOR OTHER AREAS OF FLIGHT ENVELOPE
- . WEAPONS RELEASE LESS PROBLEMATIC
- : SIMPLIFIED CONTROL REQUIREMENTS COMBINED WITH CAREFREE HANDLING
- : BETTER ENERGY MANAGEMENT FOR COMBAT
- : POTENTIAL FOR REDUCED DESIGN CYCLE
- : AVOIDS PENALTIES OF MASS & COST ASSOCIATED WITH THRUST VECTORING

FIGURE 16

USE OF AoA SIGNIFICANTLY BEYOND $C_{L\,Max}$

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

- : CAN WEAPON SYSTEM PERFORM ?
- : CAN A WEAPON BE RELEASED AT HIGH AoA OR IS IT A GUN ATTACK ONLY ?
- : IMPLICATIONS ON AIRCRAFT CONTROL & RECOVERY ?
- : IMPLICATIONS FOR INTAKE DESIGN ?
- : REQUIREMENT FOR THRUST VECTOR / REACTION CONTROL ?
- : EFFECT ON AIRCRAFT ENERGY STATE IN COMBAT ?
- : SAFETY IMPLICATIONS?
- . ARE THERE PENALTIES ON MASS, COMPLEXITY, AERODYNAMICS WHICH COMPROMISE THE PERFORMANCE OVER THE REST OF THE FLIGHT ENVELOPE ?
- . WHAT IS THE COST EFFECTIVENESS ?

FIGURE 17

QUESTIONS TO BE ANSWERED

BRITISH AEROSPACE
(MILITARY AIRCRAFT) LIMITED

- . TO LIMIT OR NOT TO LIMIT ?
- . WHAT PARAMETERS MATTER FOR AGILITY ?
  - TURN RATE ?
  - TURN RADIUS ?
  - ROLL RATE AND / OR ACCELERATION ?
  - ENGINE RESPONSE ?
- . IS IT CONTROL OF THE VELOCITY VECTOR SUCH THAT THE WEAPON SYSTEM CAN AQUIRE AND LAUNCH WEAPONS ?
- . HOW DOES AVIONIC SYSTEM PERFORMANCE INFLUENCE AGILITY ?
- . WHAT ABOUT WEAPON RELEASE ASPECTS ?

FIGURE 18

X-31 A AT FIRST FLIGHT

W. B. Herbst
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- Unternehmensbereich Flugzeuge -
Munich, W-Germany

Abstract:

The X-31A accomplished its first flight on October 11, 1990. Thus, it is appropriate to present a summary about the objecties of this international experimental flight test development program, its status and follow-on planning.

1. Summary

The X-31A is an experimental aircraft dedicated to explore the controlled flight beyond stall and enhanced agility (supermaneuverability). It it the first aircraft using thrust vector control in pitch and yaw and it is also the first experimental aircraft being developed and tested internationally. The X-31A has the potential of providing the means of superior short range air combat capability without sacrifice to supersonic performance and thus also superior beyond visual range combat effectiveness. It will be the supersonic aircraft with the lowest minimum speed, it will be superior to any existing fighter aircraft in terms of the ability to make tight and quick turns and any measure of agility.

The X-31A will be most interesting for pilots to fly:
It will not depart and spin but will be fully maneuverable at and beyond stall conditions. It will be controlled with the stick only, without noticeable sideslip even at very high angles of attack up to 70° AOA. Rudder pedals would be obsolete except for intended sideslips and cross wind landings. In high performance post-stall (PST) maneuvers very peculiar attitudes and angular motions will be encountered, however, a new flight display will keep the pilot from getting disoriented and help him to maintain flight path control. For an opponent the maneuver of a fighter with X-31A capabilities will be hard to predict due to its attitude during PST-maneuvers and the quickness to roll and pitch into an unexpected new maneuver condition even at flight conditions critical to a conventional aircraft. Thrust vector enhanced sideslip maneuvers would allow head-on gun attacks to very short closure at safe collision distances and would provide more and longer shooting opportunities during close-in air combat.

The Enhanced Fighter Maneuverability (EFM) hat the potential of improving the overall effectiveness in close air combat by a factor of more than two, based on extensive manned and computer simulations.

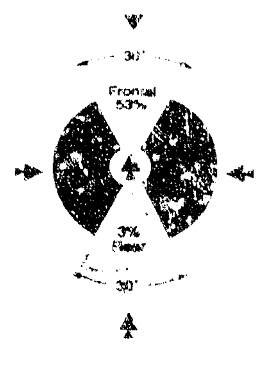


Fig. 1-1 Weapon launch aspect in Future Air Combat Statistical result of Combat Simulations

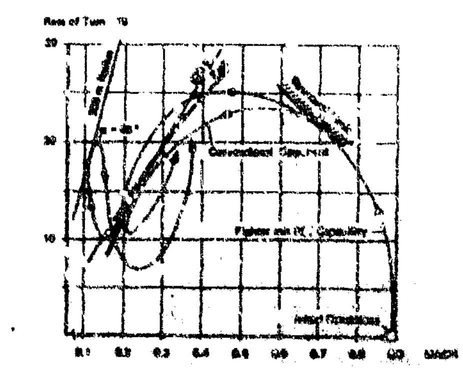


Fig 1-2 Maneuver Cycle in Future Short Range Air Combat as Experienced in Combat Simulation

2 The Concept

The concept of supermaneuverability was originated around 1978 in response to the developing all aspect capability of short range missiles. The ability to successfully launch a missile in almost any clockwise position against an opponent seemed likely to alter the tactics of air combat and thus the performance requirements of fighter aircraft. It was found in extensive manned and computerized air combat simulations that appropriate tactics actually would result in mutual head on launch opportunities (fig.

1-1) and thus in the dilemma of potential mutual kills amongst equal high performance aircraft. The analysis of such engagements revealed a new maneuver cycle (fig. 1-2) characterized by dominance of instantaneous maneuvers and a tendency to slow speed (ref.(1)). At slower speed an aircraft would achieve a smaller radius of turn at a given rate of turn and, obviously, a tighter turn in a developing mutual head-on situation would allow for an earlier weapon launch at any given off boresight angle (fig.1-3).

Conventional aircraft, however, are limited in controllability at slow speed and may even get uncontrolled at stall speed just as they are achieving the smallest radius-of-turn. Any significant reduction of radius-of-turn could only be achieved by deeply penetrating the post stall regime. With thrust vector control and a proper aerodynamic design it was anticipated that an aircraft could be maneuvered safely at and beyond stall limits. Very soon it was found that the biggest design challenge was to roll the aircraft at high angles of attack around its velocity vector quickly enough to achieve the desired tight turn performance (ref.(2))

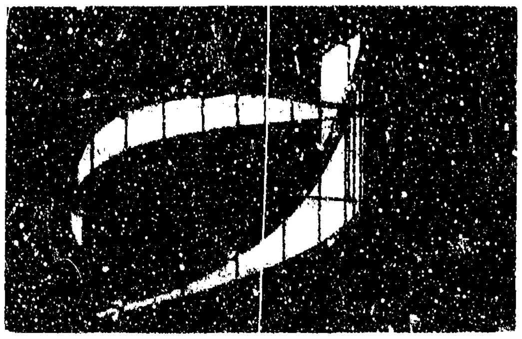
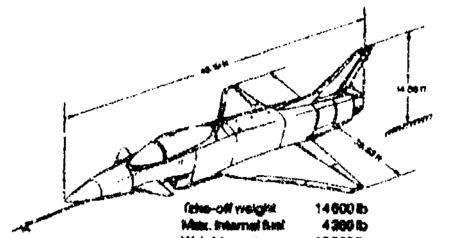


Fig. 1-3 Importance of Turn Radius in Future Air Combat



Take-off weight	14600 lb
Max. internal fuel	4300 lb
Weight empty	10900 lb
Structural weight	5300 lb
Wing ref. area	226 ft²
Canard ref. area	23.5 ft²
Vert. tail ref. area	37.6 ft²
Wetted area total	1008 ft²
Limit load factor	+9 g/−4 g

Fig. 1-4 The X-31 A

Generic fighter aircraft with post-stall capability were evaluated first in manned combat simulations at the German IABG in 1977 (ref.(3)) and at Mc Donnell's facilities in St. Louis in 1978 (ref. (4)). By that time about 1? operational pilots of USAF and GAF have had the opportunity of familiarizing themselves with this new capability and to generate statistical data about its effectiveness. Many technical features now being installed in the X-31A have been empirically developed during those simulations, for example

- mechanization of lateral stick input to roll the aircraft around the flight path at zero sideslip angle rather than around the familiar aircraft body axis
- angle of attack and nz demand with proper blend-over
- PST entry mechanization of the flight control system
- gravity and gyroscopic moment compensation
- consideration of inertia coupling
- scheduling of control surfaces and thrust vectoring blend-in
- response characteristics and maximum deflection of the thrust vectoring system in pitch and yaw and the criteria for body axis roll

All pilots had to go through a learning process of how to use the new capability in order to achieve a tactical advantage and many of the maneuver characteristics now being defined as "PST-performance" in X-31A have been developed during manned simulation. This includes the effect of weapons and fire control. As an overall result, it was found

- that combat effectiveness in one vs. one engagements can be expected to be improved by a factor of at least two
- that supermaneuverability would provide a fair chance to survive against two opponents of similar conventional performance
- that the benefit of supermaneuverability tends to get bigger in multi boogies in outnumbered situations

Results of all operations analysis work performed prior to the X-31A program are summarized in the first X-31A report (ref.(5)). It was the basis of the acceptance of the program.

The X-31A Configuration

It was requested that results of X-31A flight testing should be directly transferable to a potential aircraft. Therefore several existing US-fighter aircraft were investigated during phase I of the program (ref.(6)) as potential candidates for modifications and implementation of supermaneuverability. Unfortunately, none of those candidates appeared to be sufficiently suitable or advisable in view of anticipated cost of modification and/or penalty for conventional performance.

Supermaneuverability needs:

- a thrust-to-weight ratio of at least 1.0
- an electronic flight control system
- an intake configuration to allow full power engine operation at up to 70° angle of attack
- a low wing loading and high leading edge sweep
- certain aerodynamic characteristics to allow smooth transition into the post-stall regime
- a horizontal control surface that moves into the wind at increasing angle of attack
- a configuration layout which is preferably unstable in pitch at subsonic speeds for better supersonic performance
- resistance to enter a spin and an easy recovery from a spin once entered, needed to avoid the thrust vectoring system to become a safety critical item.

Eventually it was decided to build a dedicated new aircraft. (fig.1-4). A derivation of the German "TKF", a predecessor of the European Fighter Aircraft (EFA), was selected because it was designed to meet the above requirements and much of the existing engineering data and experience could be utilized.

The X-31A is a delta-wing configuration with a "long coupled" canard. It distinguishes itself from all other currently existing "short coupled" delta-canard fighter aircraft - except for the British/Italian EAP which, historically, is also a derivation of the original German TKF - by the position of the canard relative to the wing. On the X-31A the canard is to be used for pitch control and trim rather than as a high lift device. In order to reduce trim drag in supersonic flight the X-31A is designed with the center of gravity aft of the subsonic center of lift which makes it aerodynamically unstable. As a result, however, the canard moves downward into the wind at increasing angle of attack and thus always maintains its control effectiveness.

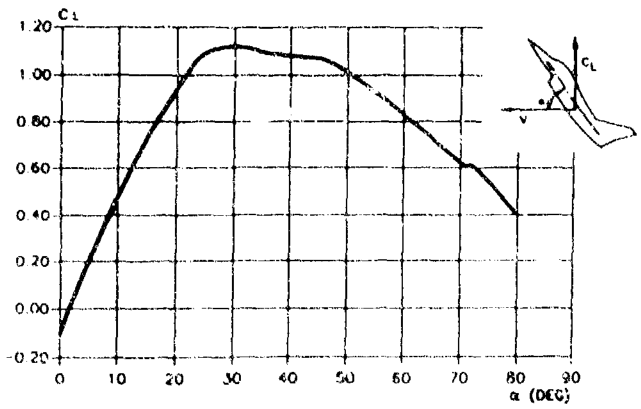


Fig. 3-1 X-31A lift in the conventional and PST regime

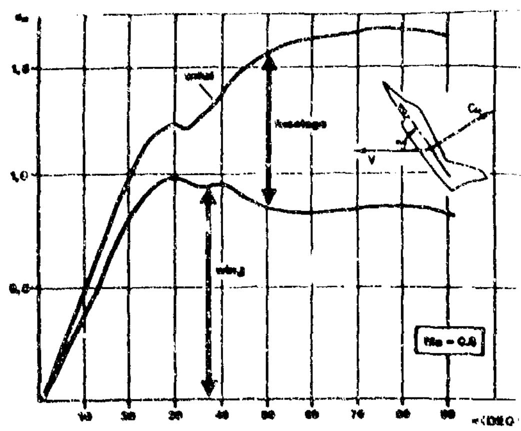


Fig 3-2 Split of normal force on wing and fuselage

The delta wing with its high leading edge sweep and large area in combination with the unstable layout lends itself particularly well for superior supersonic and subsonic performance. It comprises,, however, a fairly complex design because it also has to satisfy the requirements for transonic flight and for the new post-stall regime and to minimize the penalties of the canard interference.

The belly position of the intake in combination with the movable lower lip allows the air to enter the intake at post-stall conditions with minimum disturbance, a prerequisite for the GE-F 404 engine to operate with full power even at extreme angle of attack. In supersonic flight the lip will be moved into an upward position in order toreduce spill drag.

The most unusual device and a necessity for PST-maneuverings is the thrust vectoring device at the aft fuselage of the X-31A. The three devices of the X-31A allow to deflect the engine exhaust at full power condition conically up to about 10°, which produces about 17 % of engine thrust in any lateral direction. Thrust vectoring is used for control only, the "paddles" will be immersed into the hot exhaust for the very short time periods. In the conventional flight regime they can be used as speed brakes in addition to the conventional devices.

On an operational aircraft the "paddles" would be replaced by an engine integrated thrust vectoring system resulting in a cleaner rear fuselage. Unfortunately, such vectoring nozzle was not yet available for the X-31A and could not be developed within the time and funding limitations of this program.

Much of the external appearance of the X-31A was dictated by the use of existing components (such as the F-16 landing gear and the F-18 canopy) and the aim for low cost (such as carrying all fuel in the fuselage and keeping the wing dry).

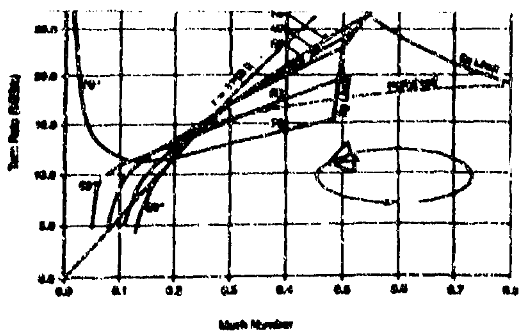


Fig. 3-3 PST performance in level flight

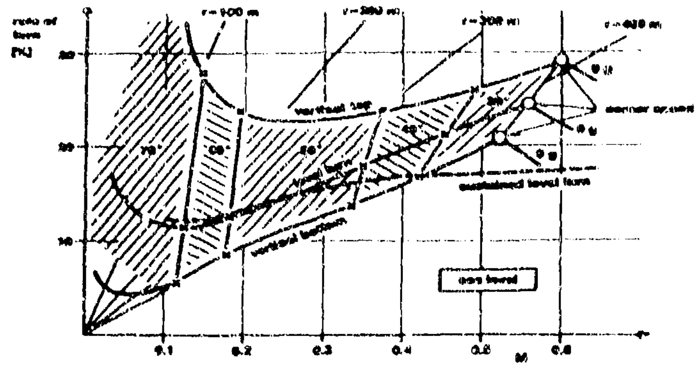


Fig. 3-4 Summary of PST maneuver performance

3. X-31A Performance

The X-31A is designed to maneuver beyond maximum lift angle-of-attack up to about 80° AOA. Maximum lift is achieved at about 30° AOA, beyond that point lift is decreasing with increasing AOA (fig. 3-1). Aerodynamic load on the aircraft in terms of normal force is carried by wing and fuselage. In the PST-regime a major portion of total load is acting on the fuselage (fig. 3-2). Total normal force is increasing steadily up to 90° AOA, comparable with the drag of a flat plate.

A simple mass point analysis leads to fig. 3-3. It shows aircraft rate-of-turn for a level maneuver for various AOA as function of M-number. There is a less of turn-rate and only little improvement of turn-radius as AOA is increased only marginally above maximum lift AOA. It needs a deep penetration into the PST regime until substantial benefits can be realized. Also, there seem to be little overall advantage of PST maneuvering in level flight.
Fig. 3-4 shows a similar analysis covering a varity of maneuver states. Maximum performance is achieved with maximum help of earth gravity. The shaded areas indicate maneuver states with optimum angle-of-attack usage. Also indicated are best radius-of-turn performance for inverted vertical maneuvers.

Optimum PST-maneuvers have been analyzed in Ref.7 , although, disregarding control power limitations and actual 6 DOF aircraft dynamics. Fig. 3-5 is showing a 180° heading reversal as performed in manned simulation using realistic X-31A aero data and flight control characteristics. Circles indicate the entry into and recovery from the PST flight regime. In this case a very tight and quick turn was achieved with maximum instantaneous performance of 31° rate-of-turn and about 70 m radius of turn. The same maneuver typ with maximum lift AOA limitation is shown in fig. 3-6. Best rate-of-turn and radius of turn now is limited to 25° and about 400 m respectively.

Key for maximum instantaneous maneuver performance is velocity vector roll rate/acceleration to initiate a turn and stop the turn maneuver. As a rule of thumb velocity vector roll rate needs to match the desired turn rate. This is why none of the existing aircraft could perform high performance turn maneuvers at high AOA even though they might be able to pitch up to very high AOA. There is no PST-maneuvering capability in any tactical sense without sufficient body axis yaw power at high AOA and automatic avoidance of excessive side-slipping. Such capability can only be provided by multi axis thrust vectoring in combination with an appropriately designed flight control system. Fig. 3-7 is showing demonstrated X-31A velocity vector roll performance at 60° AOA.

Fig. 3-7 is showing a typical tactical PST maneuver as seen in many manned air combat simulations. In this particular case the aircraft is outmaneuvering a first opponent by menas of a very quick and tight PST maneuver and subsequently chasing a second opponent. Conventional X-31A performance (fig. 3-9) is not significantly different from contemporary fighter aircraft. X-31A flight envelope is depicted in fig. 3-10 up to M=0.9 and 40.000 ft. The PST envelope is largely overlapping with the conventional subsonic flight regime.

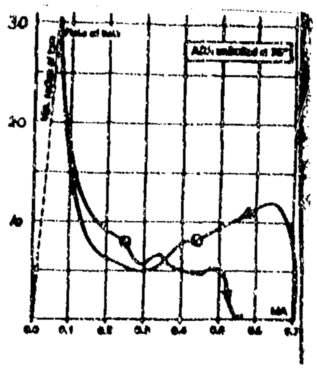


Fig.3-5 Maneuvering for best PST-performance

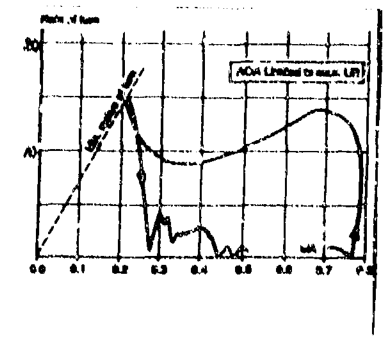


Fig.3-6 Maneuvering for best-turn performance with maximum lift limitation

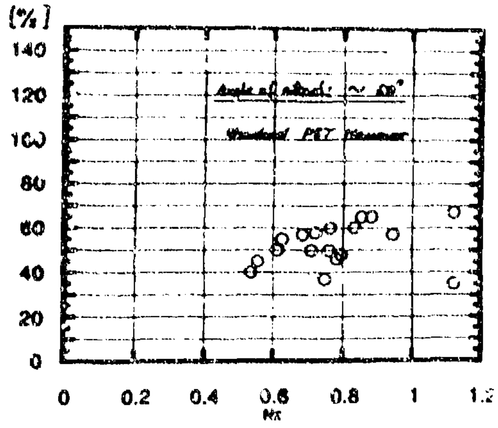
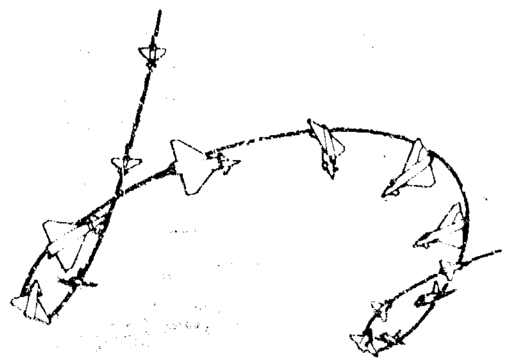


Fig. 3-7 X-31A Velocity vector roll performance

Multi axis thrust vectoring - as being a necessity for PST maneuvering - can also be used to enhance agility in the conventional flight regime. Fig. 3-11 shows a maneuver by which agility will be measured in the X-31A program. It consists of a high performance level wind-up followed by a maximum performance heading reversal.

Flight path decoupling (RCFAM) is a special mode of the flight control system by which the pilot could point the aircraft nose (or gun) independently from its trajectory. Upon a stick input by the pilot the aircraft in RCFAM mode would react adversely in pitch and yaw. Of course, the angular range of such decoupling is aerodynamically and structurally limited, however, these limits could be enlarged by thrust vectoring and thus would be expected to be larger in comparison to conventional aircraft.

Longitudinal deceleration is important for any aircraft to quickly reduce speed for best turn performance and to effectively enter the post-stall regime. For this purpose the thrust deflection devices can be deployed outwards to act as drag devices in addition to the conventional speed brakes.



Typical Post-Stall Manouver Fig. 3-8

Thrust / t.o.weight (s. l. st. uninst. / t.o. weight)		1.1
Wing loading		85 lb/ft²
Fuel / t.o. weight		23 %
Empty weight / t.o. weight		75 %
Structure / t.o. weight		36 %
Conv.	Max. SEP	740 ft/s
	Max. sust. turn rate	17°/s
	Max. inst. turn rate / turn radius	28°/s, 1100 ft
	Approach speed	134 Kts
	Max. speed (potential), 36 K	1.6
	Max. deceleration at M = 0.6 / 10 K	0.8 g
EFM	Max. inst. turn rate / turn radius	looking forward to flight testing
	Agility	
	Fuselage aiming	
	Max. deceleration [illegible]	

X 31A Performance (s.l.) Fig. 3-9

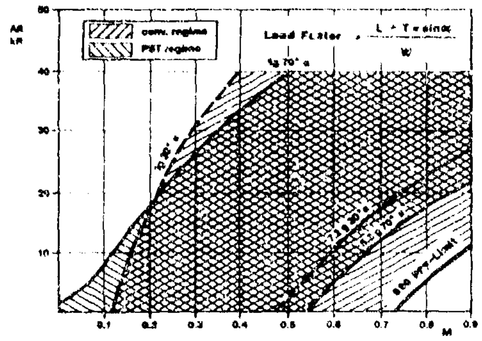
4. Building the plane

After MBB's withdrawal from the Agile Combat Aircraft (ACA) program (which later on became the Experimental Aircraft Program EAP and Rockwell's disappointment of not being selected to build the X 29A both companies met in October 1982 to jointly pursue the concept of supermaneuverability.

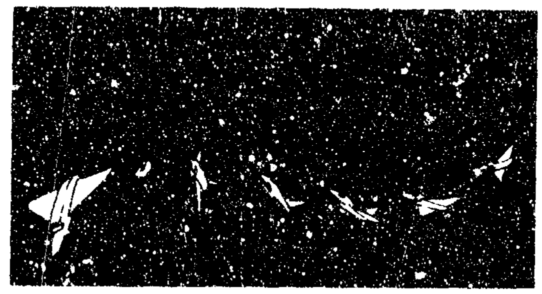
Rockwell together with MBB eventually convinced DARPA in 1984 of the potential of the supermaneuverability concept and its peculiar capability with DARPA's "high risk - high pay off" policy. MBB jointly with Rockwell managed to get support from the GMOD for a coordinated joint study which was named Phase I of the X-31A program.

As a happy coincidence the Nunn-Quayle initiative supporting cooperative international armament programs was promoted at that time and helped the program to materialize. Eventually a government contract was signed in May 1986. The program was called "EFM", Enhanced Fighter Maneuverability. A "phase II" contract was awarded to MBB by GMOD and to Rockwell by DARPA/Navy in September 1986. The objective of phase II effort was to perform the preliminary design of a research aircraft and the necessary wind tunnel testing to allow detailed design and fabrication in the subsequent phase. Also, a formalized mode of Rockwell-MBB cooperation had to be found and a share of work respectively. Basically, there was

- a split into Rockwell workpackages to be performed at Rockwell with MBB local participation and MBB workpackages to be performed at MBB with Rockwell local participation (fig. 4-2)
- a rule that Rockwell engineers working in Munich on MBB workpackages should report to MBB management and vice versa
- no duplication of effort
- no transfer of money between RI and MBB except for special subcontracts



X-31A Flight Envelope Fig. 3-10

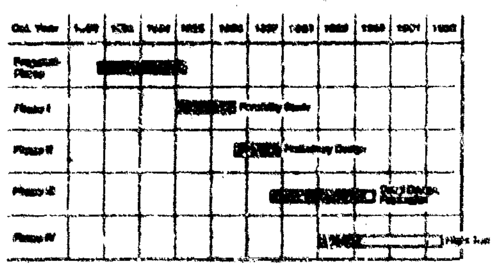


Typical Agility Maneuver Fig. 3-11

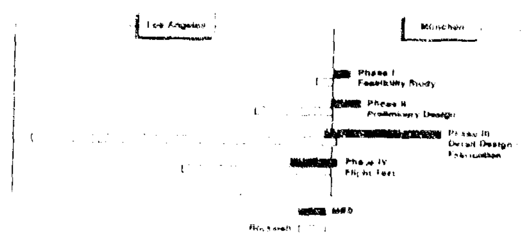
It was decided in phase II that MBB would design and manufacture the two sets of wings and the thrust deflectors and would develop the control laws for the flight control system. Also, X-31A performance would be defined and analysed at MBB. Later on, it was suggested to develop a special display to avoid pilot disorientation in PST maneuvers. The development of this display was also charged to MBB (fig. 4-3)

Maximum use of existing equipment was part of the approach selected very early in the program. This was in particular true in the areas of landing gear (F-16), propulsion system (GEF-404 engine), canopy and windshield (F-18) and a multitude of subsystem components pulled together from fighter aircraft such as the F-16,F-18 and F-20.

A special challenge for Rockwell was the acquition of the extensive list of GFE and CFE equipment. Special help, dedication and persistance was required on all sides to get the parts in time to support the rollout and the first flight of the X-31A. Installation of system components, hydraulic tubing and electrical wiring harnesses were as usual affected by late deliveries of components in many areas and required special attention and treatment.



Program Phases (FMOD Contracts to MBB) Fig. 4-1



Engineering Manpower Distribution and Location Fig. 4-2

The fuselage with the canards and the vertical tail was built by Rockwell, while the wings and the thrust vectoring vanes were developed and manufactured by MBB. A number of trade studies were performed to select the optimum design and manufacturing concepts with consideration given to weight impact, material aspects and manufacturing cost. The fuselage tooling was simplified by using the major NC bulkheads as templates/tools for the installation of longerons, fittings and finally fuselage skins.

Aluminium, aluminium/lithium, steel and titanium were used in appropriate areas utilizing the available NC capabilities existing at Rockwell's facilities. The carbon fibre system used extensively on the B-1 was utilized for honeycomb skin pannels on the fuselage, the rudder surfaces as well as on the skins for the vertical tail. Rockwell Tulsa was responsible for the manufacturing of these carbon fibre parts.

MBB selected the carbon fibre material, also used in EFA, for the manufacturing of the wing box skins as well as the leading and trailing edge flaps. The wing box has a metal substructure with NC milled and metal sheets spars and ribs. The design and manufacturing were simplified by the fact that the wing does not carry any international fuel.

2D and 3D design tools such as CADAM and CATIA were extensively used during the development phase. Exchange of technical data (e.g. loft line definitions for structural parts) between Rockwell and MBB were performed via a commercial communications network which allowed data exchange overnight.

While wing and fuselage were relative conventional in terms of the structural concept, the development of the thrust vectoring vanes required additional considerations, since these parts would be immersed into the hot jetstream of the GEF-404 engine at all power settings, including afterburner. To determine and verify design loads a full scale test was performed behind a Navy F-18 aircraft, utilizing metal structure for the thrust vectoring vanes. However, metallic vanes would result in a relatively heavy weight structure at the very end of the aircraft, thereby moving the C.G. aft, which was not desirable. Therefore, the German Company SIGRI was given the subcontract to manufacture the thrust vectoring vanes from carbon/carbon material. This material is extensively used in space vehicles for thermal insulation and for hot structures.

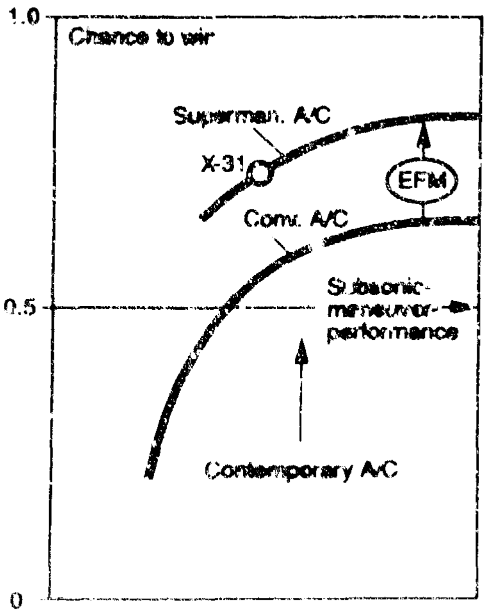


Fig. 5.1

To avoid building a third airframe for structural test to destruction the proof load concept was agreed to by the Navy, DARPA and the German Certification Agency ML. All major components were tested to at least 110% of design limit load. The German ML is responsible for the certification of the wing and thrust vectoring vane structure, whilst the US Navy has the authority to provide the flight clearance for the overall aircraft structure, including all systems.

Another challenge was and still is the development, integration and testing of the Flight Control System (FCS). Initially the FCS was planned to be a sole MBB package, however, in the concept definition phase it became apparent, that manpower and equipment resources available at Rockwell and at MBB as well as requirements for simulation and test made it strongly advisable to change this initial plan. Rockwell took over responsibility for the definition and design of the overall system architecture including redundancy management. The control law development, handling qualities and stability analysis and initial simulation were conducted at MBB's Ottobrunn facility.

Honeywell is the subcontractor of Rockwell providing the computer soft- and hardware including the control law and redundancy management coding and verification.

5. Flight testing

At this time (Dec. 1990) aircraft I has made 9 flights by 3 pilots and a substantial part of the conventional envelope has been covered. All pilots are very satisfied with handling and response characteristics. Very soon aircraft 2 will follow and it is anticipated that the PST envelope will be opened subsequent to about 40 hours of initial flight time.

Tactical flight evaluation, at this time, is still in a stage of definition. The aircraft is not equipped with any tactical sensors, fire control system, weapon representation and suitable cockpit facilities. However, once all the effort and money is spent on developing/building two aircraft, clearing them for the new envelope, developing all the maneuver schemes for its various capabilities and measuring its flight performance, there is a strong desire to substantiate the expected tactical utility by actual flight experience.
The expected tactical benefit is shown in fig. 5-1 as result of many computer and manned air combat simulations.

Ref. 1 W. B. Herbst: Dynamics of Air Combat
Journal of Aircraft, Vol. 20, No. 7, International issue 1983

Ref. 2 Phase I conf. dev. report X-31A Performance Data
MBB X-31A-M-I and Vol. 12 SDM Performance

Ref. 3 H. Ross: Taktische Auswirkungen von PST-Manövern im Luftkampf
MBB Report UF 1498. 1977 und R. Haiplik: TKF/STY/0071 und
R. Polis, W. Frenzl: Untersuchung des Einflusses von
Supermanövrierbarkeit im I vs. I Luftkampf
TN BT 13-88-22/79. 1979

Ref. 4 AFTI, Techn. Report AFFDL-TR-75-86 1975

Ref. 5 R. Haiplik: SNAKE OR Analysis
MBB/LKE 152/STY 001, 1985

Ref. 6 SNAKE phase I report

Ref. 7 K. Well: Optimale dreidimensionale Steuerungen von
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DFVLR Report A 552-78/2, 1978

ROUND TABLE DISCUSSION

chaired by

Mr H.Wuennenberg

Mr Wuennenberg

Ladies and Gentlemen, we come now to the last session of our Symposium which is entitled "Round Table". You will miss a round table here; there isn't one, but hopefully you have sufficient imagination to think there is somewhere a round table within the audience. The procedure which we discussed to operate our round table is the following:

From the individual sessions of our Symposium we asked the Session Chairman and the authors, and also some specialists, to come up with a general view of the session — not only of the session but on the session subject — so this group formed or should have formed, a general opinion on the subject. The intention was to discuss whether we have covered within the conference the main aspects, only a few aspects, or what is the situation, the technical situation; what is left open for further activities. We will now ask the spokesman of the individual groups who have prepared these comments to give their presentations and then leave the floor open for discussion on these opinions. You are all invited to give your own comments on what these gentlemen will present to you.

Our first session of papers was the Pilot's View, and Professor Mulder will give the opinion of this group.

Prof. Mulder, The Netherlands

Thank you Mr Chairman. We, Bob Russell and I, were Chairmen of Session 2, the Pilot's View and as you recall we had three papers. They were presented by Mr Thomas, Mr Morse and Mr LeTron. After our session, we came together, with these authors and some members of the audience, and we had a discussion and finally came up with several different issues together with equivalent recommendations.

The first issue is on the communication aspect of designing a new aircraft. The communication between engineers, pilots, and designers is something that needs to be thought about. They should form small groups, not more than 4 to 5, from aircraft concept, through design, development, test, etc. to insure a successful civil or military platform in regards to flying and handling qualities.

The recommendations are, first of all, that engineers and designers need a reasonable flight experience, to allow communication obviously, and pilots need reasonable engineering skills.

The next part of the recommendation is to involve designers in flight testing, and the last part of the recommendation is to train engineers and pilots as a test team, not as individuals. This is done, for instance, during education for test pilots at Istres. There this approach is really explicitly followed, team experience rather than individual test pilot. This will provide better communication between specialists within the team. The second issue addresses the topic of civil vs military handling qualities criteria. Civil aircraft designers are in our view not exploiting advances in military handling qualities criteria and some civil aircraft manufacturers are not addressing deeply enough the aspect or subject of flying qualities. And next, civil aircraft are built and tested more and more against product liability or legal public liability, and flying qualities requirements are becoming more or less secondary.

Our recommendation is to invite the civil aircraft community to participate in flying qualities working groups, for instance, in the context of the joint airworthiness agency's work.

The next issue is slightly different in scope and it relates to the control of highly automated transport aircraft. In response to a question after Mr LeTron's paper on the Airbus 320 fly by wire lateral control system, the more general problem was raised of keeping the pilot in the loop of highly automated civil and military transport aircraft. Pilots' control inputs no longer have a direct relation to control surface deflections and control input may be made without corresponding movement from the main manipulator, for instance the thrust levers would remain stationary while thrust variations are being applied. In slightly broader context, the pilot of such a highly automated aircraft must use controls which are less natural when compared to classical controls such as control column and thrust lever. There have been complaints that mental workload may, in some instances, be higher than in conventional aircraft and the recommendation would be that flying qualities criteria should be made to reflect operation of these highly automated aircraft. Basic research might still be necessary which would critically review present cockpit layout, that means controls and displays, control laws and procedures.

And our last issue, is rather controversial I think. It relates again to handling qualities criteria, and it says that handling qualities requirements should state fixed numbers only when vital for the safe conduct of flight. The recommendation would be to leave as much freedom as possible for creative aircraft design and then, the blank part of MIL-1797A might be a good approach.

Mr Wuennenberg

There are some very stimulating recommendations which your group has made, and the question is whether the floor agrees with these, or wants to make any additional comments, or questions, or contradictions.

Mr Smith, United States

One of the things Ralph A'Harrah always told me to do was try to say something provocative and clearly Number 4 is provocative. One thing I was struck by early on was the statement that specifications are rules for fools and guides for wise men. I[illegible]rally, I don't agree with that, but I do think that in my exposure to the handling qualities world we spend far too much time being technical lawyers and not enough time being technical explorers and guides. So I would go along with Number 4 in that sense and would like to say that I think we should spend more time sharing lessons so that we can guide people as much as is feasible in the context of national interests and secrecy. So I agree with Number 4.

Mr Wuennenberg

Now we come to the next group of papers entitled "Experience with Specifications", and the statement of this group is given by Mr Sella.

Mr Sella, Italy

Session number three was dealing with experience with specifications and the discussion between the session chairman and the authors following the session has identified three major items which need to be recorded again here. The first one really comes from the conclusions and recommendations from the work that Working Group 17 has done, particularly realizing this very important statement, the third one, which says that sufficient knowledge is available for the design of a longitudinal FCS and the use of adequate design criteria. Now this is a very important result: the demonstration of success of 10 years of research work. We have seen that coming from the aircraft of the 70s, which had a number of handling problems like YF-16, Tornado, Shuttle, just to note a few, the aircraft of the 80s, at least most of them like Rafale, X-29, and/or EFP, and recently also the F-22/23 and X-31 have been generally flaw free in terms of flying qualities. I think it is very good news for the next generation — ATF and EFA-like aircraft. This achievement shows that indeed we have sufficient design guidelines and techniques for flying qualities and control law design. Gibson's criteria have been particularly successful in this field.

However, these techniques may address areas for design optimization rather than provide level boundaries. Now this is perfectly reasonable taking into account the fact that most of the designs which were mentioned before are prototypes or demonstrators rather than production aircraft. But the complete specifications, including level boundaries for degraded modes or failure states and also for the definition of service and permissible flight envelopes, are required for the next generation production fly-by-wire aircraft. Now these requirements are to a large extent still missing; the group feels they could be derived from the existing data base, but the necessary effort has not yet been available.

Another important point which has been identified is the fact that flying qualities are inherently subjective while specifications tend to be quantitive. The purpose of showing this viewgraph, figure 1 (from Leggett and Black's paper Number 7), is to say essentially that whatever the specifications say, the flying qualities of the aircraft must satisfy the mission needs. Now this might seem obvious to a flying qualities specialist, but in accepting this figure, there are problems because accepting this requires that someone will have to admit that they got it wrong and will require additional expenditure from a government agency or from a contractor.

So the group feels that flying qualities specialists should really try to educate their managers to accept this as a fact of life and to accept the risk involved; in other words not feel that the flying qualities specification is a performance specification. Now the third point is very, very much related and was also identified in the previous session, and this point to be emphasized once again is the fact that flying qualities specialists and pilots have to work together as a team. It's very, very important that in establishing this dialogue, engineers must not hide themselves behind the book, taking the specification as a Bible, but must understand the relation of the flying qualities to the aircraft task and mission. On the other hand, pilots must help engineers to understand their environment by talking the same language. This concludes my summary.

Mr Wuennenberg

Are there any comments?

Dr Buchacker, Germany

With respect to the last, to communicate with other people you have to learn the language of the other people — if it's German or if it's an engineering language. I would like to make another comment. We talked about numbers and handling qualities specifications etc. I think what we have to bear in mind, especially as we're talking about handling qualities, what is molded down in these specifications is always a mirror image of the abilities of the pilot, the abilities of the human being. If we apply these specifications we have always to bear in mind that more or less mirror image of the human being, and so if we deviate from these numbers we have to reconsider the prerequisites and to reconsider whether the deviation will not require other abilities of the human being; then we have to check that the human being will be able to do that.

Mr Wuennenberg

Now we come to Session Number 4. We have separated Session Number Four (Ongoing Research and Application) into two groups and we are wondering what the two groups independently have found out on the same subject. Mr Woodfield will give the first view of this subject.

Mr Woodfield, United Kingdom

What we thought, as we looked at the somewhat mixed batch of topics which have been covered, all of great interest but somewhat different, was that there were one or two features which perhaps needed to be highlighted.

One feature we did feel was worth highlighting is that although we are concentrating, quite rightly, on the importance of the pilot, and his aircraft and flight control system; the pilot actually interfaces into that system via two other environments: his cueing environment, and in particular, displays and the information those provide; and, of course, his control or tactile environment, the way in which he actually makes demands upon that aircraft. Although it's vital that we do get that relationship between the pilot and his aircraft right, we mustn't ignore that, from time to time, the way in which we provide his cue environment and his tactile environment can strongly influence the overall handling qualities.

Thinking particularly about the cue environment, three aspects of that were brought to mind. First is display of flight information itself, its content, and its dynamics, and I think there are many occasions when one has to only think of the failure of head up displays in some fighter aircraft to provide a safe flight information source. And although we believe that can be overcome, nevertheless it can also produce a major degradation of the flying of that particular aircraft.

Next, the time sharing of tasks. The proportion of your flight time that is available the concentrate on the flying task, in many cases is decreasing. This has been brought out several times. We have to make sure that our requirements for handling qualities take into account the varying load or varying time share available for that. And as I'm now concerned with simulation as well as with flying tasks, I couldn't resist putting in the third one. But I think it is very relevant. If we're looking at activities, as we must, in the simulator and in flight, the relevance of the changes of the cueing environment between the two has to be understood and recognized and taken into account in the translation of results from one environment to the other.

We have much less to say on the tactile environment. The impression I was left with, and I'm not an expert on the various implications in this area, the impression I was left with is that we still have some way to go in understanding the way in which the pilot compensates and relates to his environment and the relationship between the force and position dynamics of those control systems. And the way in which we deal with, perhaps, effective time delays and things of this nature needs to be very carefully thought out in relation to the source of those time delays, I believe, as was emphasized by some of our authors, and cannot be left as a global parameter which explains everything, wherever it comes from.

The third point which was made by our authors, which I felt was an interesting one, didn't come out of the talks. It has, I think, come out of many of the papers in the overall concept and the overall coverage of the Symposium. But my view so far is that we've heard an awful lot of very good work during the last week. People in their research environments are doing a great job.

At the end of the day we have a lot of flight test data obtained by test pilots in research aircraft and research simulation. We're getting very little feedback still, and I think we deserve more feedback from the behaviour of operational aircraft and how the pilots are seeing their own vehicles. At the least, that means that we're not actually establishing quite as firmly as I think we might wish to, the areas where there is a really firm need for doing the work that we're doing and I just leave that thought with you as a concluding thought.

Mr Wuennenberg

Is there any comment to these statements, especially the last one which I think is very provocative, but it hits the point exactly.

Mr Morgan, Canada

I'd just like to comment on the last suggestion there and it's a lovely idea, but I think there's an intense difficulty in taking a body of pilots at large in their day-to-day machines and trying to get any feedback at all that would be meaningful in the handling qualities research field. In the general tenor, or environment in which the line pilot works, be he civil or military, it is such that he is, he gains stature in his job amongst his peers by adapting to and coping with machine difficulties. If you ask the average line pilot about his aircraft, "It's the best thing that ever flew".

Mr Wuennenberg

I would now like to ask the second group on the same subject for their views which are presented by Mr Baillie.

Mr Baillie, Canada

Clearly, as we got together in our groups you'll see some common themes emerging, some of the stuff that we decided to look on is common with what people have already talked about. We're using a term that was around for awhile, I haven't seen it lately "control configured vehicles"; clearly we're accelerating the trend to do that. Some of the papers that we've seen, especially in that last session, talked about that and we've got to optimize our applications of that type of technology for the new tasks and missions that we're foreseeing. The question that we raised a number of times is: in what main areas of research should we be focusing our limited abilities? How should we address the possibilities of configuring the aircraft for a specific mission in a specific time? We've got to look at the interface between these specific roles and we're possibly not doing that at the moment.

This second one was a common theme. A lot of our research results which, of course, we present in fine form up here and we get great charts, but the design community gets them only through the specifications. So, as we're learning to design new vehicles we're going to have to continually reassess our specifications. Rather than updating every 10 years, we should almost be doing updates every couple of years or have ongoing updates.

Part and parcel with that, and the fact that we're dealing with these vehicles configured for a given task, is the fact that we've got to use this mission task oriented principle as is happening in the rotorcraft world and the fixed wing world. We've got to continue to use that. But we've also got to make sure that the specifications reflect the intent rather than the absolutes that we seem to be able to find all the time. And clearly the reason for doing that is avoiding the use of a spec as a cookbook. We've got to make our specifications reflect a lot more the intent of what the possible problems can be if you don't meet the criteria rather than saying, "You must meet this criteria". And to do that the helicopter spec has some nice things which I think should be followed including sets of manoeuvers that *may* expose the problems that are otherwise inherent if you don't meet the spec and also descriptors of the criteria to allow a reasonable engineering interpretation by other parts of the community.

The third point we made: as our vehicles become more integrated we've got to get more and more cooperation between the various people putting hardware and software into the vehicle. We're the flying qualities community, but the flight control guys are sometimes down at the other end of the hall or in another building. The aerodynamics guys, some are represented here, some are not. Avionics clearly is becoming more and more of an important part of the vehicle ability to do the flying qualities and to do the task and somewhere in all of that are our human factors colleagues — [illegible] each we're all human factors colleagues when we start to think about it. The important thing that I put forward [illegible] group is: how do we improve our cooperation between those groups and how can we educate our counterparts so that they understand what our criteria are trying to get at?

The final point was touched on a little by Mr Woodfield, simulation is becoming a design tool that's used universally throughout the field these days and we are getting better and better at it all the time. It's a subject for different conferences. But we must continually ask ourselves where is it appropriate and where is it inappropriate and try to make the absolute best use of this scarce commodity of time on different simulators. I ask you people, what improvements should be pursued in our simulation areas? What sort of things can we do to improve the data that's coming out of simulation in the flying qualities world? And finally we have a number of full scale in-flight facilities, but the majority of them are getting old and we should be starting to continue to develop new facilities so that we can use these as tools to validate our other work on the ground. I hope that stimulates a little discussion.

Mr Wuennenberg

I think there was some overlap to the foregoing statements, but nevertheless I think also some new aspects that have not yet been covered, so we have been glad to come up with these two groups on the same subject. Is there any comment on the statements presented?

Prof. Campos, Portugal

I would comment on one. This was the cooperation between different subjects and I would mention in particular aerodynamics and control. Sometimes I wonder whether the complexity of the control systems is not partly due to inadequate aerodynamics, I mean, if the vehicle is designed and these qualities turn out to be less than desirable, then the tendency is to say, "We will make a control system which will fix the problem". And I wonder whether sometimes it might not be simpler to fix the aerodynamics and have a simpler control system. In a sense, the question I'm putting is whether we don't fall into a situation where we have people who know a lot about aerodynamics and less about control and those who know a lot about control and less about aerodynamics. Of course, one has always the argument, "Well if we're to fix the aerodynamics we have to change the configuration and that's very costly". But then a complicated control system may also be costly to validate in the number of flight hours that you have, so I'm not sure whether we always arrive at the best compromise. I mean, some people argue we can make almost anything fly, but is that the best solution or should it not in some situations be better to try to improve the aerodynamics and perhaps have simpler control?

Mr Wuennenberg

Well it's challenging; we should give this question to the Fluid Dynamics Panel to clarify whether they could improve the aerodynamics. I don't know if any one of you want to comment on that?

Mr Elbel, Germany

In fact, there was a meeting in Lisbon about this some years ago and I attended that and the message which came out of that meeting essentially was "fix the aerodynamics first!" I think that should be taken to mean not going to a degree of instability which you fix with a flight control system or fancy computers; fix the aerodynamics first as far as you can do that.

Mr Wuennenberg

Yes, but we have seen this morning in the nice video from Rogers Smith how complicated this could be at high angle of attack this elimination of the vortices coming from every edge and from the fuselage nose and so on. They are very dynamic so it may be very difficult. I know that the aerodynamists are using tremendous amounts of supercomputing to get it clear, but this technology seems not to be solved at the moment.

Prof. Schaenzer, Germany

It is my present feeling that one of the real problem areas is education of the students and you will really find only a few students who are able to be good in aerodynamics and good in flight control at the same time. You have a real separation in these two areas and I think we have to try to educate them in a better way so that they can understand both areas.

Prof. Ward, United States

I'd like to second that thought and particularly in the United States as I think Prof. Schaenzer has a much better opportunity to do that than we do in this country. Having come from the military after 20 odd years and watching specialists work and then trying to train young people to be specialists, I've come to the conclusion that we've just about cut our feet off by not being able to have people who can integrate and design. The Germans, to a very great extent with your emphasis in university on the integrator and his importance are doing well. Try to find a US university that *has* emphasis on design, and they are very few and far between.

Prof. Innocenti, United States

I agree with what you said. On the other hand, the cooperation — the integration of such different activities towards a better flying qualities design — it appears to me more nowadays a management choice. That could be either corporate management or could be government; there's a lot of complexity involved in putting together people from aerodynamics. I agree on the lack of integrated design capabilities in our universities. As a matter of fact, I happen to be very lucky to have a dual education and experience both, with Old World and New World education. In this country, unfortunately, the level of technology and the ease in generating numbers through computers has phased out most of the old engineering and manufacturing capabilities that we used to teach or — they used to teach me, let's put it this way. And it seems to be a one-way street in that we have just this year completely eliminated any CAD/CAM course in our curriculum; and not to tell you about materials and manufacturing and drawing and all that stuff — it's very difficult. I think it's a very general national problem that clearly cannot be solved by a single place and, if it is a problem, has to be addressed in a larger context.

Mr Cooper, United States

I work for the Navy, Point Mugu and philosophically, certainly, getting back to the original question of not using controls as a fix for inadequate aerodynamics, I agree with that. Operationally, certainly in the military, we're doomed to always have less than the best aerodynamics. For instance when you're carrying significant external stores and you drop them, you change the aerodynamics and the mass properties of your vehicle. Another operational activity would be mid-air refuelling and yet another, which I wanted to touch on later, was battle damage. You're just going to have to have something to fix situations that degrade your aerodynamics, hence control systems.

Mr Wuennenberg

We now turn to the final session which was entitled "High Angle of Attack and Large Attitude Maneuvering". I have to say this group has a disadvantage: since they couldn't meet after the session, they had to meet prior to the session; but hopefully they have an opinion on their subject. This statement will be presented by Mr Robinson.

Mr Robinson, United States

One advantage of being the last one is that you can have a very messy viewgraph and use that, the excuse that you're the last one and didn't have time to do it; thereby my excuse. Before I show it, I do believe that we concluded that in the conventional flight regimes that we see handling qualities in terms of criteria and, I think, specifications -- the two kind of go together because, as we pointed out, specifications really are becoming the guides, the rational guides, which you design to, and then these become the criteria, especially with tailoring.

I think the biggest problem we identified is the one ironically which was just discussed. That's the one that the advent of electronic flight controls, in particular, digital flight controls has created a schism, where what I, in our parochial language say, the "electron pushers" -- in other words, the electronics people -- do not relate to the aerodynamics people; and bridging that gap is a place where there's tremendous progress yet to be made. However, in the other area, and I think Rogers Smith said it beautifully this morning, "discovery is in progress"; I coined that slightly differently. In the area of high angle maneuvering and agility at all angles of attack there is progress being made in the areas of theory as Carmen Maza pointed out. We didn't talk a lot about the computational capabilities, but certainly for high angle of attack, highly agile vehicles, be they helicopters or fixed wing vehicles, to be practical we're going to have to have computational design capabilities. With the advent of supercomputers, use of Navier-Stokes through computational fluid dynamics is coming along. And, as a matter of fact, we are going to use the X-31 and the F-18 HARV as points of correlation to show that those computational tools in fact, work. I mentioned specifications, the specifications also include the design tools. The thinking of the designer is going to have to be different; particularly the thought of an airplane performance man, the sizer, the guy who puts the configuration down, establishes the basis of whether the aerodynamics of that airplane will be any good or not, who thinks of the airplane as a point mass. That's no longer germane in the regime we're talking about; it's a six degree of freedom point. Therefore the bridge has to be between those aerodynamicists and the flight controls people again. And certainly, nobody has any faith in any of that until you can fly it.

Agility, the current status is we're groping for a common definition. I'll have to say that I personally was very heartened here to see the ... homing in on the rate of change of the state of maneuver, or the second derivative of the flight vector so that it is coming along, but what specifically that means is yet to be defined. You hear words like functional agility that pilots tend to use. What they're really talking about, perhaps, is the time to a proper firing solution. That has embodied much more than agility in my personal opinion, but I have my definition of agility and there's probably 30 other ones, at least, out here in the audience. That's the point. But we have to have a common one that the pilots can understand, designers can understand, and, from that, we can understand the increment of combat utility that we get out of it.

High angle of attack, we have defined the pay-offs, I think quite rigorously. Both MBB and Rockwell have tools to do this and those tools are based on our own national government tools that we've adapted to be six degree of freedom models. It's a matter of validating those. However, we clearly — after we fly these airplanes — will have to develop the requirements and criteria and be able to compare those to the other configuration drivers you have, such as observables, such as conventional performance and cost.

So what are the challenges? The clear challenge I think is to develop the path from flight test back to the criteria so now the designers know how much agility, how much maneuverability, how much handling qualities do I have to put into an airplane and what are the trades between these? Therefore, we can design for high alpha and agility and we can understand those trades. That concludes what our recommendations would be, and ends my comments.

Mr Wuennenberg

I think, you have exactly covered everything that the audience has in mind, so there are no questions left.

I would like to thank all the Session Chairmen, authors and those who contributed from the floor, for making this an interesting and challenging discussion period, and an excellent ending to this Symposium.

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14. Abstract

The validity of constraining the responses of today's control dominant aircraft to conform to the classic flying qualities criteria derived from stability dominant aircraft experience has been an issue for many years. The introduction of full time visual scene enhancement with sensor fusion, and computer generated/interpreted night scenes, also escalates display dynamics into the arena of flying qualities concern.

Integrated flight and propulsion control schemes and direct force controllers have the potential for completely coupling all the sensors with all the controllers to provide any combination of controlled motion from six independently controlled single-degree of freedom systems to a single completely coupled six-degrees-of-freedom system.

These new technologies have expanded flight envelopes, reduced drag, increased manoeuvrability, provided the framework for practical gust alleviation and active flutter suppression, and provided flexibility for fault-tolerant, damage-adaptive flight controls.

However, the updating of flying qualities criteria has in general not kept pace with these technological changes. The purpose of this Symposium was to review flying qualities issues today, and to report progress towards their resolution. The following topic areas were covered:

- Flying Qualities Experiences on Contemporary Aircraft
- Flying Qualities Research
- Application of Flying Qualities Specifications
- Flying Qualities at High Incidence.

The concluding "Round Table Discussion" provided the Session Chairmen with an opportunity to share with the Symposium attendees their view of the major issues relevant to their session topic which need to be addressed in the future.

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