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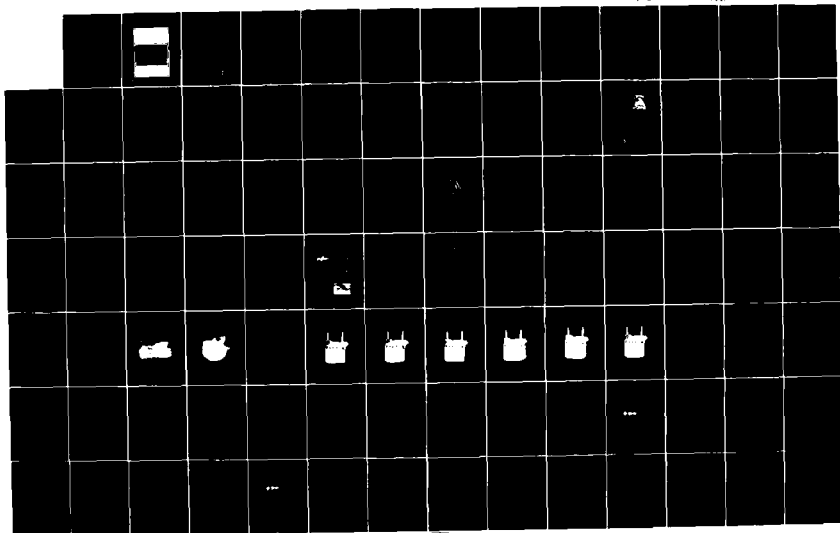
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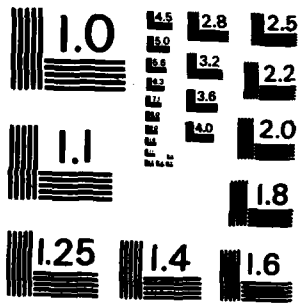
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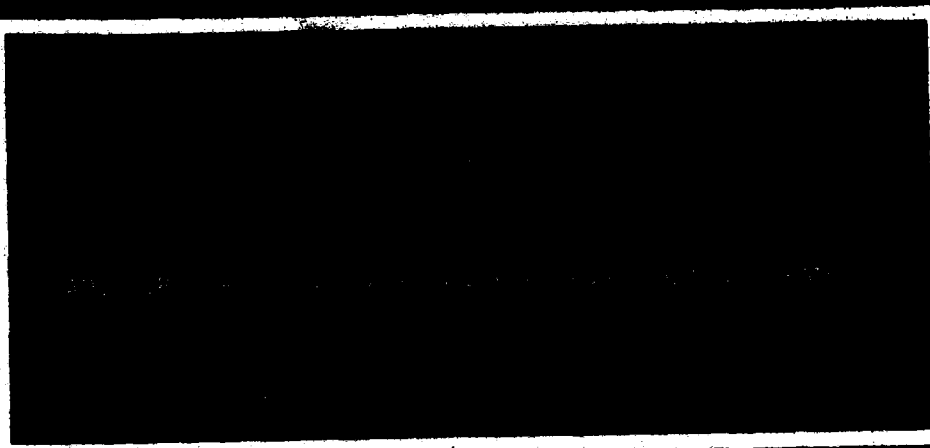
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Integration of Fire Control, Flight Control and Propulsion Control Systems

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NORTH ATLANTIC TREATY ORGANIZATION
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(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.349
INTEGRATION OF FIRE CONTROL, FLIGHT CONTROL AND
PROPULSION CONTROL SYSTEMS

Papers presented at the Guidance and Control Panel 36th Symposium held in Toulouse, France,
from 17-20 May 1983.

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The mission of AGARD is to bring together the leading personalities of the NATO nations in the fields of science and technology relating to aerospace for the following purposes:

- Exchanging of scientific and technical information;
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- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
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PREFACE

Recent advances in systems concepts allied to new technology have led to the possibility of integrating a variety of systems that have traditionally been separate.

In this symposium, the potential, and problems, of integrating mission critical and flight critical systems will be examined.

Such integrated systems can be expected to improve the performance of an aircraft in all phases of a mission. During the enroute and return phases fuel conservation flight profiles may be available. Prior to an attack energy management profiles will be available to maximise the energy of the attacker and during the attack phase integrated fire and flight control will maximise the firing opportunities. Similar considerations apply to missiles and other unmanned vehicles.

In addition, integration of these control systems is expected to provide enhancements of flight safety by reducing pilot workload. A further improvement in survivability is also to be expected from the use of curved attack profiles in both air-to-air and air-to-ground attacks particularly when such systems are coupled to direct force controls or vectored thrust controls.

Des progrès récents réalisés dans la conception des systèmes alliés aux nouvelles technologies ont conduit à envisager l'intégration de divers systèmes traditionnellement indépendants.

Au cours de ce symposium nous examinerons les différentes méthodes d'intégration des systèmes commandes de tir, commandes de vol, et commandes moteur.

De tels systèmes intégrés devraient améliorer les performances des avions dans toutes les phases de leur mission. Durant les phases de vol, en route et au retour, des profils de vol réduisant la consommation de carburant pourront être employés. Avant l'attaque, des profils de vols contrôlant l'énergie seront établis pour augmenter au maximum l'énergie disponible de l'attaquant et, pendant les phases d'attaque, le contrôle intégré système de tir/commandes de vol optimisera les occasions de tir. Des considérations semblables peuvent s'étendre aux missiles et autres engins sans pilotes.

De plus, l'intégration de ces systèmes de contrôle devrait apporter des améliorations à la sécurité des vols en réduisant la charge de travail du pilote. Une amélioration additionnelle de la survivabilité est également attendue de l'emploi de courbes d'attaques incurvées, tant en attaques air-air qu'en attaques air-sol, particulièrement lorsque de tels systèmes sont couplés avec des commandes par forces directes ou par poussées vectorielles.



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*Not available at time of printing.

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TECHNICAL EVALUATION REPORT

by

C. Baron, M.Sc.
Ministry of Defence
London

1. INTRODUCTION

The 36th Symposium of the Guidance and Control Panel of AGARD was held at the Ecole Nationale Supérieure de l'Aéronautique et de l'Espace, Toulouse, France, from the 17th to 20th May 1983. The Programme Chairman was Mr J.T. Shepherd, of Marconi Avionics, Rochester, Kent, UK, who was assisted by members of the Flight Mechanics and the Propulsion and Energetics Panels of AGARD.

2. THEME AND OBJECTIVES

In the early days of aircraft, and to some degree to this day, integration of the complete aircraft system was performed by the pilot, taking his information from his instruments and his own physical sensations, analysing it, computing in his own brain the required control actions, and putting them into effect. Increasing demands on the aircraft system capability led to increased workload; this was met in the first instance by improvement of sub-units, but at the same time sub units proliferated and it became necessary to integrate them in groups, so that the pilot only had to deal with the flight control system, the fire control system or the engine control system, instead of *having to involve himself* individually with all the sub units in these groups.

Pressure for further increases in capability would now be expected to lead to a more integrated approach to the whole aircraft system, and new technology has stimulated the development of new system concepts which together make major advances in this direction possible. To set against better performance and lower workload, however, there will be penalties – greater complexity, probably more in the software than the hardware, with attendant cost and reliability problems, possibly reduced flexibility, and the need to ensure safety while integrating flight critical systems with others which are only mission critical. If progress is to be achieved there are complex balances to be struck – how best to take the pay-off, how to minimise the penalties, and the best balance to strike between them. This will only be successfully achieved by bringing together people of many disciplines – the operators and the operational analysts, the experts in flight control, fire control and engine control, who have traditionally worked to a degree in isolation, specialists in human factors, in reliability and maintenance (of software, particularly) and so on.

The aim of this symposium was to examine the potential and problems of integration, particularly where *flight* critical and mission critical systems are both involved. Its timeliness, insofar as the current state of the art is concerned, is quite apparent, and the potential value of bringing people from the different NATO nations together on the subject is great.

Unfortunately, delegates arriving at the conference expecting to hear the published programme of 24 papers were surprised to learn that 5 papers, all from the US, had been withdrawn at the last minute. The 129 people whose time and travel costs had been committed had every reason to regard this situation as scandalous, and it is thankfully without precedent in the history of AGARD. In the event, both organisers and audience responded to the situation very well, and good use was made of the extended discussion periods which resulted.

3. TECHNICAL CONTENT

Opening Session

The symposium was welcomed to the college by its Director, Ing. Gen. Flourens, who gave an interesting account of the place of college in the French system of higher education, its history and organisation. Having been founded in Paris in 1909, ENSAE claims to be the oldest aerospace school in the world.

The Keynote Address was given by *Dr* Werner M. *Dr* rich, of the West German Federal Ministry of Defence. After touching briefly on the history of *fire* control systems he showed how the advance of digital

techniques has now made the comprehensive integration of these systems possible. However the great complexity of the software that would be involved makes consideration of software reliability most important, and it will set limits to the degree of integration practicable. He postulated a major difference in software complexity between complete or total integration and "reasonable" integration, which he defined as comprising optimised subsystems coupled together through well defined interfaces. This he felt would not fully meet user requirements for system performance and reduction of workload, but would come much closer to their requirements for cost, serviceability, testability and flexibility of integration of new weapons and equipment. It is the responsibility of technical people to ensure that the right compromise between these factors is achieved.

Session I - Integration of Fire Control Systems

The first paper (2, Aden) provided an example of an integrated weapon/fire control system concept, developed to meet the perceived need to attack numbers of tank targets simultaneously and at low cost, from an aircraft at high speed and low level. Innovation features were the case of a hyper velocity missile, with corrected beam riding guidance based on a small amplitude, precision raster scanning laser beam, some features of which had already been tested. The concept seemed likely to stand or fall, however, on the ability to obtain rapid reaction through automatic target acquisition, using FLIR in the first instance, or later, perhaps, a laser radar. The paper aroused considerable interest and provoked many questions, mostly on matters of fact.

The next paper (3) by Brodersen described the integration of an, essentially, existing weapon, the Penguin ASM, with an existing aircraft, the F-16. It demonstrated that the flexibility of modern digital systems can provide for the interfacing of two complex equipments and still allow great flexibility in use. The conclusion that no hardware changes were necessary but very considerable software was required is perhaps typical of our times.

Barling in his paper (4) dealt with the integration of three normally separate sub units to form a single sub unit, with the main benefit of lower cost. Previous conferences have heard descriptions of the combination of headup display and weapon aiming computers, and this was now carried a stage further by the inclusion of the navigation unit. In this case integration was more physical than functional, and the interface with the pilot was therefore different mainly in being confined to a very limited panel space - this constraint was elegantly dealt with by the use of illuminated push buttons. Having thus achieved economy largely through the use of a common computer, the author was led to consider whether this indicated a trend towards total integration through a single central computer; like Herr Fraedrich he rejected this idea, however, because of the need then to raise the integrity of all parts of the system to the level of the most demanding, and also because of problems of common mode failures.

Next Triebe presented a paper (5) written in conjunction with Schrammer, describing the auto attack modes in the Tornado weapon system, with particular reference to delivery of the German MW1 and Kormoran weapons. Unfortunately the paper was almost entirely descriptive, with little reference to the logical basis of the design or to the lessons learned from it; this was probably why this was the only paper in the session to arouse no question from the audience.

The final paper of the session (Roquefeuil, 6) set out in very logical fashion the arguments involved in the design of an electro optical sight and fire control system for helicopter air to air gunfire. The need for quick reaction and minimum crew workload had led to maximum use of automatics and careful choice of the functions required of the crew. Flight tests to verify operation of automatic target tracking and determine optimum parameters were briefly discussed.

The mostly active question periods after the individual papers were supplemented by a final more general discussion, which quickly turned to the role of the aircrew - are they needed and if so what for? While the view was expressed that most of the flight control tasks should be automated to allow the crew to concentrate on the weapon/target tasks, it was also stated that the pilot's feel for the aircraft is most important for man-machine integration, therefore if there is a man in the vehicle he should be a pilot. It might have been pointed out, though it was not, that the objections of Herr Fraedrich and Mr Barling to total integration tend to disappear if there is no pilot - conversely, with total integration there may be no need for a pilot, as, for example, in guided missiles. There was general agreement, however, that air staffs composed of pilots are never likely to produce requirements for aircraft without pilots.

Session II - Integration of Propulsion Control Systems

This was an interesting session in which the feasibility of beneficial technology transfer was demonstrated in generous acknowledgement by engine control system designers of their debt to flight control system technology. The opening paper (7) by Seemann and Lockenour, demonstrated the application of modern flight control system design techniques to engine control system design; it was claimed that digital control systems designed in this way would enable full performance to be extracted from the engine without the constraints normally imposed by control system limitations. The advantages of system design testing by simulation were stressed, particularly the cost savings from getting the design right at the earliest possible time. This was an impressive and well presented paper, though there were those who felt that the design philosophy advocated might lead to excessive complexity.

Rambach (8), in spite of the title of his paper, dealt mainly again with engine control system design, pointing out the need to separate the flight critical basic control elements and treat them differently from the rest. The information

← *Integration of Engine and Control Systems*
normally available within a modern control system can be used for automatic test and status evaluation, so relieving pilot workload.

In a well presented paper (9) by Seabridge and Edwards, various possibilities for integration between powerplant control and other aircraft systems were described, with concentration on integration of the engine and its control system into the aircraft utility systems architecture. Techniques for integration, practical benefits and problems were enumerated in a realistic way. The authors also pointed out the inevitability of integration of engine and flight control in advanced VSTOL aircraft using thrust vectoring.

The next paper (10) was presented for its absent author, McNamara, by Seabridge. It described the design concepts of a system combining engine control, powerplant instrumentation, and data communication with aircraft systems to form an integrated engine control and management system. It took the principles outlined in the first three papers of the session a stage further towards practical design, demonstrating how a very high level of fault tolerance can be obtained by making maximum use of redundancy through the flexibility of digital control and data bus management.

The last paper (11) of the session carried the progression a stage further with a discussion by Eccles of a system under development for the AV8B aircraft. In addition to the engine control its integration with flight control was also dealt with, showing how major performance improvements could be secured.

This was an excellent session, with all the papers reaching a high standard. It was unfortunate that the schedule allowed no time for discussion after the individual papers, but a period at the end of the session provided an opportunity for a lively exchange of views. A good deal of attention was devoted to software testing and integrity, and it was interesting to find the speakers feeling that the problems in this area are over-rated, providing that careful specification and proper procedures are followed.

Session III - Integration of Diagnostics, Self-test and Built-in Test

The possibility of using a common set of inertial sensors to meet all requirements of navigation, fire control, flight control and other systems has been under discussion for some years, and paper 12, by Young et al., described a pioneer attempt to put this into practice. The laser ring gyro is an important building brick for this purpose, since it combines the accuracy required for navigation and the wide bandwidth and quick warm up needed for flight control. This was an excellent paper, with a frank description of the problems encountered which stimulated a lively discussion period. Perhaps the most important conclusion was the need to bring together inertial system designers and flight control system designers - integrated systems need integrated design teams!

In the next paper (13), McKinlay set out to consider the need for increased integration to reduce pilot workload while providing greater accuracy in three dimensional flight path control. His paper demonstrated once again the difficulty in finding the best line of approach to meet the needs of ground attack, in comparison with the relative simplicity of the air-to-air combat situation. The conclusion seems to be that an optimum blend of automatics and pilot control is likely to yield the best compromise between workload, cost, performance and flexibility.

'A diagnosis scheme for sensors of a flight control system using analytic redundancy' described in the paper (14) by Stuckenberg, is an attempt to use software to reduce the need for redundancy of hardware in flight critical systems. The success or failure of such a scheme depends essentially on practical limitations, and flight test results were described.

The last paper of the session (15), by Courtois covered the design of an integrated system of maintenance, in which built in test is managed and recorded by the central computer via the data bus. Thus low level testing can be interleaved with operational use, with the test record available on landing. The advantage of testing under flight conditions should help to remove that maintenance bugbear, the fault which only manifests itself in flight. The following discussion indicated a strong interest in this paper.

Session IV - Integration of Propulsion and Flight Control *in a Integrated Engine and Flight Control*

Cooperation between AGARD Panels was apparent when Prof. Jacques of the Propulsion and Energetics Panel, took the chair for this session. Unfortunately it was also at this point that the symposium was most seriously affected by the problems already referred to, for not only were 4 papers withdrawn from the last two sessions, but also there were no preprints available for 3 of the surviving papers, and no paper appears in the Proceedings for 2 of them, with a consequent impact on the quality of this report.

Paper 16, Design Methods for Integrated Flight/Propulsion Control Systems by Skira and Small, AFWAL/POTC, Wright-Patterson AFB, Ohio, USA, noted the trend which will lead to future fighter aircraft being required to have a wide range of capabilities to deal with a variety of different types of mission and situation. This requirement for versatility, they considered, is likely to lead to the use of two dimensional thrust vectoring and reversing nozzles to obtain better low speed manoeuvrability, and variable cycle engines for faster thrust response and freedom from stall problems. Flight control, inlet control and engine control will all be linked by an overall feedback loop, with the pilot operating a comprehensive manoeuvre command control, designed to make the control loop transparent to the pilot. In response to a question on the difficulty of pre-flight validation of such a system, the answer was by a comprehensive programme of simulation and rig testing.

In the only paper in the programme dealing with systems for civil aircraft, Wuest (18) described a system designed for flight economy in the Airbus, providing automatic control to achieve specified profiles of speed and height rate.

Pattinson (19) presented an aircraft designer's view of engine/flight control system integration, as applied to a future VSTOL fighter. Postulating, like other speakers, a manoeuvre demand control over all components of acceleration, he anticipated benefits in performance from the resulting freedom from the need to design the aircraft to basic handling criteria. The importance of thinking out the interface with the pilot, and of adapting thrust priorities to the flight regime, were clearly brought out. This was a challenging paper, and brought a good response from the audience in a question period ranging over redundancy philosophy, the man-machine interface and computer hardware and software design.

Session V - Integration of Fire and Flight Control

The final session was chaired by Mr A'Harrah of the Flight Mechanics Panel: it was reduced to only two papers.

Paper 20, 'AFTI/F16 Automated Manoeuvring Attack System - A Concept in Combat Automation' by Ramage, AFWAL/FII, Wright Patterson AFB and Lydick, General Dynamics, Fort Worth, Texas described a system designed to operate effectively in the very short time available in high speed low level flight for target acquisition, identification and weapon delivery manoeuvres. The F-16 aircraft, fitted with a triplex digital flight control system and two vertical canards under the engine intake which can give 2 g lateral acceleration in a flat turn, was already in flight test at Eglin AFB, this comprising Phase 1 of the programme. Phase 2, the Automated Manoeuvring Attack System (AMAS) was being prepared for flight testing in Spring 1984; its main features were a FLIR/laser target acquisition and attack sensor, an integrated flight and fire control system, a pilot's voice command and helmet sight system, and automatic fuze setting. The gimbaled FLIR sensor/tracker would provide the information to compute lead angle error and line of sight steering commands, enabling the flight control system to make an automatic attack on the target. The following control law modes will be selectable by the pilot: normal, air-to-air gunnery, air-to-ground gunnery, air-to-ground bombing. To ensure pilot confidence in the system considerable attention is being paid to integrity management, including built-in test, carried out continuously in flight where necessary, incorporation of multiple operating limits, and an arrangement enabling the various computers in the system to test one another. When the system is operating automatically its intentions are depicted to the pilot on the head-up display (HUD).

Answer to questions elicited the following further information: initial target acquisition will be by the pilot using either the helmet sight on the HUD; whether the helmet sight could be successfully used under g was as yet unproven. Voice command had proved very useful in laboratory simulation to insert corrections to an inaccurate helmet sight designation; however, voice command was proving much less successful in flight because of the effects of the oxygen mask, pilot stress etc. In order to provide assurance of not hitting the ground the radar altimeter has 360° coverage in roll, and latest time to pull up is continuously computed and displayed. Pilots apparently like the flat-turn capability provided by the canards because they can line up the aircraft more rapidly: they are not likely to use it above 1 g. In the triplex flight control system, after an initial failure a self test routine is initiated which allows one of the remaining two channels to be selected for use; the original analogue system is also retained as a back-up. Danger of common channel failure due to the use of common software can be averted, it is hoped, by the built-in test, in-flight monitoring and continuous checking of the safety of proposed manoeuvres. (Because no text is available this paper and No.16 have been reported as fully as possible here.)

The final paper of the symposium (21) by Dang Vu and Mercier described a theoretical approach to air to ground gun aiming, in which improvements are obtained by sophisticated multivariate processing of pilot steering commands. The pilot acts essentially as an estimator of aiming errors, using the control stick to command sight line velocity. Although the control law derived is essentially non-linear, a simplified linear version had been tried in simulator tests with some success, and pilots had found it easy to control. Separate controls might become necessary if pilots found it confusing to changeover between this and normal flight control.

4. AUDIENCE REACTION

The symposium was attended by 129 people, and the generally high quality of the question periods and discussions suggested that the majority at least were expert in some part of the field; very few, however, came from the aeroengine industry.

Questionnaires were completed by 19 attendees; their responses have been summarised as follows:

Question	Good	Average	Poor
a1 Quality and relevance of papers, sessions and questions	7	8	3
a2 Did the papers support the theme?*	6	11	
a3 Did the symposium live up to your expectations?*		11	8
b1 Views on operational issues and requirements	3	1	6
b2 Assessment of technology (state of the art)	4	3	

*But there were many complaints about the missing papers.

The response to question a3 is disappointing, since AGARD symposia are usually score better than this; moreover some respondents indicated that their expectations were in any case low. However, there is no doubt that cancellation of so many papers was very strongly resented by the audience.

Responses to questions on particular technical challenges and unresolved problems, while giving a very wide range of answers, focussed particularly on three points. First, what is it that integrated systems, and, it might be said, future aircraft in general, are really required to do? ... there was a general feeling of lack of definition of the problem. Second, the correct role for the pilot in a highly automated system needs much more consideration. Third, there was a general consciousness of the need for much more interdisciplinary interaction (particularly, perhaps, with the engine and human factors experts who were largely absent from this symposium).

5. TECHNICAL APPRECIATION

The answers to question b1 of the questionnaire reflect the fact that we are in a period of considerable uncertainty as to the requirements for future aircraft. Is a high degree of sophistication in the aircraft justified for the better delivery of old fashioned weapons such as bombs and cannon fire? Brodersen's paper (3) demonstrated that a certain level of integration is required by the most modern weapons, but could this be generalised into a more universal requirement? The technology to provide major improvements will clearly be available over the next few years, but which direction to proceed presents a real problem; the thoughtful paper by McKinlay (13) aired some aspects of the problem, but came to no very definite conclusion. Given these doubts as to what future systems may be aiming to do, it is perhaps not surprising that there is also uncertainty as to what the role of the man in the system should be. Even so, there are grounds for feeling that more progress could be made in defining the role of the pilot in semi specific terms, and at the detailed level there should be scope for working out the optimum interface with speech recognition systems and other recent developments. More man-in-the-future-cockpit simulation is certainly required.

On the more directly technical front the conference did very well, the missing papers being at least partially compensated by much useful discussion. Almost certainly it served to put the problems of digital system integrity with a better perspective for many of those attending. The importance of some peripheral areas which are often paid little attention in the formulation of the original system concept, but are nevertheless vital, was well demonstrated by the paper on integrated maintenance by Courtois. A number of papers dealt impressively with the design of systems of a relatively high level of integration, and in particular the prospect of a comprehensive flight control system for VSTOL vectored thrust aircraft in which engine control actions are treated simply as components of flight control, provides one promising direction for future progress.

6. MILITARY-POTENTIAL

As has been indicated already, the potential military benefits of integration were not clearly brought out, there being a general taking for granted that more integration must be good. Integration can be applied, however, in a wide variety of different ways, and the military and the scientists and engineers need to evolve clearer perspectives of which of the wide range of possible benefits should be aimed for, and what cost would be acceptable. There can be no doubt, however, that further progress in integration will be of genuine value in some areas, and clearly the flight control of thrust vectoring aircraft is one of these; more generally, worthwhile improvements will come in both air-to-air and air-to-ground combat, but the best way to obtain them has yet to be thought out.

7. PRESENTATION AND ADMINISTRATION

The French authorities are to be congratulated on the local organisation of the meeting and the facilities provided; apart from very minor problems with busses, everything went smoothly and efficiently. As with meetings in most large cities, delegates were scattered over many hotels, and a recommendation of, say two neighbouring hotels would have helped to promote social activities in the evenings. The Programme Committee undoubtedly put together an excellent programme: had it survived intact it would probably have maintained a higher general level than most AGARD symposia. Even the revised programme served as the basis for what was undoubtedly a worthwhile conference, though it was unfortunate that the rescheduling did not allow a more even distribution of discussion periods; some discussion time after every paper should be the rule.

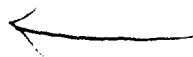
8. WITHDRAWAL OF PAPERS

Although those attending made the best of this occasion, any repetition of a large scale withdrawal of scheduled papers is likely to be disastrous for AGARD, for the word would soon get around that the programmes of AGARD conferences are not to be relied upon, and there is no shortage of travel budget cutters who would exploit that. To insist, as many delegates suggested, on an absolute guarantee of all papers in the published programme would probably be quite impracticable, or at best would lengthen the process of preparation to a degree incompatible with fast moving technology. Undoubtedly something can be done to see that US authors produce papers more in line with their censor's

requirements, and this need not necessarily reduce their value to a symposium - it is not so much technical details that are appreciated by an audience as arguments for and against different courses of action, descriptions of problems and their solutions, etc., which often seem to be quite allowable. Ultimately, however, it must fall on all those with any possible influence on US policy in this matter to attempt to make it more compatible with an alliance of free nations, otherwise not only AGARD, but in the end NATO itself may be irreparably damaged.

9. RECOMMENDATIONS

- (a) In spite of the collaboration of the Flight Mechanics and Propulsion and Energetics Panels, there was little evidence of participation from experts in these areas. Possibly AGARD should seek means of achieving better interaction between different disciplines, in areas of common interest; if this had been the only Spring meeting for all three panels a better cross section would probably have been observed in the attendance.
- (b) If such multi-disciplinary participation could be assured, there are a number of aspects of this topic which many delegates, and I share their views, considered worth further action; in particular symposia designed to bring out the broader issues of where greater integration will really be justifiable, and to deal more positively with the human role in highly integrated systems are advocated.
- (c) Attention is again drawn to the recommendations contained in paragraph 7 above.



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1-1

INTEGRATION OF FIRE, FLIGHT AND PROPULSION CONTROL SYSTEMS,
AN OVERVIEW, RETROSPECTIVE AND PROSPECTIVE

by

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SUMMARY

System integration requires extensive use of digital data processing. Therefore, after a brief glance back into time, some peculiarities of data processing will be shown in more detail. With two examples the paper will demonstrate what the problems are that can arise with a totally integrated system and how some of them may be avoided.

1 INTRODUCTION

In its course, a great deal is sure to be said about the advantages of integrating the individual systems, so I am going to dispense with any remarks about that side of the business and do some reminiscing instead. The result will help you see that data processing alone has enabled complex systems to be made and integrated. Last autumn, the Avionics Panel held a symposium on Software in Avionics. Addressing that symposium on "Avionics Software - the State of the Art", Dr W. Ware said that progress in hardware led to umpteen percent improvement in performance, but that progress in data processing enabled performance to be improved by whole orders of magnitude.

I should like to stress that statement and assert that progress in hardware components is for the most part achievable only when data processing is used in all the various individual developments.

Considering the magnitude of the importance attaching to data processing, and because the problems that can crop up in developing software are frequently underestimated by subsequent system users, I shall be going into some detail in the second part of my paper on software and its peculiarities.

I shall be closing with a pessimistic example of what the problems are that can arise with a fully integrated system and a few hints as to how some of them may be avoided. But knowing the problems does not mean that you have not any more! Everybody probably knows the problems that icy roads can lead to. But I personally do not know of any solution - except the obvious one, namely staying at home - which is worth its salt (fig. 1).

2 A BRIEF GLANCE BACK INTO TIME

There were no such things as control systems when military aviation began (fig. 2):

- Heading and attitude were controlled by the pilot directly through the stick and his pedals.
- The engines were controlled by using the throttle and advancing or retarding the spark.
- Fire control was rule of thumb on the part of the crew - you only have to think of the first bombing raids.

It was not long before they began to think about how to relieve the pilot and crew of some of their work and how to improve the probability of the weapon's success.

In the case of flight control, this led to the development of stabilisers, heading controls, and dampers, which made it easier for pilots to hold their attitude and course and improved the manoeuvrability of aircraft. Almost simultaneously, the development of the autopilot and its integration in the flight control system began. That led to the guidance and control system, to be found nowadays in nearly every aeroplane.

Fire control was at first improved with fixed sighting devices, which, however, in the course of development allowed certain parameters (such as speed and crosswind) to be varied.

Engine control was at first - when aeroplanes flew with piston engines - similar to engine control in automotive engineering, but more complicated due to the different flight levels used. Engine control of modern turbo engines is a more complex task, done by the fuel control unit, which has to consider a great number of parameters. But this is where, as demands upon the engines increase, demands upon the fuel control unit rise considerably, especially in respect of engine handling. It has to

- (1) protect the engine from excessive temperatures, speeds, and pressures,
- (2) ensure rapid acceleration and deceleration without any faults (flow separation, pumping, flame-out), and
- (3) ensure that thrust is always directly related to throttle lever position.

...

Now, after that brief glance at the individual systems, back to more general matters.

Those controls were all analog and contained mechanical, hydraulic, pneumatic, and/or electric components. Since at least flight and engine control have to be reliable in operation - their failure may well be safety critical -, there was a limit to their complexity, and consequently to the integration of the individual control systems.

The fact that fire control was merely mission critical, but not flight critical, led both to the use of complex control units such as mechanical or electric analog calculators, and to the comparatively early employment of the digital technique: after all, failure in flight was not critical.

The digital technique has now overcome one of its teething troubles: it has become very reliable. Another advantage is the tremendous drop over the past few years in the price of hardware components. Digital computer programmes - which is to say software - make it possible nowadays to have associations and carry out functions of almost unlimited complexity. I shall be going into their reliability a little later. And all that makes the comprehensive integration of flight control, fire control, and engine control systems possible.

As I showed at the beginning of my paper, the use of data processing makes possible boosts in performance which are not feasible simply by improving the hardware components. One important reason is that, as I have already said, very much more complex associations and functions are possible. That realisation should not be allowed to lead to the stereotype - and consequently premature - decision, "it is too tricky for hardware; let the software do it", for software has its problems, too (fig. 3).

3 DIGRESSION INTO DATA PROCESSING

Considering the importance of software, I want to delve a little into its peculiarities, specifically

- its complexity,
- its errors and its reliability, and
- its development.

At the same time I am going to try and correct a few widespread, but false, ideas, such as the view that "software is cheap to make and easy and cheap to change". In reality, it is confoundedly difficult to put together software that is free from error, adequately tested, and able to meet all that is demanded of it.

3.1 THE COMPLEXITY OF SOFTWARE

One of the reasons for the widespread assertion that software is cheap is the fact that the complexity of software is vastly underestimated by the majority of software users. If you compare software with hardware (fig. 4), you should always keep in mind that a programme (actually quite a small one) consisting of 10,000 machine code instructions (which is about 2,000 to 3,000 HOL instructions) is equivalent to circuitry consisting of about 10,000 circuits. Whilst the software is apparently easy to survey as a listing, when the hardware is made up of individual components the expense becomes discernible by the extent alone (roughly 200 plug-in cards)!

3.2 SOFTWARE ERRORS/SOFTWARE RELIABILITY

There is a fundamental difference between software and hardware errors. There are no signs of wear and tear or fatigue in software. Software errors are in a programme right from the beginning, but they only come to light when the programme route containing the false instructions is run. Even then, however, it is not certain that the error will emerge if its effect is too slight in respect of the current feed-in conditions. There are examples of errors that have been in programmes for years on end without being discovered simply because the deviations in the final results have been too small. Fundamentally, the same problem exists in the case of hardware, especially if the hardware is complex. But software is complex almost by its very nature, so that this problem is never missing.

Let me give you a small example. A computer with fixed comma arithmetic had to work out the sin/cosin value of angles measured with sensors. It was overlooked that the scales used with the individual attitude angles were not identical, and the result for one angle was the wrong $\sin(2x)$ instead of the right one (x). That error remained undetected until the factor containing the term should have assumed its maximum value (at $x = 90^\circ$), but showed 0 ($\sin 180^\circ = 0$).

But to get back to the general facts about software errors.

We can make these distinctions (fig. 5)

- (1) Systems errors in design and
- (2) programming errors.

...

In the case of system errors in design, the mathematics and logic with which the problem is described and solved contain an error or errors. These are often traceable to difficulties of wording and understanding, both in describing the problem and in specifying the system.

In the case of programming errors, one can again draw a distinction between

- (1) errors on the part of the programmer and
- (2) compiler errors.

Whilst in the case of (1) the programmer puts down wrong instructions, so that his programme does not fulfil the system specification, in the case of (2) the programme, written in a higher programme language, tallies with the system specification, whilst the controllable machine code compiled by the compiler does not. This error, of course, can be traced again to

- a system design error or
- a programming error

in compiling.

What, then, can be done to enhance the reliability of systems containing software?

- You can try to eliminate errors.
- You can try to tolerate errors; for instance by having redundant systems in the software as well. (If you do that, the reliability of a duplex system must be calculated on fig. 6b and not fig. 6a.)

But there is another property which enhances the reliability of software, and that is robustness. By that I mean that the software concerned is capable of standing up to untoward events, such as feeding in data outside the specification.

3.3 THE DEVELOPMENT OF SOFTWARE

I am sure what I have just been saying sounds very nice. But the question is, how can it be used?

To answer it, let us take a closer look at the genesis of software. Broken down only very roughly, it looks something like this (fig. 7):

- In the first instance, there is only a verbal description of the problem (which is to say the system planned).
- That verbal description has to be converted to a mathematical/logical description (say specification).
- On the basis of that specification, taking in a few intermediate steps which I am not going to discuss at this juncture, the programme is draughted and written (encoded) and tested.
- Finally, the software is integrated with the hardware and the whole system is tested.

As you will have learned from literature on the subject (fig. 8), only about a quarter to a third of the errors in software are traceable to errors on the part of the programmer, which is to say to errors which arise in converting the specification (i.e. the mathematical/logical requirement) to the code. Seventy-five percent of such errors come to light before the final test phase. In this sphere, the number of aids to proving agreement between specification and code is continually growing. But most of the errors (about two thirds to three quarters) are system designing errors, which is to say they crop up in converting the verbal requirement to the mathematical/logical specification. And seventy-five percent of these errors remain undetected until the final test phase or even the service phase. That is where aids are lacking, and I doubt whether there ever will be any. The main problem, one that will continue into the future (and not only in respect of software), is "how can it be demonstrated that the mathematically and logically precise description (which is to say the specification) of the project system

- (1) is correct,
- (2) covers all eventualities, and
- (3) tallies precisely with what the user wants?"

But let us get back to the subject of this symposium.

4 INTEGRATED SYSTEMS

In a modern, high-performance military aeroplane - take the MRCA/Tornado, for instance - the two-man crew have their hands full with

- flight control,
- navigation,
- fire control, and
- keeping an eye on the terrain they are overflying and their environment.

...

If all that had to be done by one man alone, he would be overtaxed, and the success of the mission would be very doubtful. But the second man in the cockpit needs a great deal of room, his own supply and rescue system, and his own displays and controls. And that increases the deadweight of such an aircraft by some few hundreds of kilograms. That increase in room and weight means more powerful engines with a greater fuel consumption, which in turn means that more fuel has to be carried. And all that means that a two-seater aeroplane cannot be below a certain size if it is to have the required radius of action in the operational conditions laid down. So user demands for small aircraft with a low fuel consumption can hardly be met. However, in the light of the worldwide oil crisis such demands are becoming more and more frequent. Notwithstanding the difficulties involved, technological progress in recent years makes it possible to come a little closer to meeting them:

Attempts are being made to automate as far as possible individual systems, and consequently the overall system, always leaving the final decision to the pilot. Such automation, however, calls for an integrated system (fig. 9), since an automatic attack, for instance, is feasible only when a fire control system sensor has acquired a target and the guidance and control system and perhaps the engine control system are so governed that the aeroplane automatically holds a heading consonant with the engagement of the target. In a weapon system of that kind the pilot - depending upon the flight phase - can devote himself to the main task, be it flying the aeroplane, using the weapons, or watching a display. This lessening of the burden upon the pilot is a decisive requirement; after all, investigations have shown that roughly seventy percent of the flying accidents that have taken place in recent years have been due to pilot stress.

Now, there I have pointed to two important reasons for integrating

- the flight control/flight guidance system,
- the fire control system, and
- the engine control system.

They are (fig. 10)

- (1) relieving the pilot of some of his work, especially in single-seaters, and
- (2) the improvement of overall systems.

However, integration is also governed by certain parameters, such as

- (1) good serviceability and testability,
- (2) ease of integration with regard to new weapons and equipment, and
- (3) the use of known standards, such as
 - the Data Bus-STANAG 3838 (= MIL-STD-1553 B) and
 - Aircraft/Stores Electrical Interface STANAG 3837 (= MIL-STD-1760).

To wind up this paper, I want to show you where I see the limits of integration and how I think integration should be accomplished.

4.1 THE LIMITS OF INTEGRATION (fig. 11)

If you want to design an integrated system to the optimum by which I mean so that it always behaves in the best possible way - for instance with regard to

- weapon effect,
- fuel consumption,
- pilot stress, and
- survivability,

you can only do so by taking integration into account right at the beginning, in the earliest design phases, which is to say by establishing all the control laws in accordance with the ultimate design. That also - and especially - means that the overall system has to be optimised with all its control laws. There is no doubt that such a system can be developed, but it is going to be very extensive and complex. I do not intend to go into any detail at all on the difficulties that would attend its development.

For the sake of simplicity, let us assume that such a system exists, and that it is used; consequently, it must be maintained and serviced. Before and after missions, you have to find out whether a sub-system or a piece of equipment is defective. The more complex the system is, the more difficult your job is going to be. That applies in particular when you have to find the cause of the error, even though the effect is known: is the error to be found in unit X, in the coupling with unit Y, or is it merely due to the fact that several items are at the edge of the admissible tolerance?

Another thing to be considered is that the requirement for testability is going to increase the complexity of the entire system and, consequently, reduce its reliability, since it is not only in the system itself that errors exist or can occur; they can also be present in its monitoring and testing elements.

We can sum up by saying that a "totally integrated" system will certainly not meet the justifiable demand of the user for good serviceability and testability.

But such a system will not only be used; the time will come when it will have to be modified - to integrate a new weapon, for instance. If you want to ensure that

...

the system behaves optimally at all times with this new weapon as well, you have to re-engineer the control laws of the entire system. It may even be necessary to use other approximation procedures if the required real time behaviour cannot otherwise be maintained (for example the frequency with which the necessary parameters are updated). It will probably be necessary to rewrite a large amount of the pertinent software. And amendments to software are beset by the unpleasant fact that each one may quite unintentionally inject new errors into the programme (or cause errors already present, but thus far dormant, to take effect). You may think that sounds pessimistic, but anybody who has had anything to do with software and its amendment, I am sure, will agree with me. Changes in software is the very sphere where Sod's (Murphy's) law comes into its own: if anything can go wrong, it will!

Since this rather gloomy example is concerned with a totally integrated system, we cannot rule out effects upon the basic flight control system. And in that case the licensing authority will call for the aircraft to be relicensed - after passing all the pertinent tests. Let us not look at how difficult that is with such a complicated system. Here again, we can sum up by saying that a "totally integrated" system will not meet the justifiable demand of the user for the easy integration of new weapons and equipment (fig. 12).

That means that two important parameters demanded by the user are not met; only the requirements for pilot relief and the improvement of the overall system have been taken care of. The third parameter, the use of known standards, may be fulfilled, but in this case, apart from the simplification of the electrical and logical integration of the hardware, it offers none of the advantages hoped for. So integration must not be practised in the way this example portrays it.

4.2 REASONABLE INTEGRATION (fig. 13)

There is not the slightest intention underlying the example (I have just given) to bedevil integration. But to my way of thinking you have to exaggerate if you want to bring the necessary emphasis to bear on the problems that can beset it. And that was the only intent of the example.

There is no doubt that integration is a sensible aim, provided that it is prosecuted in such a manner that it remains controllable. In my view that means that you have to try to design each system, for instance

- guidance and control,
- navigation,
- fire control, and
- engine control,

to the optimum and give it an electrically and logically clearly defined interface through which it can transmit commands to the other individual systems or receive commands from them. That means in particular that standards are used wherever possible for

- data transmission in and between the subsystems as per STANAG 3838,
- aircraft/store interface as per STANAG 3837, and
- programming - here a standardised high order language; for instance Ada.

One system (fire control for example) then controls the others by transmitting command signals through the defined interface to the others (guidance and control for instance). The individual system receiving a command then determines whether it can be carried out, or is admissible, which would certainly not be the case if the fire control system in an aircraft flying at 600 knots ground speed and an altitude of a thousand feet wanted to have a 45° dive to bring the weapon to bear on target. In such a case, the flight control system must report back, "No; only dive angles below 20° possible". That, of course, may mean that the weapon cannot be used. That example will show you that the flight critical systems (in particular the flight control system, but the engine control system as well) have to have a "veto", which can report back and so bring about changes to the commands transmitted by the "requesting individual system".

Now, let us take a look at maintenance and service in this case, too. Since the individual subsystems of the weapon system are connected together through logically well defined interfaces, it is very much easier to trace errors to specific subsystems. Since, however, errors are to be localised if at all possible at LRU level, the subsystems themselves have to be very carefully designed to meet this requirement for good serviceability.

But let us return to the example of subsequently integrating a new weapon. What has to be done in an integrated system of this kind? In this case, the fire control system alone has to be modified (and that little word "alone" does not mean that it is going to be a simple, cheap modification!). Commands issued to the other individual systems still go out through the well defined interfaces. So no changes are necessary in the other systems. Such an aeroplane will therefore not lose its general licence, which - after the requisite tests have been carried out - will merely have to be extended to cover the new weapon. The complexity of the modification here, then, depends mainly upon the complexity of the various subsystems.

There is another advantage of such a system that I would like to touch on briefly, and that is that a considerable amount of the cost of development and testing can be saved by real time simulation (where appropriate as hardware-in-the-loop simulation). Such simulation is much easier if you begin by simulating the individual systems. With step-by-step integration of the individual systems through well defined interfaces you can go over to simulating the entire system. Here again, modifying a single subsystem will in general have no effect upon the other subsystems. The only exception is where the interfaces have to be modified; in that case, however, the original system would not meet the parameters I have severally referred to.

To sum up (fig. 14), then, it may be said of an integrated system consisting of optimised subsystems coupled together through electrically and logically well defined interfaces that it will probably not meet the user requirement for pilot relief and overall system optimisation quite so well as a "totally integrated" system, but that it possesses the decisive advantage of being able to meet the user parameters of

- good serviceability and testability,
- easily integrated new weapons and equipment, and
- the use of known standards,

provided that its subsystems meet that requirement.

5 CLOSING REMARKS

In the examples I have given you, I have demonstrated what problems can occur with integrated systems and how you can get around them. Even an integrated system consisting of optimised systems connected together through well defined interfaces must be very carefully analysed and designed. As I have already said, knowing the problems does not mean that you know the solutions as well!

Looking back at our two examples, let me ask a purely rhetorical question. Which of these two weapon systems can be better handled by a user:

- one that is totally integrated, using standards, or
- one that is integrated through well defined interfaces which do not conform to any standards?

I hardly think it is possible to reach a decision on that point. Either weapon system will give the user a very bad headache. The conclusion is that standardisation and intelligent integration gives a user advantages. One without the other is not enough.

Before I close, I want to say a few words about a matter which has no direct bearing on the integration of individual systems: namely, the multi-use of sensors. The individual control systems all call for a knowledge of certain environmental data, such as

- atmospheric pressure (pilot and static),
- the angular rates of the aircraft, and
- the attitude of the aircraft.

Here, there is a possibility of using the data from one sensor in several control systems, whether as master sensors or as redundant sensors. This field of multi-using sensors is another field which has to be very carefully analysed when the system is designed, since it frequently happens that the various control systems call for differing degrees of precision and failure behaviour. The high degree of precision demanded by the fire control systems must not be allowed to result in having several highly accurate - and consequently expensive - strapdown platforms in a single overall system. In such a case it is quite certainly cheaper simply to have one in the fire control system and to give the flight control its own, less expensive sensors, and to use the platform data there at the most as a back-up (for instance in deciding which of the channels is defective).

I hope I have made it clear to you that systems of well-nigh unlimited complexity can be created by using digital technique. But the cost of such systems is out of all proportion to their degree of complexity.

Our task in future will be to look for a reasonable compromise between the understandable wishes of users and what is financially and technologically feasible. We do not want the next generation of NATO military aircraft to consist of mechanised, armed hang gliders (fig. 15), simply because there are not enough funds to pay for other systems in the requisite numbers, although hang gliders do meet a few important requirements, such as

- a wide radius of action,
- a short take-off and landing capability, and
- a very small radar cross-section.

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Cockpit
World War One

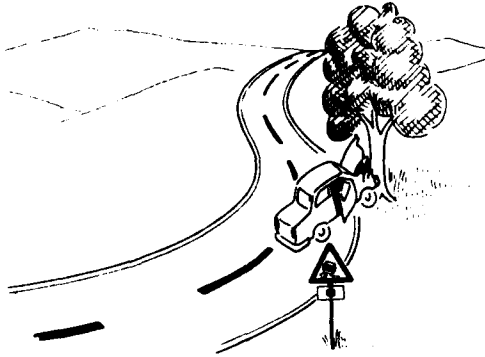


Fig. 1



Fig. 2

ITS' ONLY A MINOR SOFTWARE BUG SIR !
IT WILL NEVER HAPPEN AGAIN

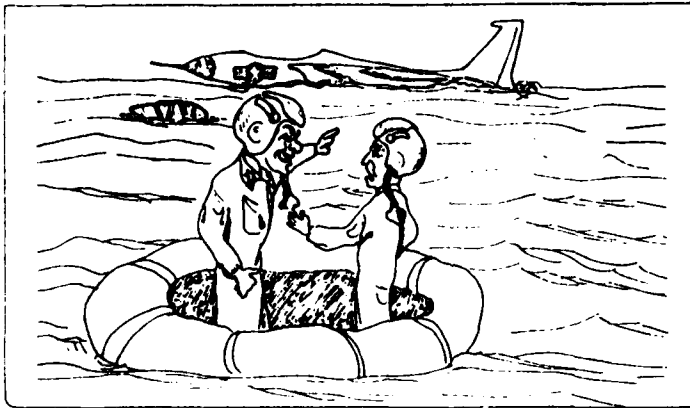


Fig. 3

200 PC BOARDS ↔ 2000 ... 3000 HOL INSTRUCTIONS

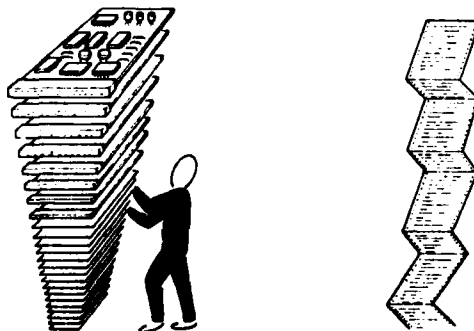


Fig. 4

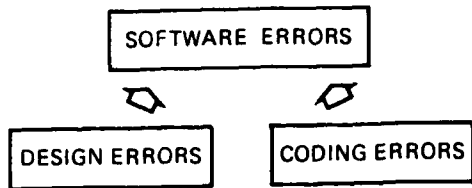


Fig. 5

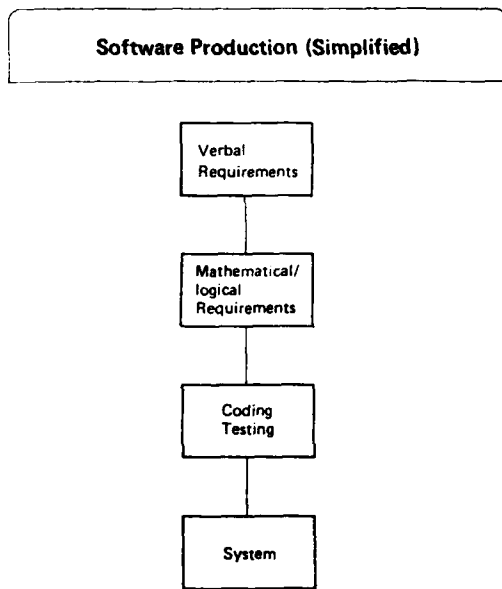
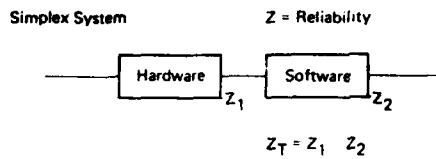
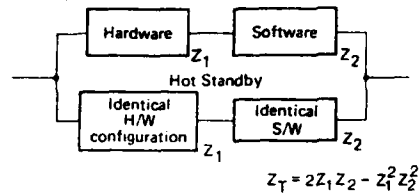


Fig. 7



Duplex System

(a) Naive approach



(b) More realistic assessment

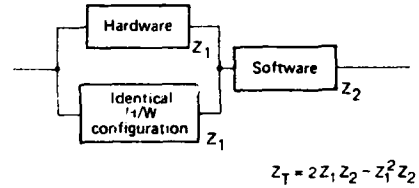


Fig. 6

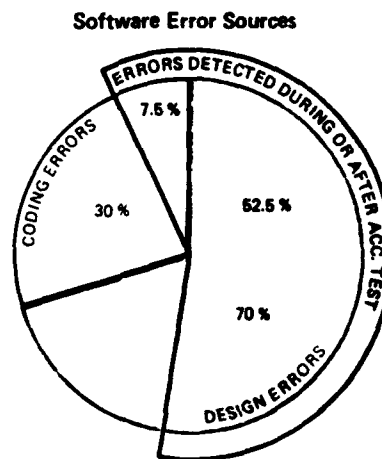


Fig. 8

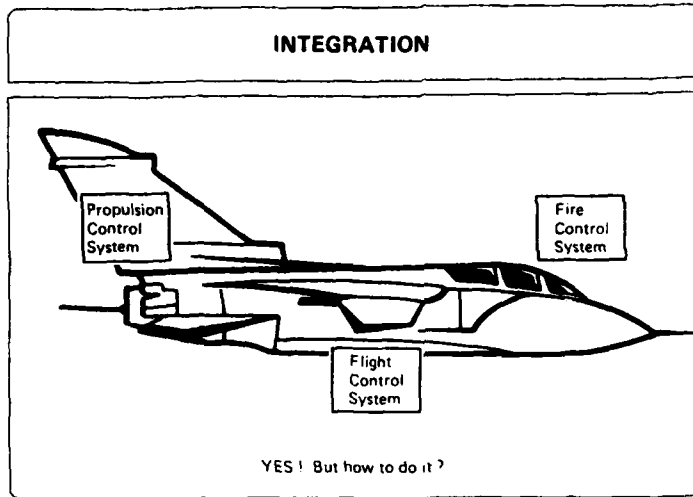


Fig. 9

INTEGRATION

- to reduce pilots's workload
- to optimize the overall system behaviour

Fig. 10

ON CONDITION THAT

- maintainability is enhanced
- new weapons/equipment can be easily integrated
- well known standards apply

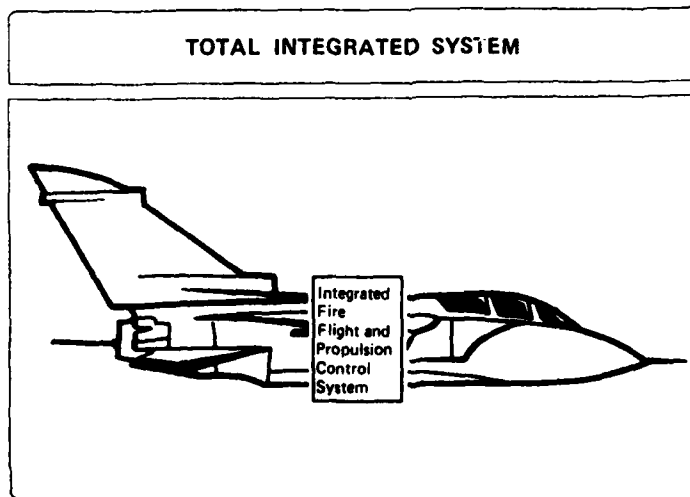


Fig. 11

INTEGRATION

- to reduce pilots's workload ✓
- to optimize the overall system behaviour ✓

ON CONDITION THAT

- maintainability is enhanced -
- new weapons/equipment can be easily integrated -
- well known standards apply ?

Fig. 12

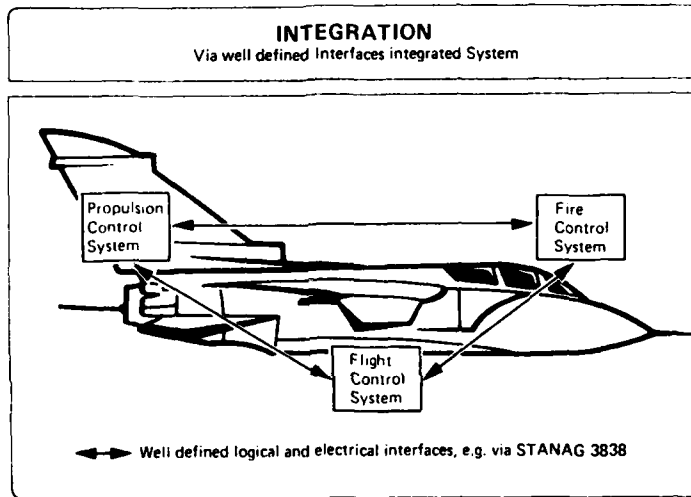


Fig. 13

INTEGRATION

- to reduce pilots's workload ✓
- to optimize the overall system behaviour ✓

ON CONDITION THAT

- maintainability is enhanced ✓
- new weapons/equipment can be easily integrated ✓
- well known standards apply ✓

Fig. 14

Attacking Aero Glider

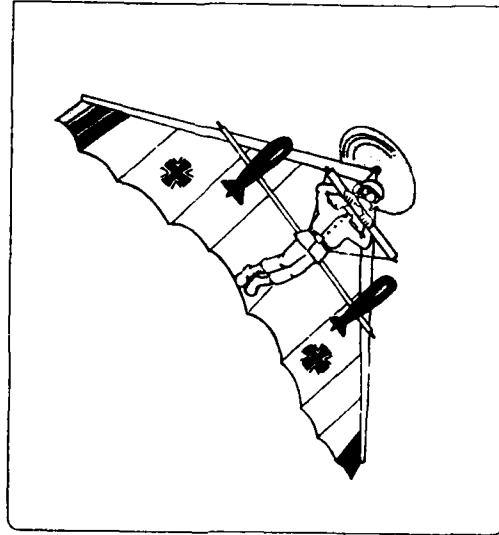


Fig. 15

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2-1

HYPERVERLOCITY MISSILE - BASED ON THE CONCEPT OF AIRCRAFT/MISSILE INTEGRATION FOR MAXIMUM FIREPOWER

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SUMMARY

Close Air Support / Battlefield Interdiction

The conventional approach to weapons for CAS/BI missions has not provided NATO forces with affordable, effective firepower of the order required to defeat SA/WP armored thrusts. The conventional approach has led to weapons that are either too expensive for stockpile and training purposes or too expensive to go to war. It has also failed to provide NATO strike aircraft assigned to CAS/BI missions with the ability to achieve a large number of kills per pass or kills per sortie. The HVM concept integrates current and emerging FCS technology with a low-cost, lightweight missile to provide NATO forces with a significant firepower improvement. The U.S. Air Force has completed a series of ground-launched flight tests that have successfully resolved all technical issues critical to missile guidance, control and accuracy questions. This series of successful demonstrations permits immediate continuation into an air-launch flight test environment to demonstrate the integration of the three critical elements -- the aircraft, the FCS, and the missile.

1.0 INTRODUCTION

The Soviet Army (SA) and its Warsaw Pact (WP) allies have been provided with an awesome array of combat and support vehicles (See Figure 1). The SA strategists and tacticians rely on the precepts of (1) aggressive mobility through extensive use of mechanized infantry and associated support equipment/vehicles, (2) use of their quantitative advantage to generate massed armored assault forces for offensive thrusts and, (3) heavily structured plans to integrate the mobility of forces with the "punch" of massed firepower.

The Soviet and Warsaw Pact threat is both dynamic and responsive. Traditionally, the tactical forces of the Soviet Union and its WP allies have been cast in a role of overwhelming numerical superiority over NATO forces, both in the air and on the ground. NATO has attempted to compensate by developing technological superiority in both anti-vehicular weapons and in armor protection of NATO vehicles. However, the Soviet Union is aggressively pursuing similar technology advancements for its forces while maintaining high production rates for tanks, support vehicles, and artillery. The obvious intent, and unfortunate result, is to ensure continued numerical superiority and to minimize the effects of NATO technical advancements.

The current U.S. response is production of new strike aircraft (F-16 and A-10), improvements in the MAVERICK missile, and development of the LANTIRN fire control system (FCS) by the Air Force; development of the new M-1 Main Battle Tank, the advanced attack helicopter (AAH), and a new anti-vehicular missile (HELLFIRE) for the AAH by the Army. The two major deficiencies in this response are: (1) the weapons are oriented towards improved performance against current Soviet armor and will experience degraded performance against future Soviet armor, and (2) current U.S. weapon systems (including the aircraft/helicopter launch platform) have the capability to attack a limited number of targets due to both cost and carriage (load-out) constraints. This response and related deficiencies are typical of the activities within NATO.

NATO forces need a tactical weapon system capability to defeat the SA/WP massed armored assault forces of the post-1990 era. To meet this challenge, the weapon system must be effective against future armored vehicles; the weapon must permit high load-out of aircraft so that the massive number of targets can be serviced with a minimum number of aircraft sorties; the weapon must be low in cost so that NATO forces can afford the number required to defeat this array of targets; and the weapon, or its variants, should be adaptable to launch platforms of all NATO forces (ground and air) that have a role in the anti-vehicle mission. Such a weapon system would be a true force multiplier that could provide NATO a valid non-nuclear deterrent. The U.S. Air Force is now in the process of demonstrating this capability with the Hypervelocity Missile (HVM) concept.

2.0, CONVENTIONAL APPROACH

2.1 Historical:

The current approach for using tactical aircraft for the Close Air Support (CAS) and Battlefield Interdiction (BI) missions can be traced to World War I. Through World War II, the Korean Conflict, the Southeast Asian involvement, and the Arab-Israeli clashes; technology has provided continually improved aircraft and weapons but the basic operational approach has not undergone a major change. Through the Korean

Conflict, tactical aircraft attacked ground vehicle targets with unguided "iron bombs" and machinegun/cannon fire. Historical data through that period reveals that 4,500 Kg and 10,000 Kg of "iron bomb" ordnance was required to kill truck and tank targets, respectively. This level of sortie inefficiency explains the fact that modern tactical aircraft are still equipped with cannon. It also has provided the post-Korea era impetus to develop accurate guided weapons to hit targets and aircraft fire control systems to find and direct fire on targets.

2.2 Current:

Technology today allows tactical aircraft to perform the CAS/BI missions with modern weapon systems (aircraft/fire control system/missile) that significantly improve sortie efficiency in comparison to the historical data. However, the massive number of targets and limited number of aircraft resources drive one to the important question of whether or not the improvement is adequate. Critical factors prevent the desired answer with current and emerging weapon systems: (1) The physical and operational characteristics of modern guided weapons will not permit tactical aircraft to achieve large numbers of kills per pass or kills per sortie, (2) the cost, physical, and operational characteristics of modern guided weapons have a major detrimental impact on the number of weapons available for training and war and on the demands placed on the FCS. These demands reflect the need to ensure that the few, expensive weapons available in a sortie are used to maximum effectiveness. These same factors are equally true, in an analogous manner, of weapons systems for NATO ground forces which share the responsibility of defeating SA/WP armored assault forces.

Production weapons for defeat of armored assault elements are of two types. The non-autonomous weapons are typically command-link or semi-active guided weapons, such as the Laser Guided Bomb (LGB). The LGB weapon group is comprised of converted "iron bombs" that weigh 250-1000 Kg. The autonomous guided weapons, such as the U.S. MAVERICK, have warheads of reduced weight to reflect the guidance accuracy of these weapons. These weapons have extended range relative to the LGB group. However, even these lethal weapons weigh on the order of 250 Kg. Using the F-16/MAVERICK as the nominal example, a single sortie carries 4-6 missiles and can reasonably anticipate the destruction of 2-3 targets. This represents a dramatic improvement in sortie efficiency compared to the WWII and Korea experience. However, considering the target array in a SA/WP armored assault thrust, production weapons require a number of aircraft sorties to accomplish the CAS/BI missions that exceed practical limits if all NATO aircraft were assigned to these missions. This problem is a direct result of the multitude of targets to be destroyed and of the restricted number of production weapons that an attack aircraft can carry into combat.

The operational characteristics of production weapons pose another problem. The semi-active guided weapons, such as the LGB, typically limit the aircraft to attack one target per pass over the target area. This is a result of the limited velocity/range of the weapons and of the fact that each weapon is guided by continuous designation of the target. The autonomous guided weapons have greater range/velocity than the guided bombs. However, the timelines required to sequentially acquire/track targets by the aircraft FCS and for each missile seeker to lock-up on its assigned target permit a maximum of 2-3 targets to be attacked in a given pass over the target area. Even this number of attacks per pass can be reduced by 50% or more if there exists a requirement to discriminate high priority targets. The limited carriage (load-out) constraints combined with the weapon cost factor and the typical pressure of the tactical situation will usually impose this discrimination requirement. The operational characteristics of the semi-active and autonomous guided weapons prohibit an attack aircraft from achieving a high target service rate. In point of fact, it should not be considered too extreme to say that target service rate is on the order of the aircraft sortie generation rate. Considering the limited number of aircraft allocated to the CAS/BI missions and the limitations on aircraft sortie generation/turn-around, the ability of NATO aircraft to achieve a desirable high target service rate is a serious question.

The cost of production guided weapons is a major factor in several respects. Unfortunately, the cost factor is often misinterpreted. The cost of production weapons varies, based on complexity/capability, from a nominal \$25,000 to \$100,000 or more. A superficial examination would indicate a very favorable cost-to-kill for these weapons against high value targets such as tanks. However, the higher cost and more capable weapons have a degraded cost effectiveness against targets other than tanks and Air Defense Units (ADUs). The lower cost weapons, such as the LGB, would appear cost effective against targets that are not high value. These absolute cost aspects are procurement cost -- the cost to buy the weapons for training and to put on the shelf for war. Procurement cost plays an important role in peacetime defense budgets in that it determines the number of weapons which are to be available for training and for war.

A more subtle and more important cost aspect is often overlooked or under emphasized. This aspect is the cost to use the weapons in war, which is why they were developed and procured in the first instance. This cost aspect includes the definitive cost of aircraft and their avionics/FCSs lost during operations to employ these weapons in combat. There are also costs such as fuel and operations/maintenance which are a minimal contribution to this aspect. An additional and important element of this aspect is the cost of adequate training to establish and maintain aircrew confidence and competence in the use of the weapon. While the value of training is qualitative and intangible, it is inherently obvious that it is essential and plays a critical role

in the ultimate utility/effectiveness of the weapons during war.

The cost of lost aircraft and their avionics/FCSs is by far the dominating factor in the cost-to-kill equation, even to the extent of negating the impact of the procurement cost of the weapons carried on a given sortie. The specific quantitative cost of killing targets is a function of scenario; the number of sorties flown against the target array; the number and nature of the SA/WP ADUs during ingress to, egress from, and the operational tactics used over the target area. The qualitative nature of this dominating cost of aircraft lost in combat is directly coupled with the number of passes per sortie and the number of sorties required to accomplish the CAS/BI missions.

This dominating factor has the effect of increasing the cost-to-kill by a factor of 5-40. The lower procurement cost weapons require multiple passes per sortie and very large numbers of sorties to kill the quantity of targets required to successfully conduct the CAS/BI missions. That is the reason that production non-autonomous weapons experience the upper range of the cost-to-kill increase factor. Concentration on the peacetime budget process has lead NATO to stockpile quantities of low procurement cost weapons that will be very expensive to use during war.

The production weapons of greater operational capabilities and greater procurement cost experience the lower range of the cost-to-kill increase factor. This would then appear to make the weapons that are more expensive to procure the lower cost weapons for tactical air warfare. While this is true, on a relative scale, these weapons also have an absolute high cost-to-kill value.

In addition, the high procurement cost creates two major peacetime problems that will become manifest during war. The first problem is two-fold in that the high procurement cost makes it difficult to stockpile adequate numbers of weapons for either competency training or for war. This is reflected by the strong concern over this problem which is expressed by all NATO forces. The second problem is related to the limited stockpile availability and aircraft load-out of these weapons which makes it imperative that they be fired against high priority targets in the CAS/BI missions; i.e., tanks. This places an immediate burden on aircrew members. It also places a major burden on the attack aircraft avionics/FCS. Both aircrew and FCSs are driven to discriminate high priority targets under combat conditions. The net effects are to reduce the attacks per pass while increasing aircraft/aircrew exposure to ADUs due to the associated increases in required passes and to increase the technical complexity and the associated development/production costs of the FCS.

2.3 Status:

The current weapons and the aircraft systems to deliver the weapons are the result of technological evolution. The technical capabilities represent awareness that targets must be hit to be efficiently destroyed. This awareness is reflected in the guidance accuracy of modern weapon systems. However, the current systems are either low cost to stockpile but unacceptably expensive to use in war or low cost for war but too expensive to stockpile. In no event do any of the production weapons permit attack aircraft to achieve dramatic kills per pass and kills per sortie. This most important facet is driven to some degree by cost but primarily by the limited number of weapons that can be carried in a sortie. Product improvement programs of production weapons are oriented to provide improved performance, but they do not and can not address the inherent problems described above.

3.0 A SOLUTION TO THE TACTICAL AIR WARFARE CAS/BI PROBLEM

Solutions to the CAS/BI mission problems must be affordable, in peace and in war. The solutions must minimize the impact on the aircraft FCS and aircrew in terms of target discrimination. The solutions must permit NATO forces to perform sufficient training to gain both competency and confidence in the weapon system -- the latter only follows the former. The solutions must maintain the guidance accuracy for the weapon to hit/destroy its assigned target. The solutions must exhibit physical and operational characteristics that allow the attack aircraft to carry a large quantity of weapons that will provide the aircraft with high kills per pass and kills per sortie. True solutions will not have any single characteristic but will integrate all of these critical attributes into an effective weapon system. Current and near-term technology will not support the embodiment of all these critical attributes in the missile alone. However, current technology will support a well conceived approach to closely couple aircraft avionics/FCSs and the missile to achieve these important goals. The U.S. Air Force is in the process of demonstrating such a concept -- the Hypervelocity Missile (HVM).

3.1 Concept of Operations:

The HVM concept of operation is based on the following tenets (see Figure 2):

(a) The strike aircraft (such as the F-16 or A-10) will approach the target area at an altitude between 60 and 300 meters. The aircraft FCS has the capability to acquire and continuously track multiple targets simultaneously. This portion of the FCS is functionally designated as a Target Acquisition/Tracking System (TATS). The TATS may be based on FLIR (such as LANTIRN) or active electro-optical radar technology. A conventional radar TATS would not provide the required tracking accuracy

nor the desired covertness.

(b) The strike aircraft FCS also has an Active Electro-Optical Guidance System (AEOGS). The function of the AEOGS is to use target position information obtained from the TATS to maintain the active optical guidance raster on the target. The missile uses a time-position scheme to calculate its position relative to the target (which is centered in the raster) and guide to the target with "hit-to-kill" accuracy. The AEOGS does not "designate" the target by use of reflected energy nor does the FCS track the missile. The AEOGS is capable of providing multiple guidance rasters, to simultaneously guide multiple (6-10) independently-targeted missiles, at a high sequential rate within the AEOGS field of view ($50 \times 25^\circ$). The combination of the TATS and the AEOGS serve the function of the conventional missile seeker.

(c) The aircraft FCS also performs two tasks that are important to missile accuracy. First, the missile time-position guidance scheme requires an accurate synchronization of the missile clock and the FCS/AEOGS clock. This synchronization is accomplished by the FCS immediately prior to missile launch. Second, the FCS initializes the missile with a number of launch aircraft and target variables. This information is used to initialize the missile guidance filters and to compute missile guidance for that short period after launch prior to receiving the guidance raster.

(d) The missile is small, lightweight, low cost, and has high velocity to target impact: 7.5-10 cm diameter, 8-23 Kg, \$5000, and 1550 m/s. These characteristics enable a strike aircraft to achieve a high load-out (for example, 40 missiles on the F-16) of affordable, effective weapons. The high velocity, coupled with the off-boresight capability and the multi-missile guidance of the AEOGS, permits a large number of targets to be serviced at a very rapid rate. These characteristics also serve to minimize exposure of strike aircraft to the air defenses that accompany SA/WP armored assault forces. The high load-out, low cost, and high velocity permit the strike aircraft to attack all elements of the assault force. This negates any requirement for the TATS to perform target classification (tanks versus trucks, etc). In effect, a single aircraft can essentially perform a barrage attack with highly accurate, lethal weapons. If classification were available from the TATS, there would be some enhancement of mission effectiveness since the ADUs could be removed quickly and the strike aircraft could attack the remaining assault force elements with impunity. The combined effect of these tenets is to provide CAS/BI aircraft with an order of magnitude increase in affordable firepower.

(e) The HVM capability has potential for application to the role of ground forces in this mission area. NATO AAFs must not only destroy SA/WP armored assault forces, but they are increasingly concerned with defeating Soviet attack helicopters. The ground forces may also use ground vehicles for this dual role. The U.S. Army has expressed interest in the HVM as a means of meeting these requirements from both launch platforms.

3.2 The FCS/AEOGS Guidance Function:

The missile has no seeker, in the conventional sense, yet the missile is capable of "hit-to-kill" accuracy. The FCS/AEOGS plays a critical role in achieving this accuracy. The AEOGS uses target position information provided by the TATS to center the guidance raster on the target. The missile guidance scheme drives the missile to the time center, which is also the geometric center, of this guidance raster at target impact. Therefore, the accuracy of the positioning of the raster on the target drives the accuracy of the missile.

The AEOGS generates the raster by "painting" raster lines in a sequential manner. The AEOGS requires a fixed elapsed time to generate this $10 \times 10^\circ$ raster. After completion of a raster, the AEOGS goes "silent" for a time equivalent to the raster generation time. The AEOGS uses both electronic and mechanical devices to reposition the raster during the "silent" period. Multiple missile simultaneous guidance is accomplished by positioning this raster on each target sequentially. This process puts the entire missile guidance problem in the time domain rather than the spatial domain. As the TATS establishes track files on each target in the array to be attacked, the FCS logic establishes a reference time slot. The AEOGS is directed to position a raster on each target, with each raster scan to be initiated at a determined point in time relative to the reference time. After all targets in the attack array have been serviced by a raster, the AEOGS repeats the sequential process until a total elapsed time (1-3 seconds) equivalent to impact on the maximum range target has been reached. The FCS logic can then repeat the entire process for a new attack array. The FCS does not explicitly assign a missile to a target; instead the FCS assigns each missile a time corresponding to the start of a raster scan. It is necessary for the missile to have a clock and the missile clock must be synchronized with the AEOGS clock. This synchronization and the missile time slot assignment is done as part of the FCS initialization of the missile.

The AEOGS generates two separate rasters, each from a separate active source. Neither raster is spatially modulated (coded). The function of the Course Guidance Raster (CGR) is to provide data to the HVM during missile flight from a point immediately after launch to a time position in the CGR such that the HVM can receive data from the Fine Guidance Raster (FGR). The CGR has a fixed field-of-regard (FOR) of $60 \times 25^\circ$. The CGR is not agile but is slaved to the TATS FOR.

The function of the CGR is simply to insure the HVM is rapidly positioned such that the missile can receive the FGR. The CGR is generated by a separate source in the AEOGS. The CGR has a fixed FOR that is only effective over a limited range (approximately 3500 feet from aircraft). As part of the missile initialization and clock synchronization, the missile is provided a definition of its assigned time slot in the CGR which corresponds to the spatial position on which the FGR will be centered. The missile uses the time-of-arrival technique to determine its position in the CGR relative to its assigned time slot and hence to determine the missile control commands required to maneuver itself toward the proper time slot in the CGR. Once positioned in the assigned time slot in the CGR, it maintains this relative time position until it receives energy from the FGR. In essence, the missile treats the assigned CGR time slot as a "target" until it receives the FGR. The missile has a "time gate" in which it is to expect its assigned FGR to appear. Once under control of the FGR, the missile "time gates" out any future CGR input. These "time gates" provide the missile the basic mechanism to discriminate CGR energy from FGR energy. A similar mechanism is used by the missile to discriminate its assigned FGR energy from energy due to the FGR of other missiles in flight simultaneously.

The function of the FGR is to provide time-position data to the HVM to insure target intercept. The FGR has a FOR of $10 \times 10^\circ$ that is centered on the target. The FGR is generated by the AEOGS which provides FGR agility by the use of mechanical scanners which can position this FGR within a total FOR of $50 \times 25^\circ$. The details of how missile guidance and control is derived from the FGR are included in paragraph 3.4.

The FCS system provides a number of aircraft and target variables to the missile as part of the initialization procedure. These variables include wind and velocity components, aircraft state data, target state data, and aircraft/target relative geometry and range. These variables are used by the missile to initialize the guidance filters and for initial guidance calculations prior to the missile receiving the guidance raster shortly after launch. A more detailed discussion of missile initialization is included in paragraph 3.4.

3.3 The Simple Missile:

The HVM has generic subsystems similar to traditional missiles except it does not have a seeker, per se, nor does it have a fuse or explosive warhead. In another sense, it is lacking those subsystems that nominally account for 75-85% of the cost and reliability problems of traditional missiles. Further, the absence of these subsystems and the associated weight, volume, and form factor allow the HVM to be small and lightweight. These aspects provide the critical contribution that the HVM offers to tactical strike aircraft -- high load-out of affordable, effective weapons.

The HVM is aerodynamically stabilized by a split-petal flare at the base of the missile. The missile is controlled by the Attitude Control System (ACS) which is mounted in the forward portion of the missile. The ACS has two plenum chambers, each with a nozzle. Each chamber is pressurized by the discrete firing of a high impulse, short duration squib. The missile spins at a nominal rate to commutate the ACS. A roll reference sensor provides reference roll position and permits the missile to calculate roll rate and, thus, when a squib should be fired to achieve the desired course correction. The primary mechanism for achieving a major missile attitude change is the contribution from motor thrust rather than ACS thrust. The motor has a boost/sustain thrust ratio of approximately 10:1. Therefore, missile attitude change response during boost is approximately ten times the response during sustain or coast.

Initialization -- It is desirable to initialize the HVM prior to launch (see Figure 3) with aircraft data (velocity, altitude, roll attitude, and angle-of-attack), environmental data (windspeed vector, air temperature, and density), target data (range, azimuth, and elevation relative to the launch aircraft), and AEOGS data (clock calibration -- to establish the time reference, CGR time slot relative to the reference time for the assigned target, and time for start of the assigned FGR scan relative to the reference time). The HVM uses this data to compute time-to-go, launch jump, filter states, and predicted aircraft/target geometry at intercept. This data is also used to initiate the missile clock synchronization, batteries, the receiver cryogenic cooler, roll reference sensor, and rocket motor. If initialization data is not provided, the missile will have pre-loaded nominal values for initialization. Accuracy is degraded as a function of which data are not provided and of how far the pre-loaded nominal values are from actual values. In the worst case (no initialization data), accuracy can be recovered by restricting the launch aircraft to a dedicated attack on the target (the aircraft stays pointed at the target until intercept) as opposed to the simultaneous attack of multiple targets. Detailed sensitivity and trade studies related to initialization and preloaded nominals will be the subject of future efforts.

3.4 HVM Guidance & Control Concept:

The beamrider concept, with various implementations, has been the historical method of tactical missile guidance using electro-optical (EO) guidance links. The concept has been oriented to launch from stationary points on the ground or from relatively slow airborne platforms (helicopters). The basic concept, regardless of implementation, is for the missile to maintain itself in the center of the EO beam,

which is centered on the target, throughout the missile trajectory. The AEOGS CGR and FGR are analogous to this EO beam. In these traditional applications, the line-of-sight (LOS) between the guidance source and the target is either constant or changes at a very slow rate. For attack aircraft applications, the aircraft/target LOS is changing at a relatively high rate. This factor is a driving influence on the HVM guidance concept which is a significant variation from the beam-rider approach.

Based on initialization data/calculations, the missile computes the expected aircraft and target position at target intercept (position 3 on Figure 4) and establishes an orthogonal spatial reference system based on the predicted LOS in that intercept geometry (the x and z coordinates are shown on Figure 4). Synchronization of the AEOGS/missile clocks at launch allows the missile to establish a reference time for the start of each FGR scan and the FGR scan time is a fixed value. The time-of-arrival, relative to the reference time, of active optical energy on the missile receiver allows the missile to calculate its position relative to the center of the raster (see Figure 5). Additional on-board calculations permit estimates of the real-time velocity and displacement components relative to the reference coordinate system. Figure 6 indicates the geometry/calculations for the x-axis and elevation (the y-axis, azimuth geometry/calculations are similar). The missile now has the information necessary to implement the guidance concept.

The guidance concept is designed to avoid missile reaction to the high aircraft/target LOS rate during missile flight to target intercept and to make optimum use of the ACS control authority. The goal of the guidance logic is to shape the missile trajectory such that major missile heading changes occur early in flight (during boost) and such that the missile trajectory asymptotically approaches the predicted LOS at target intercept. Therefore, two sequential guidance laws are used during flight with transition based on the defined time for boost-to-sustain motor transition.

Boost phase guidance is an acceleration command pursuit guidance mode with the objective being to orient the missile velocity vector to the predicted LOS to the target at intercept. This permits the ACS to provide maximum missile turning during that period of time that the ACS is most effective for that purpose (see Figure 7).

The sustain phase guidance is an acceleration command proportional guidance mode. This phase will complete any remaining heading change required and then maintain the missile on the required intercept line. This is accomplished as the guidance law drives the displacement and velocity components, relative to the reference coordinate system, to zero prior to intercept. The reduced ACS control authority during sustain and coast reduces the possibility of missile over-correction as the missile approaches target intercept. The guidance laws used in both phases are incorporated in the typical logic implementation shown in Figure 8.

The key points are: The missile trajectory is shaped to approach the predicted aircraft/target LOS at intercept; the CGR/FGR are used during the boost phase of flight to shape the missile trajectory; the FGR is used in the latter portion of flight to insure target intercept by nulling the cross-course velocity and displacement components. It should be noted that as time to intercept approaches zero, the actual aircraft/target LOS approaches the predicted LOS. As this occurs the HVM guidance concept approaches, in a gross sense, the conventional beamrider as the missile is driven to the center of the FGR which is centered on the aircraft/target LOS. However, there is a most significant difference. A typical beamrider missile requires the EO beam to be maintained continuously on the target until intercept. The HVM guidance concept uses the guidance raster to update the guidance filters. The missile actually flies on these filters and the absence of updates has a graceful degradation effect on the accuracy of these filters rather than a catastrophic effect. The robustness of this approach has been demonstrated in ground-launched flight tests where the missile has maintained guidance accuracy while not receiving a portion of the updates provided during flight.

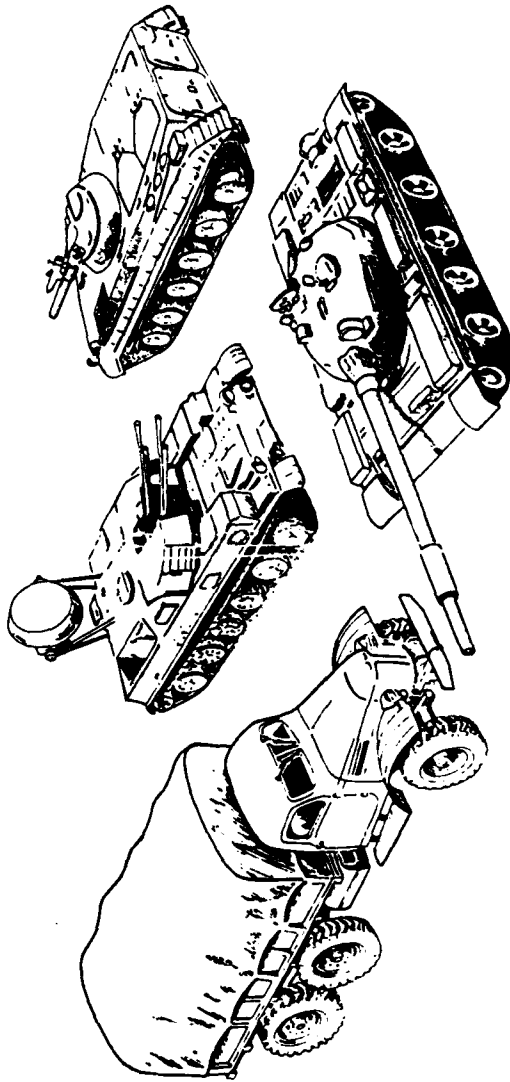


FIGURE 1. ELEMENTS OF ADVANCED ARMOR ASSAULT FORCE

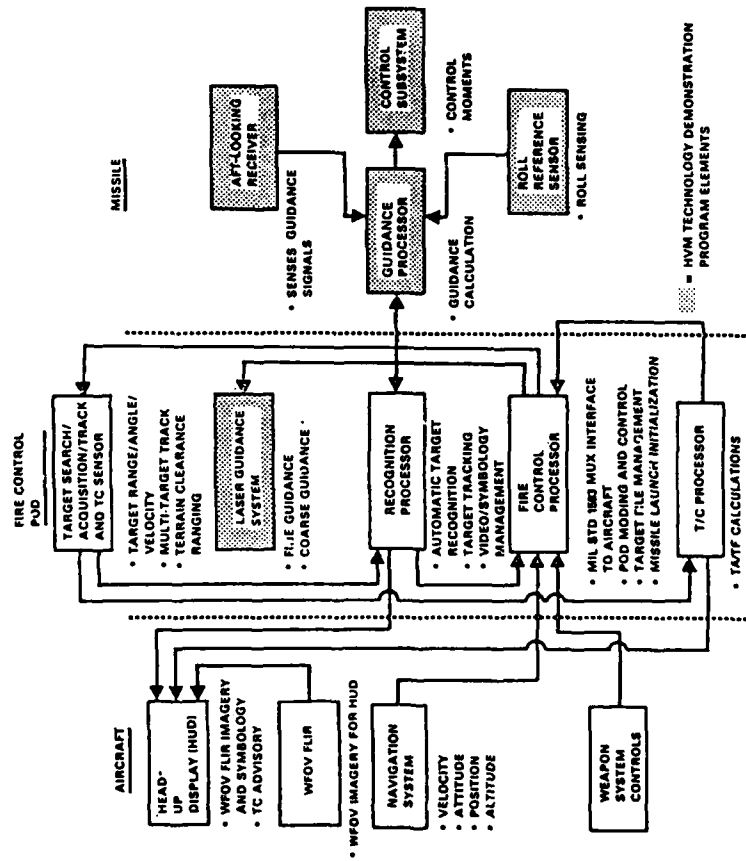


FIGURE 2. WEAPON SYSTEM FUNCTIONS

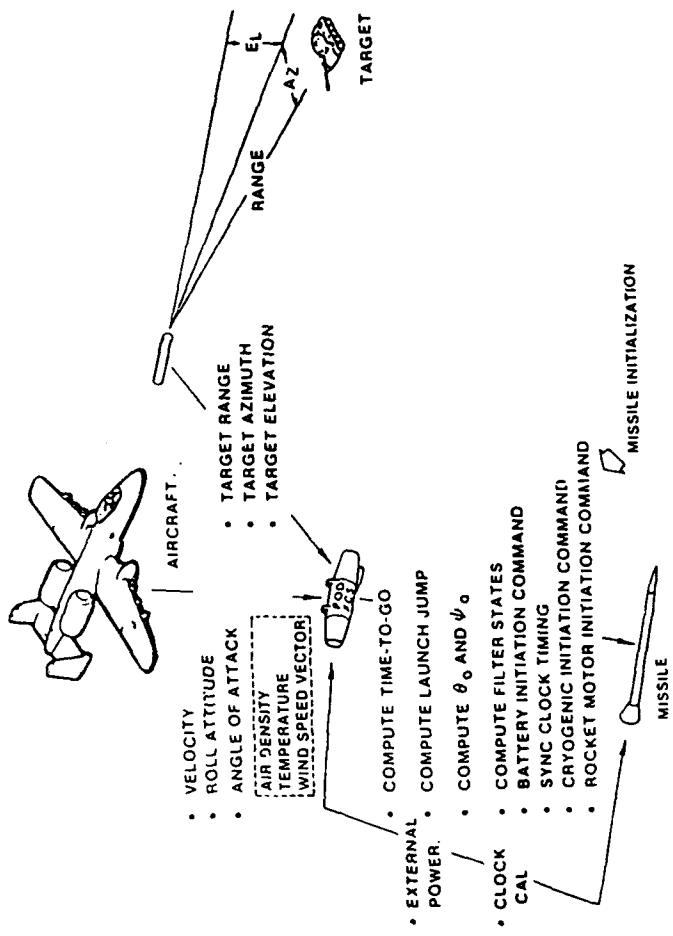


FIGURE 3. WEAPON SYSTEM INTERFACE

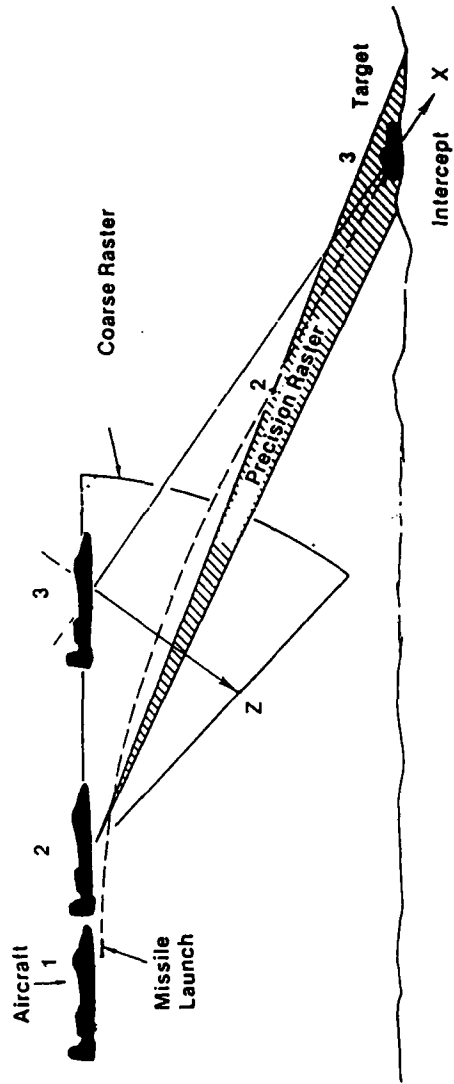


FIGURE 4. GUIDANCE APPROACH

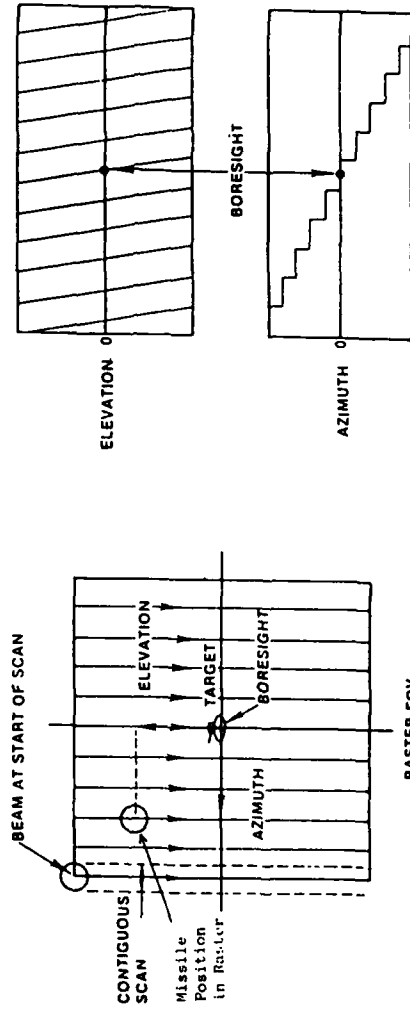
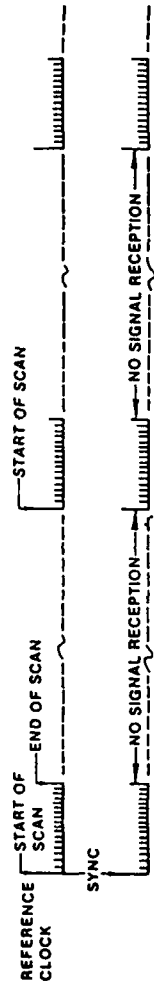


FIGURE 5. GUIDANCE SIGNAL

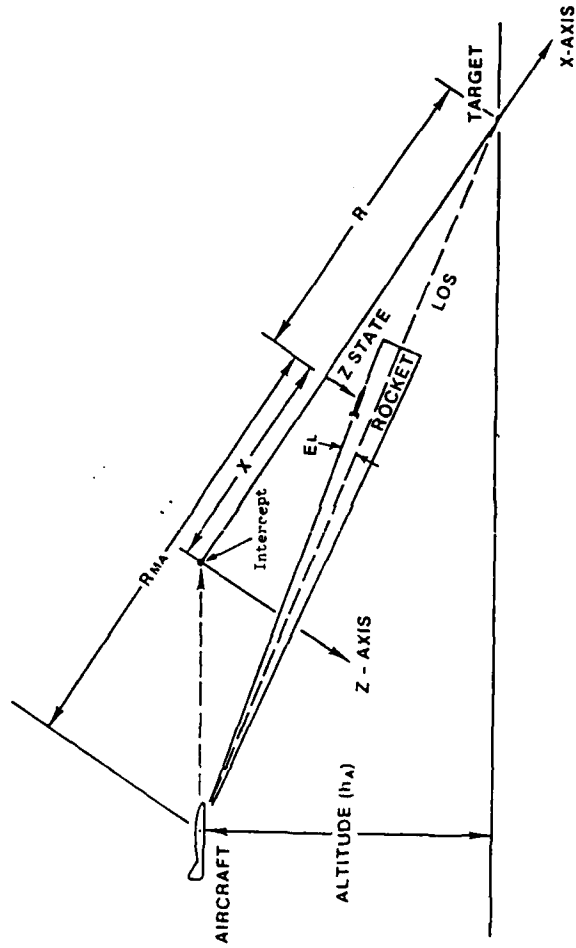


FIGURE 6. SIGNIFICANCE OF GUIDANCE COORDINATE SYSTEM

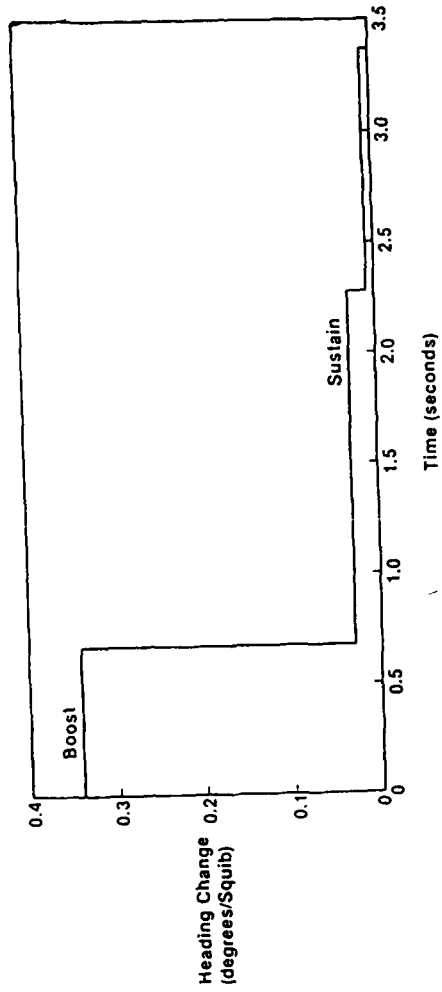


FIGURE 7. AVERAGE HEADING CHANGE PER GUIDANCE FIRING

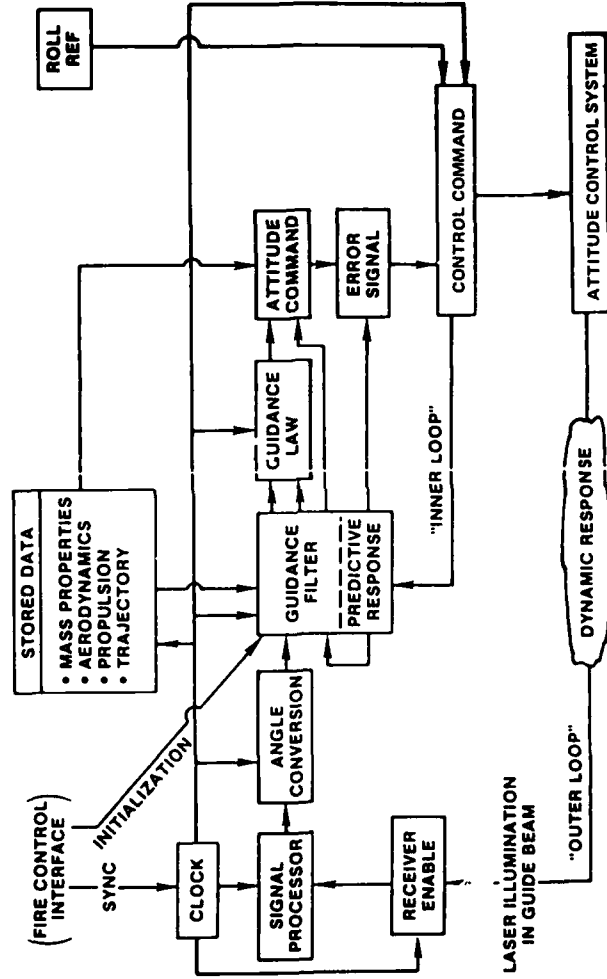


FIGURE 8. GUIDANCE AND CONTROL LOGIC

F-16 PENGUIN ANTI-SHIP MISSILE INTEGRATION

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SUMMARY

The Norwegian Penguin Mk3 anti-ship missile is currently being integrated for operation on the F-16 aircraft. Due to the inherent flexibility of both the aircraft and the missile system, no hardware changes are required on the aircraft. Software changes are designed such that the pilot can operate the weapon to its full performance by using existing cockpit controls and displays in a way quite similar to other air-to-ground missions.

1 INTRODUCTION

The Royal Norwegian Air Force has established the requirement for the Penguin Mk2 anti-ship missile for carriage on the F-16 aircraft. Kongsberg Våpenfabrikk is contracted for the missile development. The Norwegian Defence Research Establishment has been responsible for the development of the target seeker and the inertial navigation system. General Dynamics is contracted (via USAF) for the aircraft/missile integration and certification. Prototype test firings from a Norwegian F-16 are scheduled during 1984.

This paper outlines the main features of the Penguin weapon system as integrated into the avionics and fire control system of the F-16 aircraft. Figure 1 shows an F-16 aircraft equipped with Penguin during captive tests.

2 THE PENGUIN Mk3 AIR LAUNCHED MISSILE

This chapter describes the main characteristics of the Penguin Mk3 air launched missile. This is necessary as a background for understanding the integration requirements and mechanization.

2.1 Outline Description

Penguin Mk3 is a major redesign of the ship-to-ship Mk2 version, now operational with the Norwegian navy among others. Obvious differences between the Mk3 and the Mk2 are that the Mk3 is longer, has wings of reduced span and has a greater range. The general layout though, is similar.

Figure 2 illustrates the exterior and interior of the missile. The target seeker is based on infrared detection. The canard fins are actuated by a cold gas powered hydraulic motor. The altimeter is a radar altimeter. The control unit is all digital and contains the autopilot, trajectory generation, missile internal information bus control and other functions. The inertial navigation platform is a semi strap-down platform (roll axis is gimballed) using two-axes dry tuned gyros. The platform provides 3-dimensional navigation and angular information. The warhead is a modified Bullpup type with a delayed-action impact fuze. Penguin Mk3 will have a new single-chamber sustainer rocket motor using a composite grain. The airframe is roll stabilized by electrically powered ailerons.

2.2 Main Specifications

These are the Penguin Mk3 main specifications:

Length	3.20 m
Wingspan	1.00 m
Body diameter	0.28 m
Weight	350 kg
Weight incl launcher	400 kg
Warhead weight	120 kg
Range	5-40 + km at sea level
Speed	high subsonic
Turn angle after waypoint	0-90°+
Launch altitude	150-30 000 ft
Launch speed	Mach 0.7-0.95
Off-boresight angle at launch	0-50°+
Captive flight altitude up to	40 000 ft
Captive flight speed up to	Mach 1.2
Aircraft maneuver limit	8 g

2.3 Modes of Operation

The basic requirement for the Penguin Mk3 weapon system is to provide the Norwegian F-16s with stand-off "fire-and-forget" anti-ship capability. The pilot must be able to effectively attack targets in open waters, coastal waters and in fjord areas while flying over land or sea. All existing F-16 targeting and fire control avionics must be at the pilot's discretion for effective missile delivery. Fulfilling these requirements and still maintaining simple system operation has been one of the greatest challenges during the weapon operation mechanization effort. A number of experienced fighter pilots have made the most important contributions to the current mechanization solution.

The pilot operation of the Penguin weapon system is best understood by describing the main weapon delivery modes which are (in parenthesis are the display mnemonics as viewed on the Stores Control Display):

- RADAR MODE (RDR). This mode is optimized for targeting by means of the fire control radar. The mode also include the capability of delivery against preprogrammed target coordinates.
- HEAD UP DISPLAY MODE (HUD). This mode is optimized for targeting by means of the Head Up Display optical sight. The mode has two options, namely HUD TURN (HLT or HRT, left or right turn respectively) and HUD DIRECT (HUDD). The HLT or HRT option include the possibility of establishing a turnpoint or waypoint by pointing with the aircraft. The missile will turn to the left or to the right respectively at this waypoint. In addition this option gives the pilot the possibility of preprogramming a descent point where the missile levels out to low level flight. The HUDD option provides no such possibilities, it is essentially an "aim-and-shoot" option. However, a triangulation target ranging procedure will be at the pilot's discretion when using the HUDD option.
- MANUAL MODE (MAN). This mode is a back up mode in case some failure occurs in the aircraft's fire control, inertial navigation or avionics bus communication system. In this mode the pilot can fire the missile at degraded performance while aiming with the aircraft bore-sight axis and keeping the aircraft straight and level.

Figure 3 presents an example of attack sequence using the RDR mode. The pilot lets the radar paint the expected target area. Then he freezes the radar picture on the radar display, breaks away and seeks terrain protection while he processes the display. He slews the target and waypoint cursors to desired positions and fires the missile (or missiles). In this example the missile then separates and turns to the direction of the waypoint while keeping the launch altitude. Thus the missile keeps clear of the terrain before descending, turning and leveling out at the waypoint coordinates. Then it proceeds toward the target area where the seeker selects a target.

Figure 4 presents an example of attack sequence using the HUD mode, HLT option. The pilot aims at the target by placing the HUD target designator (TD) box on the target. He then makes a designation command by which the designated line-of-sight (DLOS) is stored by the Fire Control Computer (FCC). The pilot wants to attack the target from the rear side and holds his fire. He breaks away and seeks terrain protection while he maneuvers his aircraft to the desired launch position. Here he points the aircraft to establish the waypoint, which is computed as the sea level projection of the intersection between the aircraft axis and the vertical plane through the DLOS. The missile is fired and follows a trajectory similar to the one in Figure 3.

Figure 5 presents another example using the HUD mode, namely the HUDD option and targeting by visual triangulation. The pilot aims and designates (First Designation) as in Figure 4. A descent point (here called waypoint) is automatically computed on the designated line-of-sight (DLOS 1). In this low altitude attack the HUD target ranging computation may be extremely inaccurate and the corresponding location of the waypoint may be highly undesirable. As in the example the waypoint may fall on land forcing the missile to descend into the rocks. In order to get a better visual ranging to the target, the pilot flies his aircraft to the side, makes a second designation as shown thus establishing a second line-of-sight (DLOS 2). The DLOS 1 and DLOS 2 intersection coordinates are computed by triangulation algorithms in the FCC program giving a quite accurate target ranging. Then the waypoint (or descent point) is computed correctly close to the target. After separation the missile keeps the launch altitude until the waypoint is reached.

3 F-16/PENGUIN INTEGRATION

From a physical fit and operating envelope standpoint the air-launch Penguin is designed to be compatible with a wide variety of aircraft. A specially designed adapter containing prelaunch missile electronics and ejector release unit assures compatibility with existing launch racks. During missile release or emergency jettison the adapter is always retained on the pylon. Figure 6 is a photo showing the carriage installation, and the adapter is clearly visible.

The F-16 is currently being certified for Penguin missile carriage on weapon station 3 and 7. However, it will be an easy task to extend to a 4 missile carriage configuration (station 3, 4, 6 and 7), if that is desired.

3.1 Electrical Interface

The missile adapter acts not only as a mechanical interface between the missile and the aircraft, it also contains all the necessary electronics for interfacing to the avionics and store control system of the aircraft. The main electronic functions contained in the adapter are:

- Missile DC power supplies fed from the aircraft 3 phase 110 VAC line
- Aircraft avionics digital bus remote terminal interface electronics
- Discrete signal termination and processing electronics
- Missile data bus terminal interface
- Digital processor for processing of targeting, waypoint, launch sequencing, missile status/test and other data
- Digital processor for implementation of the missile inertial platform transfer alignment filter

Figure 7 depicts the electrical interconnection between the aircraft and the missile. No special wiring is required in the F-16 for the Penguin missile integration. Discrete signals (power, arming, release etc) are connected to the standard Remote Interface Unit (RIU) in the pylon and the missile will receive mode, target and alignment information via the MIL-STD-1553 Protocol Avionics Multiplex Bus.

The only change required to the F-16 for accommodating the Penguin missile is software change. The operational flight programs in both the Fire Control Computer (FCC) and the Central Interface Unit (CIU) of the Stores Management System (SMS) have been increased by some 8% each to include the Penguin tasks.

However, these tasks have been partitioned in such a way that the avionic subsystems are not required to accomplish functions or provide interface outside their normal duties. The SMS provides the same weapon control, arming and release functions that it normally provides. The FCC will compute targeting data for the missile while communicating with the Head Up Display (HUD), the Radar/Electro-Optical Display (R/EO) and the Fire Control and Navigation Panel (FCNP). The FCC will also pass the Inertial Navigation System (INS) velocity and angular data to the missile for transfer alignment of the missile inertial platform.

Figure 8 lists the interface wiring between the aircraft and missile. All targeting data are transmitted via the double redundant avionics max bus twisted wire pairs. However, the discrete wire noted MANUAL MODE enables the pilot to command the missile system into a back-up, degraded performance mode without communicating via the max bus.

Figure 9 lists the data messages flowing on the MIL-STD-1553 serial digital avionics max bus between the aircraft and the missile system. Note that the missile system (adapter digital processor) computes all waypoint coordinates in all HUD mode options, based upon the targeting data coming from the aircraft FCC. In the RDR mode, however, all waypoint computation is done by the FCC, thus the data noted MISSILE COMPUTED WAYPOINT are invalid in RDR mode.

3.2 Missile Inertial Navigation System (INS) Alignment

The Penguin weapon operational concept requires a precise missile guidance and navigation system to assure good target selection capability and a high hit probability even in confined coastal waters. The heart of this system is the missile inertial navigation platform. To provide the required navigation accuracy this platform must be aligned with errors down in the milliradian range, velocity initialization must be better than 1 meter per second. Also excessive bias and scale factor errors of the gyros and accelerometers must be measured and corrected for. The only way to fulfill the above requirements on a fighter aircraft is to perform a transfer alignment with the aircraft INS as the reference. This means to compare the data from the aircraft INS and the data from the missile INS and then filter out the missile INS errors based on differences in data. The most convenient data to compare in this respect are velocity and angular data and this is done in the Penguin alignment filter. The alignment filter is implemented on a 16 bits microprocessor in the missile adapter as an 18 state Kalman filter using 3 velocity and 1 angular measurement (azimuth) as input. Figure 10 shows a block schematic of the alignment filter structure. The RELATIVE MOTION COMPENSATION block in the figure compensates for missile offset relative to the aircraft INS location when aircraft rotation occurs.

Figure 11 shows a typical filter response presented as the standard deviation (milliradians) of the angular error estimates as a function of time (seconds). The pitch (PI) and roll (RO) deviations are quickly reduced from initial values of 8 mrad to approximately 1 mrad. However, the azimuth (AZ) deviation is significantly reduced only when the aircraft performs a horizontal maneuver (3 g) which occurs after 60 seconds in the example shown.

3.3 Cockpit Operations

The inherent flexibility of the existing F-16 cockpit controls and displays shown in Figure 12 makes it possible to add the Penguin missile without cockpit modifications.

3.3.1 Stores Control Panel (SCP) Operations

Selection of air-to-ground (A-G) on the SCP will automatically call up a Penguin attack program, which may be preprogrammed to fit the mission scenario. If no preprogramming has occurred, the standard attack program shown in Figure 13 will be displayed. The pilot can modify the attack program by pressing the switches adjacent to the displayed mnemonics. The options available are listed in Figure 13.

The attack program display shown advises the pilot to go to the status page (STAT) because some failure has occurred. Figure 14 shows an example of the status page display. The example depicts a four missile carriage configuration and a somewhat unfortunate situation. Station 7 is the cued station showing a degraded missile. Station 3 has a hung store. Missile on station 6 is still aligning (W) while the missile on station 4 is aligned and ready.

3.3.2 Fire Control and Navigation Panel (FCNP) Operations

The FCNP enables the pilot to key into the FCC memory up to 3 different set of targeting data consisting of target coordinates, waypoint coordinates, target estimated course and speed and the time-of-day when these data were valid. The target coordinates will be automatically updated based on the estimated course and speed and the time-of-day. The updated target coordinates are continuously displayed as cursor position on the radar (R/EO) display and on the Head Up Display (HUD). The pilot may any time correct the displayed cursor positions by slewing the cursors.

A spotter function is included in the FCC software for Penguin delivery. The spotter function enables the pilot to store into the Penguin targeting memory locations the coordinates of the display cursors. He does this by just hitting a switch button on the FCNP. The spotter function is not only designed for Penguin anti-ship delivery missions, but also for reconnaissance missions.

3.3.3 Radar (R/EO) and Head Up Display (HUD) Operations

The operation and symbology on the R/EO and HUD are very similar to normal F-16 air-to-ground display operations. The normal x-y cursor is used as target cursor on the R/EO, and an additional cross-hair symbol is generated for the waypoint. On the HUD the normal air-to-ground Target Designator symbol is used for the target and the offset aim-point Diamond symbol is used for the waypoint. In addition the FCC computes continuously the range to the target in percent of the missile's maximum range capability and puts this 3 digit number up on both displays. The pilot may fire when this number counts down below 100.

3.3.4 Hands-On Operations

Normally all attack program options are selected long before the aircraft approaches the weapon launching position. The remaining operations required are hands-on operations, that is the necessary controls are located on the throttle grip and the side stick controller. Existing switches and controls are utilized to provide the following functions:

- Target and waypoint cursor slew
- Left/Right missile turn selection in HUD mode
- Weapon station selection
- Attack/Status SCP display selection
- Target designation
- Sighting reinitialization
- Launch command

4 CONCLUSIONS

Lessons learned

- No hardware changes was necessary when integrating Penguin Mk3 on F-16. This is thought to be characteristic for advanced weapon systems and advanced aircrafts.
- It took about 2 years to develop the Penguin weapon and fire control software for the F-16 avionics to a reasonable mature state. It is an iterative process involving many people. Our experienced fighter pilots have made the most significant contributions as to cockpit operation definitions and software check-out.
- The Penguin integration required 2500 words of software program in the FCC and 3000 bytes in the CIU. This is probably a typical software volume for integrating an advanced weapon on a fighter aircraft.

Integration program status

- Integration software development completed February 1983.
- Integration certification flight tests completed spring 1983 at Edwards AFB and Eglin AFB, USA.



Figure 1
F-16 with Penguin Mk3

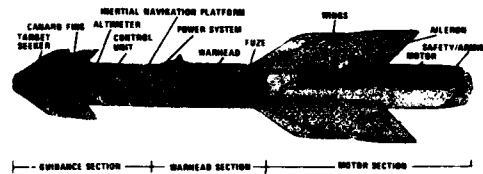


Figure 2
Outline drawing of the Penguin Mk3 missile

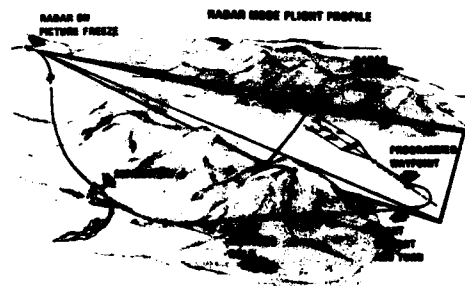


Figure 3
Illustration of a Radar delivery mode sequence

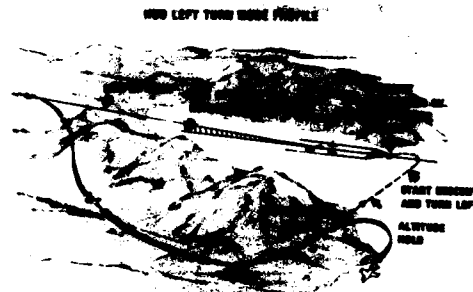


Figure 4
Illustration of a HUD left turn delivery mode sequence

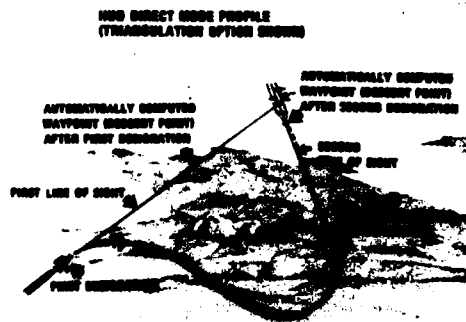


Figure 5
Illustration of a HUD direct delivery mode sequence showing the triangulation procedure



Figure 6
Penguin Mk3 and suspension adapter mounted on the weapon pylon

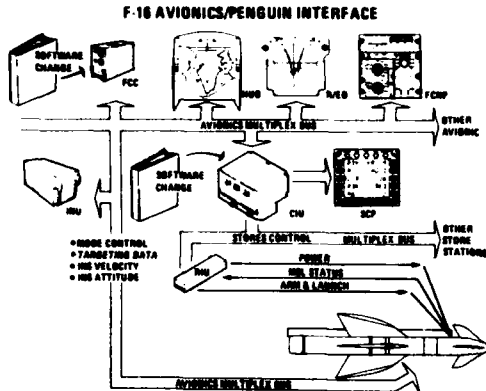


Figure 7 F-16 avionics/Penguin interface schematic

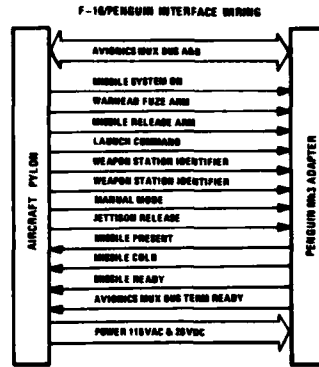


Figure 8 F-16/Penguin interface wiring

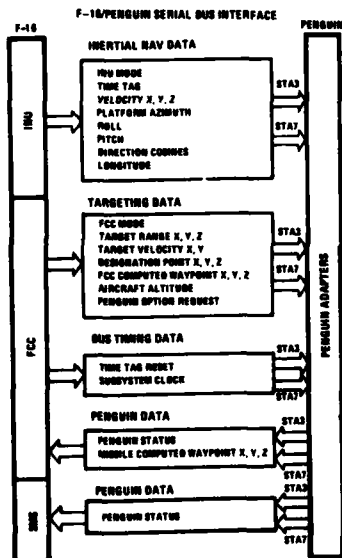


Figure 9 F-16/Penguin serial bus interface data

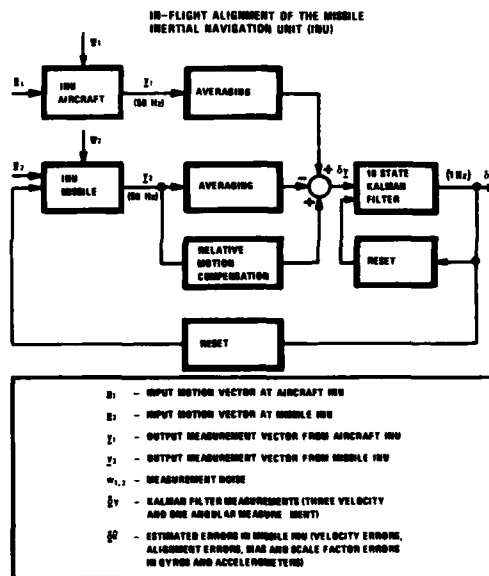


Figure 10 Block schematic of the INS alignment filter

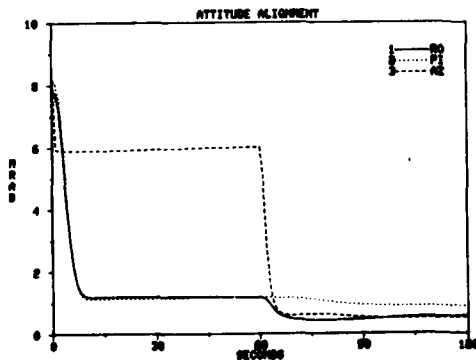


Figure 11 Example of alignment filter response presented as standard deviation of angular error estimates

INTEGRATION OF FIRE CONTROL, NAVIGATION SYSTEM AND HEAD-UP DISPLAY

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SUMMARY

The inclusion of sophisticated Navigation function in a Head-Up Display/Weapon Aiming Computer (HUD/WAC) is described together with integration of the resultant subsystem into an overall Navigation/Attack system. 47

At the end of the paper, a growing trend in airborne systems computing is highlighted.

1. INTRODUCTION

The published theme of the symposium states that advances in systems concepts allied to new technology have led to the possibility of integrating a variety of systems that have traditionally been separate.

The word "integration" is, of course, capable of interpretation in two ways. Firstly, two or more subsystems can be said to be integrated if they interface with each other and this is what is meant when integration of, for example, fire control and flight control is discussed. The second interpretation of integration means that functions which have historically been located in separate subsystems are located in one subsystem.

This paper concerns itself with the second interpretation of integration and the subject is dealt with by describing the equipment that Marconi Avionics (MAV) supplied recently to form one subsystem of an overall Nav/Attack system.

The subsystem itself is a development of a HUD/WAC which was, and still is, in production at the time of the start of the development program.

Being a development of a HUD/WAC, the new subsystem naturally came to be called a Head-Up Display/Weapon Aiming Computer/Navigation System (HUD/WAC/NAV).

As readers will know, the concept of the HUD/WAC which carries out both Weapon Aiming Computation and HUD Symbol Generation is not new. Such systems have been available for some years now and have been reported upon widely. For this reason, this paper concentrates mainly on the Navigation aspects of the system.

2. CONCEPT OF THE HUD/WAC/NAV

The concept of the HUD/WAC/NAV arose a few years ago. At that time, MAV were proposing a HUD/WAC to an overseas customer who, in the middle of pre-contract negotiations, asked us to propose also a Navigation Computer to meet the relatively sophisticated requirements of a specification which he had prepared.

Detailed study of the customer specification led us to believe that there was no need to provide a separate Navigation Computer, but that the requirements could be met by inclusion of the Navigation functions in the HUD/WAC. This proposal was accepted and the HUD/WAC/NAV was born.

(For convenience the HUD/WAC/NAV will sometimes be referred to in the rest of this paper as the HUD).

3. THE OVERALL NAV/ATTACK SYSTEM

A simplified block diagram of the overall Nav/Attack system is shown in Figure 1.

The main subsystems of the system are as follows:

- | | |
|-----------------------------|---|
| Radar | - This subsystem operates in both the Air to Air and Air to Ground roles. |
| Radio Altimeter | - Functionally self-evident. |
| Inertial Sensing Unit (ISU) | - This subsystem provides outputs of aircraft attitudes and 3 axis speeds. In the prime mode, horizontal speeds are derived from Doppler/Inertial mixing, vertical speed from Barometric/Inertial mixing. |
| Doppler | - This subsystem provides outputs of 3 axis speeds. |
| Air Data Computer | - This subsystem, besides sensing and computing of Air data parameters, also outputs angle of attack warning data. This data has two forms - one for display on the HUD Pilot Display Unit, the second is an aural output to the aircraft audio system transmitted via the HUD Navigation Control Unit. |

HUD - This consists of 5 units. These are the Pilot Display Unit (PDU), Electronics Unit (EU), Navigation Control Unit (NCU), NCU Power Supply Unit (NCUPSU) and Pilot Control Unit (PCU).

In Figure 1, the NCU is shown for convenience as a separate unit. In fact, the NCU is mounted on the rear face of the PDU as is shown in Figures 2 and 3. The rear face of the PDU also contains other HUD controls. The NCU and other controls installed on the rear face of the PDU provide a true "Up Front" system controller.

Fast selection of Air to Air Mode is provided by 2 pushbuttons on the throttle. Pushing one or both (3 combinations) causes Air to Air mode to be selected with manual (stadimetric) range set to one of three values. At the same time, the radar begins to search and, after lock-on, radar range is used by the HUD replacing the selected value of manual range.

From the stick, the HUD receives inputs of Pickle and Trigger.

The prime interface in the system is a digital data Bus conforming to ARINC 429M, the M being short for "modified". The principal modification to the ARINC standard was to make the Bus bi-directional with "handshake" discretes (Data Transfer Request, Data Transfer Accept) introduced.

Within the system, all weapon aiming and navigation computing is carried out in the HUD EU with the exception of a back-up computation of present position which is carried out by the ISU.

4. HUD COMPOSITION

As stated previously, the HUD consists of 5 units: PDU (with NCU attached), EU, NCUPSU and PCU. These, with the exception of the PCU, are illustrated in Figure 2.

Entry and display of Navigation data is achieved using the NCU. This is described in more detail later.

The EU contains the ARINC 429M Bus Controller.

The NCUPSU contains the power supply for the NCU plus a standby Bus Controller which takes over automatically when the HUD is switched off. This enables ISU computed present position to be displayed on the NCU following a HUD failure. The NCUPSU also supplies power using the aircraft battery to enable long term data storage within the EU when the EU power is disconnected. Figure 3 shows a close-up view of the PDU/NCU combination. As can be seen from the figure, the rear face of the PDU, upon which are installed the NCU and other HUD controls at the top, also has a space on the top right for installation of the display recorder camera.

The controls located on the top of the PDU rear face consist of the following:

- RAD ALT/BARO/TARGET three position switch.
This selects the type of attitude to be displayed on the PDU. (ie. Radio, Barometric or Target). With TARGET selected, Altitude may be set using the potentiometer below the switch. The RAD ALT/BARO positions also select the type of air to ground ranging used if the radar is not locked.
- TARGET/WGSN potentiometer.
This is used to set the Target Barometric Altitude and, in the Air to Air mode, to set Target Aircraft wingspan. The values are shown on the PDU.
- CAMERA ON/OFF two position switch.
This allows manual switch-on of the display recorder camera.
- BPS Potentiometer
The Barometric Pressure Setting potentiometer is used to set the pressure of the day in millibars, the value being shown on the PDU.
- HUD/OFF switch and potentiometer.
This combined switch and potentiometer switches on the HUD, except the NCU, and allows adjustment of the PDU display brightness.
- NCU/OFF/STBY combined switch and potentiometer.
This control switches on the NCU and allows adjustment of the NCU display brightness. After data has been loaded using the NCU, selection of the standby (STBY) position allows the data to be retained in the EU memory when the EU power is disconnected. This facility allows, for example, data loaded in the evening to be used for a mission the following day. Use of the standby facility is indicated by illumination of the green Light Emitting Diode installed above the switch.
- NORM/DCL (Normal/Declutter) two position switch.
Selection of Declutter removes some of the symbols from the PDU display.
- Mode selector rotary switch.
Selectable HUD PDU display modes are:

NAV	Navigation
APP	Approach
GEN	General Flying
AA	Air to Air Guns and Missiles
CCRP	Continuously Computed Release Point
CCIP	Continuously Computed Impact Point

The Navigation Control Unit (NCU) controls and displays are located at the bottom of the PDU rear face. The controls and displays consist of illuminated push buttons located on a legend-less panel and two opticators for data entry and display.

Conventional NCU's using rotary switches would require more panel space to fulfil the same functional requirements or, putting the point another way, use of illuminated push buttons enables more functions to be placed on a given quantity of panel space.

The legends on the pushbuttons and examples of NCU use are given later in this paper.

Figure 4 shows the layout of the PCU. This unit is installed on the right hand side console and contains the least used control functions for which there was no available space on the rear face of the PDU. The unit is an integrated system panel and contains the following controls

- A rotary switch for selection of ISU mode.
- On-Board Check Out System (OBCOS) control. This is a rotary switch for selection of self-test of the NCU, HUD, ISU, ADC and Doppler. The result of the self test is displayed on one of the NCU opticators.
- Doppler Off/Land/Sea three position mode switch.
- HUD standby sight On/Off and Brightness Control.
- Angle of Attack Warning System audio volume control.

5. MODIFICATIONS TO THE BASIC HUD/WAC

In order to convert, as it were, the basic HUD/WAC into a HUD/WAC/NAV fulfilling the functional requirements specified, a number of changes to the baseline HUD/WAC equipment were necessary, one of them being fundamental.

The necessary changes are summarised below:

(a) Changes to the EU

- The processor was changed to a three level interrupt type, the three levels being 50Hz, 25Hz and 5Hz.
- The Digital Data Bus was changed to ARINC 429M.
- A Bus Controller was added to the EU.
- The number of analog input channels was increased.
- The read/write data (scratchpad) store was increased.
- Scratchpad store hold-up by aircraft battery was added.
- The EPROM program store was increased in order to provide the Navigation Software.
- The NCU was added to the system, it being under control of the EU.

(b) Changes to the PDU

The NCU was added as an "Up-Front" controller and containing a Standby Bus Controller.

(c) NCUPSU

This was added to the system to provide power for the NCU and for the Standby Bus Controller.

(d) PCU

This was added to the system to provide the least used controls. That is, those controls for which it was considered less essential to be provided "Up-Front".

6. SYSTEM WEAPON AIMING FACILITIES

As previously stated, my paper concentrates principally on the Navigation aspects of the system. For completeness, the weapon aiming facilities provided are summarised in this section.

The weapon aiming facilities are as follows:

(a) Air to Ground

- Continuously Computed Impact Point (CCIP) for Bombs, Guns and Rockets.
- Continuously Computed Release Point (CCRP) for Bombs.
- Ranging using Radar, Radio Altitude or Barometric Altitude.

(b) Air to Air

- Guns Snapshot using radar ranging or stadiametric ranging.
- Maximum and Minimum range computation and display for Missiles.

7. SYSTEM NAVIGATION FACILITIES

This section describes the Navigation facilities provided by the overall system. The navigation facilities are numerous and relatively sophisticated as can be seen from the description below.

- Latitude/Longitude coordinates.
- Up to 18 waypoints are provided of which 4 are for storage of overflow points on the ground (eg, targets of opportunity or similar points of interest).
- Storage within the EU of the coordinates of 10 pre-programmed destinations. These are commonly used waypoints and may be called up as a waypoint in the aircraft route plan.
- Speeds used for Navigation are Doppler/Inertial or Pure Inertial in the prime mode backed up by Baro/Inertial following Doppler failure or if the Doppler stays in memory for an excessive period of time. A last ditch back-up mode is use of air data navigation following a partial failure of the ISU.
- Provision on the PDU of a flight director for guidance along the track, real or offset, joining the current To/From destinations or, in the Intercept Sub-mode of Navigation, guidance to intercept an airborne target. The flight director mechanisation uses a track hold control law computed in the HUD EU.
- Pilot selectable display on the NCU of One of the following:
 - o Present Position
 - o Waypoint Coordinates
 - o Distance and bearing to the current destination
 - o Coordinates of the pre-programmed destinations
 - o Groundspeed and track
 - o Cross-track error concerning the current To/From waypoints.
 - o Current Windspeed and Direction
 - o Fuel and Time necessary to fly between any 2 waypoints either by direct flight or by following the planned route.
 - o Fuel and time necessary to fly to the next destination from present position.
 - o In the Attack Sub-mode, the estimated time of day of arrival at the entry point of the attack. In order to assist the pilot to arrive at the entry point of a planned attack at the desired time, a speed error symbol is displayed on the PDU.
 - o Weapons carried on the mission (shown on the PDU).
 - o Mission route plan (shown on the PDU).
- Pilot selectable automatic or manual destination change.
- Planned Fix on a waypoint or Random Fix, both by overflight.
- Data entry either manually or using an automatic data loader.
- Data entry of
 - o Waypoint coordinates. Waypoints can be random or use can be made of the already stored pre-programmed destinations.
 - o Nominal groundspeeds and fuel consumption rates pertaining to each leg of the route plan.
 - o Time of Day.

- o Attack sub-mode data, including desired time of arrival at the attack entry.
- o Intercept sub-mode data.
- o Mission route plan (shown on the PDU).
- o Windspeed and direction.
- o Random fix data
- o Desired left or right offset from track.
- o Magnetic Variation and runway Magnetic Heading for the reference mode of ISU alignment.
- o Mission weapon data.

In the event of HUD failure, back-up Present Position calculated within the ISU and displayed on the NCU.

8. NCU OPERATION

In this section, the operation of the NCU controls is described in some detail and illustrated by two examples. The NCU is used for entry and display of navigation and other data and is, we believe, an elegant, unique solution to a complex problem.

8.1 Layout of the Controls and Displays

Shown in Figure 5 is the layout of the NCU with all pushbutton legends illuminated. This is a situation which never happens in practice since the illumination of the legends, or not as the case may be, is completely under the control of the EU software which allows illumination of only those legends which are available for selection at the particular time during the flight or during the particular phase of NCU usage. Additionally, some functions are mutually exclusive. For example, FIDO, the automatic data loader, is available for selection only before wheels up. A second example concerns the MAN/AUTO pushbutton. This refers to the type of destination change, either manual or automatic. These are obviously mutually exclusive and, therefore, only MAN or AUTO will be lit at any one time. Change from MAN to AUTO, or vice versa, is achieved by pushing the button. As a third example, selection of AUTO causes the CHANGE DEST key to be blanked.

Referring to Figure 5, it can be seen that, for descriptive purposes, the NCU controls and displays may be considered as consisting of four parts:

- Functional Pushbutton Group
- Keyboard
- Small Opticator
- Large Opticator

(a) Functional Pushbutton Group

This is the group of 10 pushbuttons located above the two opticators. The following functions are provided:

- Selection of destination change type (MAN or AUTO)
- Selection of the Attack and Intercept submodes of Navigation (ATK or INT)
- Automatic data loader operation (FIDO)
- Manual Destination Change (CHANGE DEST)
- Storage of the coordinates of overflow points (STORE POS)
- Fixes (PLANFIX, RANDFIX)
- Selection of whether the NCU will be used for display of data or programming (entry) of data (DISPLAY/PROGRAM)

(b) Keyboard

This is a group of 12 pushbuttons. The lower halves of each pushbutton are used for entry of numerical and other data. The upper halves are used to select the type of data to be displayed or programmed.

(c) Small Opticator

This is a three "window" alphanumeric display used to give messages to the pilot concerning, for example, system status or OBCOS results.

(d) Large Opticator

The large opticator is used for display of the data requested by the pilot, for example present position, or to hold keyboard entered data during programming operations of the NCU.

8.2 NCU Display Example

Figure 6 shows the appearance of the NCU in the Display Select mode. As no data is to be entered, the numerical part of the keyboard is blanked while the small opticator shows that we are flying from waypoint 3 to waypoint 1 with the Doppler in memory. The top halves of the keyboard show the menu of displays available for selection.

Suppose that display of Present Position is desired. The pilot pushes POS and the NCU appearance becomes as shown in Figure 7 with Present Position shown in the large opticator. Of the available display selections, only POS remains illuminated as a reminder of the data being displayed. A further push of POS returns the NCU to display select mode.

8.3 NCU Program Example

From the display of, for example, Present Position, as shown in Figure 7, the pilot presses DISPLAY and the NCU appearance changes to the Program Select mode as illustrated in Figure 8. DISPLAY has changed to PROGRAM and the menu of programs available is indicated in the upper halves of the keyboard pushbuttons.

Suppose that the loading of waypoint coordinates is desired. The pilot presses DEST and the NCU appearance changes to that shown in Figure 9. Of the program menu, only DEST remains illuminated and the letter D has appeared in the small opticator as a reminder that the next task is the entry of the destination (or waypoint) number. The keys 0 and 1 are illuminated ready for the entry of the first digit of the waypoint number which must be a 0 or a 1.

After entry of the waypoint number, the NCU appearance changes to that shown in Figure 10 in which a waypoint number 06 has been entered.

The next task is to ACCEPT or REJECT this data by pushing the appropriate button.

If accepted, the next task is to load the coordinates of waypoint 6 as shown in Figure 11 following which a further ACCEPT completes the entry of the waypoint number and its coordinates.

(Note: The REJECT key is red when illuminated on the real NCU and, therefore, may not appear to be illuminated in Figures 10 and 11 because of reproduction in black and white).

8.4 Small Opticator Middle Window Indication Examples

Section 8.3 has given one example of how the small opticator is used during data loading using the NCU.

Illustrated in Figure 12 are some examples of the appearance of the middle "window" of the small opticator when it displays system status during normal flight - that is, when the NCU is not in Program mode.

The referenced figure is self-explanatory except for the example given at the bottom which indicates that we are flying from waypoint 3 to waypoint 2 with the Doppler currently in memory.

8.5 Small Opticator OBCOS Indications

The appearance of the small opticator during self-test of the subsystem selected using the OBCOS control located on the PCU is shown in Figure 13.

9. TESTING OF THE NAV/ATTACK SYSTEM

This section of the paper describes briefly the philosophy of integration and flight testing of the overall prototype Nav/Attack system.

The testing comprises six phases as follows:

Phase 1

Each subsystem was acceptance tested before leaving its factory for delivery to the systems integration and test rig site.

Phase 2

Upon arrival at the rig site, each subsystem was subjected to an identical acceptance test using identical test equipment.

Phase 3

Progressive Integration Testing of the system on the rig against written test procedures. In the context of integration testing, the word "progressive" means that the subsystems were, where possible, first tested in pairs, then triplets etc.

Phase 4

Ground test of the system on the trials aircraft against a written ground test procedure.

Phase 5

Flight Test. The flight test phase was essentially qualitative in that virtually no performance measurements were taken. The objective was to prove that the system was functioning, but not in terms of accuracy measurements.

Phase 6

Flight Trials. This phase is quantitative and will determine the accuracies of the Navigation and Weapon aiming modes.

For the latter part of the Flight Test phase and for the Flight Trials phase, an Airborne Data Recorder (ADR) was installed on the aircraft to record from the data bus. The EU software was also modified to output special parameters onto the data bus for recording purposes. Since no instrumented range was available, other techniques had to be developed for accuracy testing. These techniques, allied with the ADR and its associated ground equipment, enable satisfactory flight trials analysis.

At the time of writing, the project is entering the Flight Trials phase.

10. CONCLUSION AND FUTURE TRENDS

In conclusion, this paper has described briefly a HUD/WAC/NAV system built to satisfy the requirements of an overseas customer. Integration of Navigation and Fire Control within a HUD has been shown in testing to be a viable concept. In particular, the NCU approach of using illuminated pushbuttons is an elegant solution to the problem of providing a complex navigation control and display unit in a limited space and of locating such a device where it belongs, "Up-Front". This is due, in particular, to the fact that no dedicated panel engravings are necessary.

For the future, we have to pose the question -

"Are we heading towards central computing with more and more functions, which have historically been separated, integrated into one subsystem?"

However, it seems likely that such integration should be limited to those system LRU's and sensors which are required to operate in a co-operative manner to achieve a given function and one will certainly not see a widespread integration crossing many different boundaries of redundancy, integrity and performance such as by integrating flight control, engine control, navigation, stores management, etc, all into a single computing complex, however all encompassing the theoretical integrity of such a complex might be. Not only would such a system be highly expensive because of the need to design every system up to the level of the most demanding, but it poses what are presently insoluble problems in the field of common mode failures.

Therefore, as a basic systems design technique, I do not believe that we are moving back towards the original central computer complex. On the other hand, in the area of aircraft weapon system retrofit, where one is adding one or two quite distinctive features to an existing aircraft, or integrating some of the existing functions within the aircraft, it seems possible to adopt the benefits of central computing systems whilst at the same time maintaining the existing levels of redundancy and integrity which are available within the basic aircraft to which these new functions are being added.

11. ACKNOWLEDGEMENTS

The author wishes to thank the Management of Marconi Avionics for permission to publish this paper.

Thanks are due also to a number of colleagues for their help in the preparation.

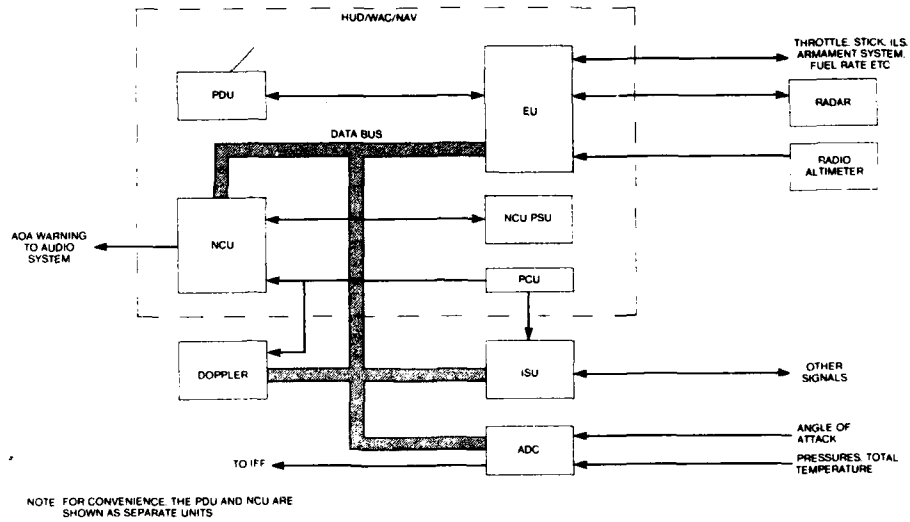


Figure 1 Simplified System Block Diagram



Figure 2 PDU (with NCU), EU and NCU PSU



Figure 3 PDU and NCU

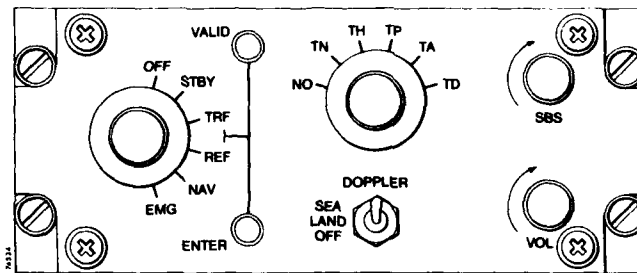


Figure 4 Pilot Control Unit

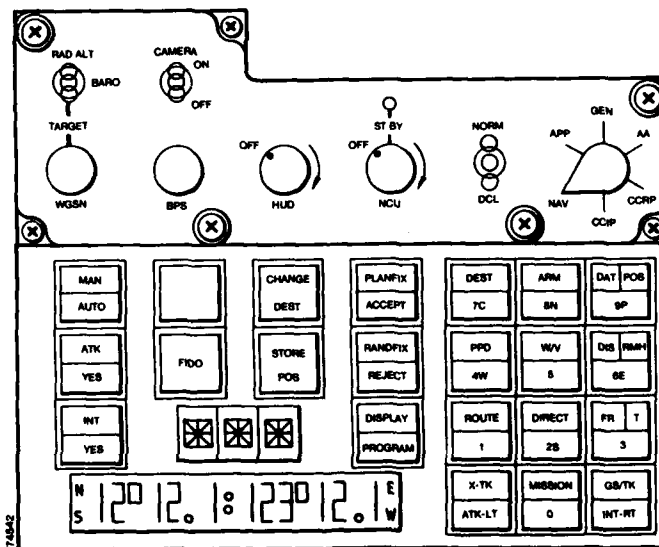


Figure 5 NCU showing all legends illuminated
(This cannot happen in practice)

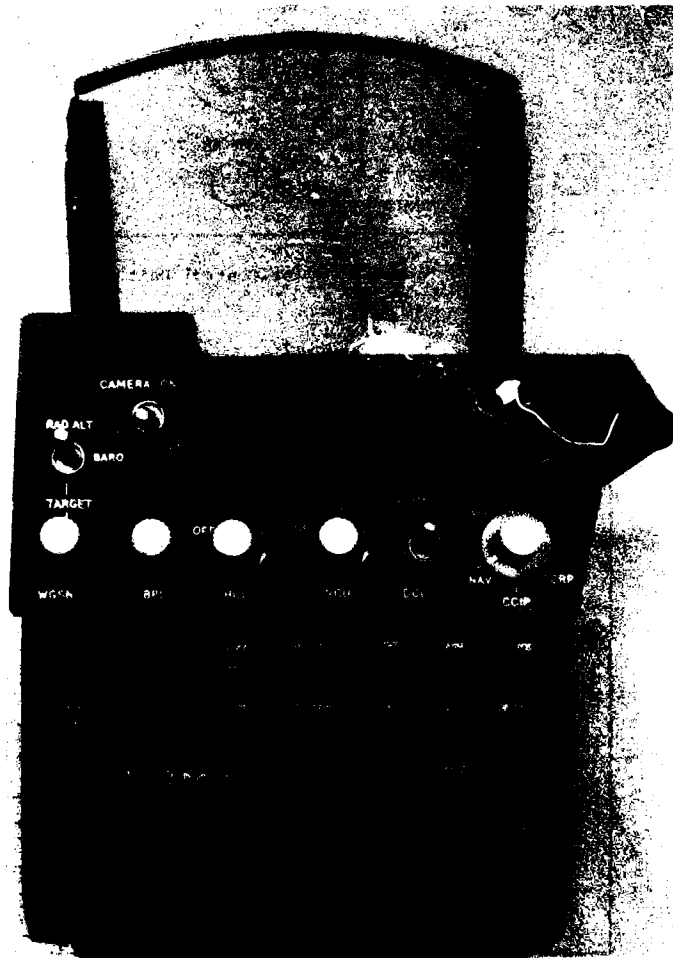


Figure 6 NCU Display Select Mode

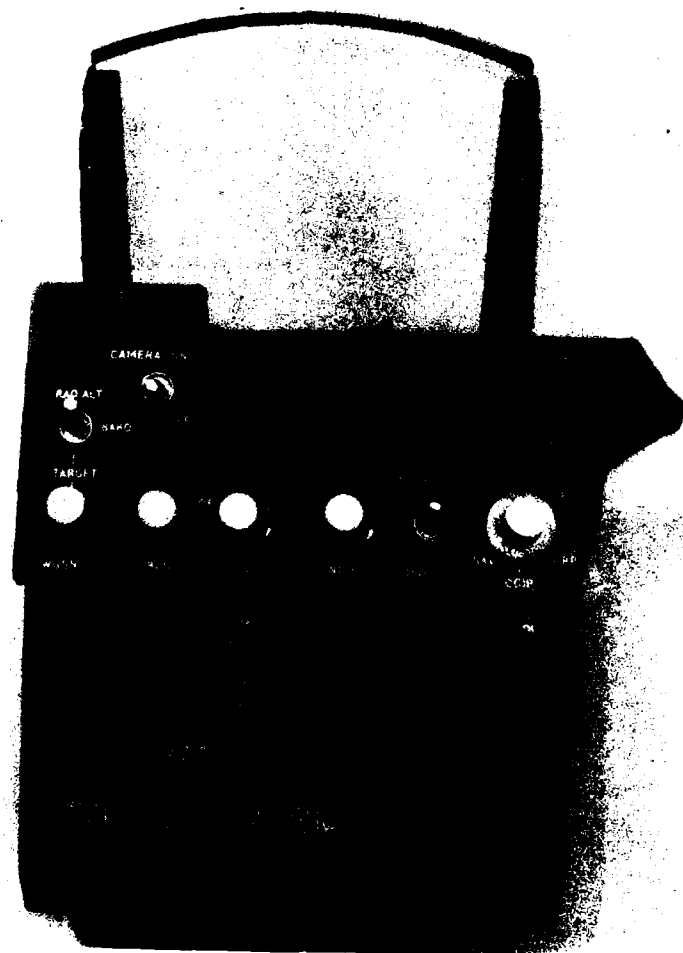


Figure 7 WCU Display of Present Position

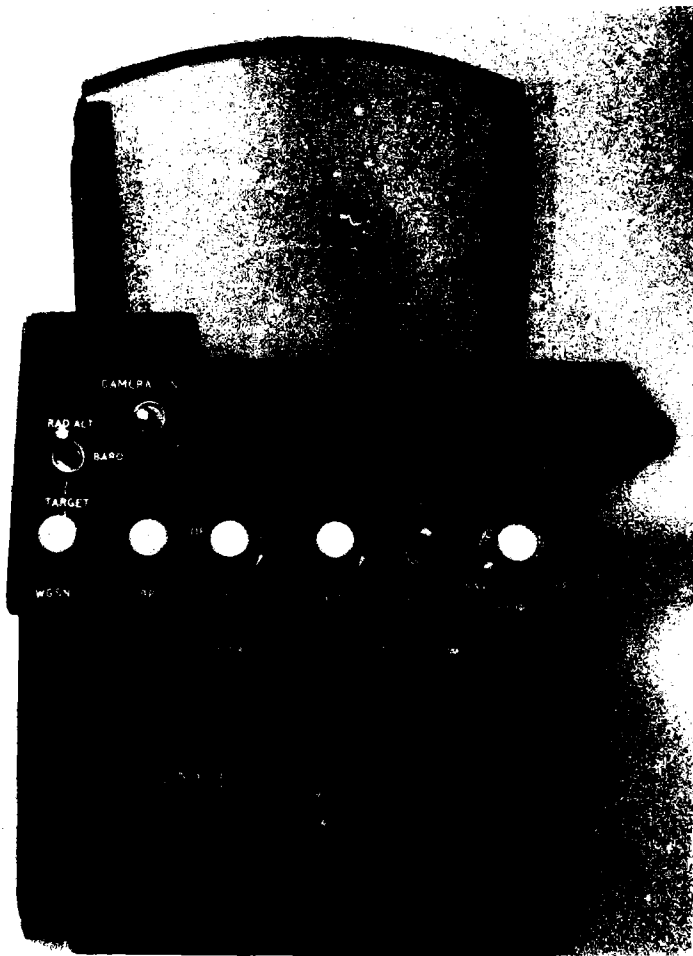


Figure 8 NCU Program Select Mode



Figure 9 WCU "DEST" Program



Figure 10 NCU "DEST" Program (continued)

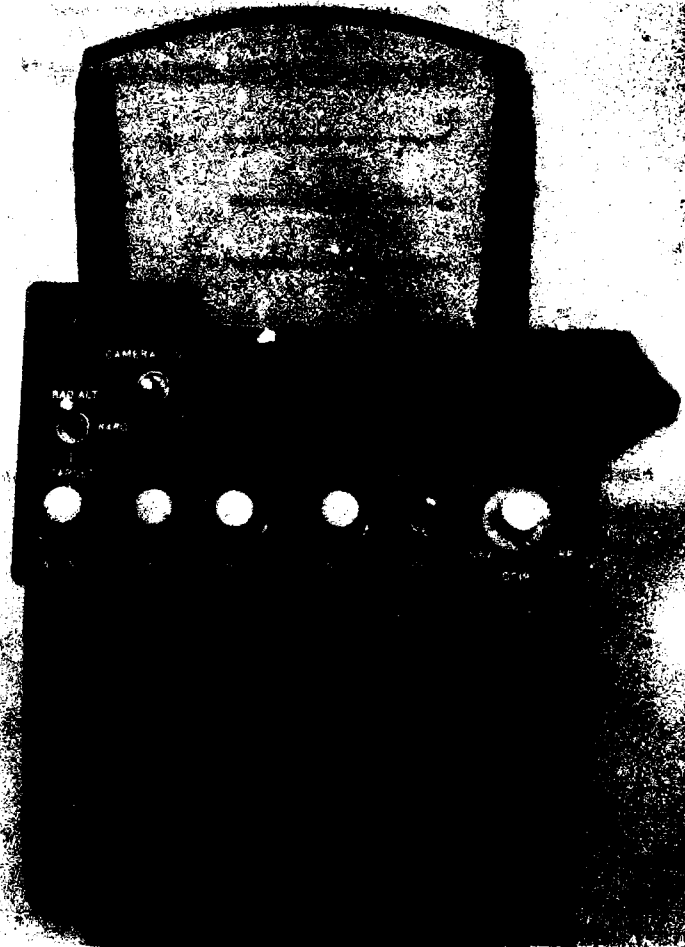


Figure 11 MCU "DEST" Program (continued)










-  DOPPLER/INERTIAL NAVIGATION IN PROGRESS
 -  PURE INERTIAL NAVIGATION IN PROGRESS BECAUSE DOPPLER IS IN MEMORY AFTER ABOUT 5 MINUTES, ISU SWITCHES TO BARO/INERTIAL MODE
 -  PURE INERTIAL NAVIGATION IN PROGRESS BECAUSE DOPPLER HAS FAILED
 -  PURE INERTIAL NAVIGATION IN PROGRESS BECAUSE DOPPLER NOT TRANSMITTING OR OFF. AFTER ABOUT 5 MINUTES, ISU SWITCHES TO BARO/INERTIAL MODE
 -  BARO/INERTIAL NAVIGATION IN PROGRESS
 -  HUD AIR DATA NAVIGATION IN PROGRESS
 -  AUTOMATIC NAVIGATION NO LONGER AVAILABLE
 -  HUD FAIL
- EXAMPLE 

Figure 12 Small Opticator Middle "Window" Indication Examples

FOLLOWING SELECTION OF A SUB-SYSTEM TEST USING PCU



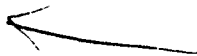
-  SUB-SYSTEM OK
-  SUB-SYSTEM FAIL

Figure 13 Small Opticator OBCOS Indications



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7-1

APPLICATION OF FLIGHT CONTROLS TECHNOLOGY TO
ENGINE CONTROL SYSTEMS

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SUMMARY

The transition from mechanical to redundant digital electronic flight controls started in 1950 and is at a rather advanced state. The first few full authority digital electronic engine control systems have completed initial development. Many of the control laws and redundancy management techniques that were pioneered in flight controls are directly applicable to digital engine controls.

Section 1 outlines the authors' recommended system design approach. Section 2 discusses some considerations for the design of electronically implemented control laws. Section 3 describes a typical redundancy management concept, based on digital flight control techniques, for a quadruplex engine control system. This is followed by the authors' recommended analysis methodology to determine the probability of control system failure. The conclusion of Section 3 discusses some implications that use the probability theory has on the control system design requirements. Section 4 discusses the recommended integration, development test and validation test approach of the authors.

1 GENERAL CONTROL SYSTEM DESIGN PHILOSOPHY

1.1 Engine Control System History

The first turbine engine became operational less than 50 years ago. Comparing today's sophisticated hydromechanical turbine control system to the comparatively simple turbine control of the first engine shows the tremendous progress made in the turbine controls field. This progress, in our opinion, will be overshadowed by the progress that will be made in the next three to six years with the new digital engine controls.

A turbine engine is a device that, unless controlled in an exact manner, has the tendency to self-destruct. Control must be obtained in a reliable, efficient and cost effective manner. In many ways this is similar to the control of an unstable aircraft with which the authors are more familiar than turbine engine controls. However, the application of digital control techniques and redundancy management concepts demonstrated in flight controls, as well as the lessons learned in flight controls, are directly applicable to engine controls.

1.2 Hydromechanical Control System Design Priorities

To adequately control a turbine engine is always a difficult job. This was especially true in the early days of turbine engine control work. There was little analytical basis for the control work and experimentation formed much of the basis for early control systems. Out of these experiences came the control system priorities that still hold true for today's hydromechanical controls:

1. Engine protection against self-inflicted damage for self-destruction from:
 - a. Temperature limits being exceeded
 - b. Speed limits being exceeded
 - c. Pressure limit being exceeded
2. Engine stability is lost as expressed in:
 - a. Engine thrust fluctuations
 - b. Fan and compressor stall margins
 - c. Augmentor spikes
3. Compatibility with aircraft inlet to satisfy engine requirements for:
 - a. Airflow corridor
 - b. Minimum burner pressure
4. Steady-State performance and accuracy of engine with regard to:
 - a. Thrust modulation
 - b. Thrust and fuel consumption requirements
 - c. Control sensitivity
 - d. Repeatability
5. Transient requirements of total engine to assure:
 - a. Thrust is a monotonic function of time for monotonic throttle command
 - b. Acceleration and deceleration times are reasonable and repeatable
 - c. Combustion stability is always present
6. Reliable start and transition to normal operation is available
7. Engine maintenance is minimum and can be performed in a reasonable manner.

It is the authors' opinion that prioritizing digital electronic control system requirements in this manner can be detrimental to a successful digital electronic control system. It is expected that proper use of the digital processors (with the necessary help of advanced multi-variable controls theory) will make it possible to meet all these criteria without necessarily assigning a priority order and without sacrificing overall safety or performance at any point in the flight envelope.

1.3 Control System Design Procedure

The recommended design procedure is shown schematically in Figure 1.1, Control System Design Integration and Validation Flow Diagram. It describes the method used by the authors in the design of flight control systems. It is felt that this method would be applicable to engine control in the same manner as to flight controls.

1.3.1 Concept Development

Any control system has three major requirements: The first is the functional requirements which relate to the plant that must be controlled. The second is the environmental considerations which must be specified to ensure the survivability of the control system. The third is the related probability of operation of the control system which is especially applicable if there are redundant control modes.

The functional control law characteristics determine the engine efficiency, utility, response characteristic and performance. These characteristics are obtained by synthesizing the control laws and are validated by analysis in the design phase and by testing prior to actual use on the engine. The environmental characteristics specified under which the control system components must survive are determined by the location of the components on the engine and/or within the airplane. These considerations influence the basic selection of the control system components that can be used. They also affect the overall control system architecture.

Probability of engine operation requirements are specified as the allowable probability of loss of engine, probability of allowable mission abort, and probability of damage to the engine due to engine control system failure. The specified requirements can be used by the control system designer to determine the required fault tolerant approach to meet the design goals for a fly-by-wire engine control system. Redundancy considerations, in conjunction with the reliability of the individual subsystems, form the basis of the engine operational readiness, the probability of operation, the probability of mission abort, as well as maintenance and life cycle costs of the engine.

Applying probability numbers based on Mean Time Between Failure (MTBF) data of components and/or subsystems to determine probability of engine loss, etc., has severe implications on the requirements for Continuous Built In Test (CBIT) and Initiated Built In Test (IBIT) of the engine control system. The standard practice of using MTBF data in control system probability analysis subtly implies that prior to each flight the control system is completely checked to verify that no latent failures are present in the system. In addition, failures that occur and have provisions for failure isolation and system reconfiguration must be identified and the system must be reconfigured. These CBIT and IBIT test requirements must be identified and implemented as part of the fault tolerant design of the engine control system.

1.3.2 System Definition and Development

The combination of functional and probability of operation requirements results in the complete definition of the conceptual engine control system. The authors feel that it is prudent at this time that the initial Failure Modes and Effects Analysis (FMEA) and Single Point Failure Analysis (SPFA) be conducted to determine if there are any failure modes within the control system that have not been properly addressed. Merely conducting a single point failure analysis and a failure modes and effects analysis is not sufficient since it only identifies potential problem areas and potential solutions. The potential solutions must be validated through testing on the actual engine control hardware to assure that the conclusions derived are correct. Therefore it is important that the testing facility be available that has the ability to validate single point failure analysis results and failure modes and effects analysis results.

The functional analysis results must be validated in a similar manner - preferably on the same controls test stand. The method proposed here for functional and redundancy management verification is to construct an engine controls test stand that will allow validation of control systems characteristics. The definition of such a test stand must be considered in the conceptual engine control system design. The validation of the control system operation as influenced by environmental characteristics is done as part of the component testing and is not considered in the test stand design.

After the conceptual design is completed, the system must be separated into individual line replaceable units (LRUs) must be done. The line replaceable units are computers, actuators, sensors, etc., each requiring its own dedicated specification. Satisfactory results from the control system analysis, the FMEA and the SPFA assure that the specification is ready for vendor selection and contract negotiations. As part of the contract award negotiations, a definition of LRU development testing, acceptance testing, environmental testing, reliability development, and life testing must be defined and made a part of the contract.

1.3.3 Verification, Validation, and Integration Testing

Based on the authors' experience in flight control systems, it seems almost impossible to integrate a high authority digital engine control system or flight control system without the use of a control system test stand. A generalized block diagram of such a control test stand is shown in Figure 1.2. A control test stand is used to integrate the control system hardware, perform functional validation of the control laws and redundancy management implementation of the engine control system design.

Experience in flight control system has shown that it is extremely important to perform formal testing with well-documented and rigidly controlled test procedures. It is helpful to perform some development testing prior to the official tests in order to assure basic system operation and to allow identification and resolution of early fundamental operation problems. Close cooperation is required between the selected LRU vendors and the buyer to assure that the vendor's development test procedures, acceptance test procedures, environmental test procedures, life test procedures, and the buyer's system integration test procedures, development test procedures and validation test procedures are properly coordinated. This coordination is especially important for the design of the integration test facility at the buyer's facility, which places special requirements on many of the individual LRUs. These special requirements will require extra features of the lab test units to allow visibility into internal operation of LRUs to aid in problem identification and debugging.

The test stand facility must serve the purpose of conducting preliminary trial runs of tests to be conducted at a later date on the engine in the test cell or on the aircraft. Time and money should, therefore, be saved in the test cell and aircraft tests by debugging the test plans and procedures and by allowing the test cell and aircraft tests to be checks of critical points and a verification of previous results.

Satisfactory completion of all testing on the engine controls test stand should be a prerequisite for the control systems testing in the engine test cell. If a good representation of the thermodynamic engine was used in the closing of the control loops on the controls test stand, the test results of the engine test cell should be identical to the results obtained on the test stand. This in turn implies that the test cell operation of the engine control system and the total engine operation is a validation of the previously obtained engine controls test stand results. Proper use of the controls test stand therefore allows the inexpensive operational use of the engine controls test stand to be substituted for many hours of expensive test cell runs.

The test procedures used and validated on the test stand can also be used on the engine test cell operation and engine installation testing on the aircraft. Again, this will lead to further cost and time savings in the final phases of the program when time is generally at a premium and delay costs are high.

2. FUNCTIONAL TURBINE ENGINE CONTROL LAWS

This section briefly reviews typical current Hydro-Mechanical control law development for turbine engine and gives a general example of a hydro-mechanical control law. This will be followed by a discussion of expected parallels between turbine engine control and flight control system developments.

2.1 Engine Model for Control Law Discussions

The turbine engine model selected for detail discussion of the control laws is shown in Figure 2.1. It is an afterburning, two stage turbine engine with the sensors and actuation system controls identified. The engine is typical of those used in current generation fighters.

2.2 Typical Hydro-Mechanical Control Law Concept

Current turbine engine control systems schedule the inlet fan vane guides and high pressure compressor vane guides as a function of component corrected speeds. These schedules are derived from component tests and are designed to maximize component efficiencies along the operating lines which will assure adequate stall margins for even the most severe operating conditions anticipated.

The fuel control is similarly designed to schedule fuel flow as a function of RPM, engine pressures, engine temperatures, static pressure, free air temperature and Mach number to assure that none of the operating limits of the turbine engine will be exceeded and the required operational characteristics are maintained.

The general approach used in hydro-mechanical controls is to use a "predictor" control law that predicts in advance the desired vane guide position, nozzle position or fuel flow desired based on pressure, temperature or other sensor signals.

In brief, this design approach is well understood by the engine controls community and has a large data base to support its use. The hydro-mechanical control computers development in support of this control concept are an outstanding engineering achievement representing a mature technology.

2.3 Recommendations for Digital Electronic Turbine Control Laws

The potential performance improvements obtainable by modern electronic control systems in increased thrust over the flight envelope, improved Specific Fuel Consumption (SFC), improved probability of Mission Success, lower probability of engine damage and engine loss and the promise of lower development, acquisition, operating and life cycle cost will, in the authors' opinion, make the current approach obsolete.

However, digital electronic computers show only small advantages when incorporating conventional engine control laws into a digital hardware/software architecture. This can be vastly improved by adding an electronic control law concept which also uses command position (or fuel flow) to drive the plant to be

controlled to desired operating conditions and is implemented by following several guidelines used in the development of digital flight control systems. The following guidelines should be observed:

1. The design approach should lend itself to small amplitude - piecewise linear systems analysis using either state variable or complex plane techniques.
2. The limit functions should be excluded from the control law analysis. The inclusion of limit functions prohibits initial use of linear analysis, which is a very powerful tool for validating the initial design concept. Table 2.1 shows a list of 38 limit functions required for a generalized two spool engine. Any or all of these limits can be functions of multiple variables used or not used in the primary engine control laws.
3. The linear analysis should be followed by time domain non-linear analysis to evaluate the effects of limit functions and other nonlinearities.
4. Actuator control loops should be designed to be single command/feedback functions with only one summing junction in the loop structure. Reduction of the actuation system(s) to first order, second order or third order linear system must be possible.

2.4 Recommended Interface Between the Hydro-Mechanical and Digital Flight Control Subsystems

Designing control laws based on the preceding recommendations is expected to produce superior performance over the hydro-mechanical system. However, since the new control laws and mechanization have not been proven to date, it is advantageous to build the initial systems such, that the control laws of the electronic system are in control, but the hydro-mechanical control system is also active and ready to take over should the electrical system perform unsatisfactorily or even fail. By the same token, it must be possible to disconnect the hydro-mechanical system if that portion of the system should fail. This switching concept is shown in Figure 2.2. For the case shown, the inlet Fan Vane control and HPC Vane control is spring-loaded to the hydro-mechanical control. By applying hydraulic pressure to the transfer valve the electrical control mode is selected. There are independent loop closures for the electrical and hydro-mechanical actuation system. Through the use of the transfer valve, either loop closure can be selected.

3. REDUNDANCY MANAGEMENT CONSIDERATION

This section first describes a typical redundancy management concept used in quadruplex digital control systems. This is followed by an example of probability analysis applied to such a quadruplex control system. The conclusion of this chapter discusses the authors' approach to probability of operation specification for a control system, some implications that use of probability theory imposes on the system design and the definition of the authors' "cost effective" design goals.

3.1 Typical Redundancy Management Concepts for Quadruplex Digital Processor Systems

One of the desirable features of a digital processor is its capability to process large amounts of data, perform complex computations, and make complex logic decisions in an efficient manner. The undesirable part about the digital processor is its failure modes. First, these failure modes can be catastrophic at any time without prior warning. This is in contrast to a mechanical system where failures are normally preceded by detectable wear, leaks, or degraded performance. Second, there are many failure modes that allow basic operation of the processor, but do not allow performing the computations and the desired logic decisions in the manner required by the system design.

3.1.1 Built in Test Concept

The problem of the processor operating but coming up with wrong solutions can be handled by using a well-structured and dedicated method of testing. One should realize that a unique feature of the digital processor is its capability of determining if the last set of computations performed by the processor were performed correctly. This is accomplished by a testing method commonly called built-in tests (BIT). BIT can be separated into two parts; continuous BIT (CBIT), which is run any time the functional algorithms are being executed, and initiated BIT (IBIT), which is run when power is first turned on or when selected by ground crew or pilot.

The concept of continuous BIT and initiated BIT must be an integral part of the control system design. It is not a feature that can be added on at a later time, but must be considered in the initial system design. The hierarchy of a redundancy management system starts with the digital processors. Once the processors' operation has been verified all the interfaces between the digital processors and between each processor and the outside world are tested. After that testing has been completed satisfactorily, testing is extended to the sensors and their interfaces, the actuators and their interfaces, and also the discrete signals and their interfaces.

The continuous BIT is running continuously and checks the normal operation of the control system. The initiated BIT checks, in addition to the normal operation, such items as failure detection monitoring circuits, control system reconfiguration circuits, and display interfaces. Failure to pass these tests leads in general to the disabling of the affected system portion(s).

A generally accepted philosophy is that a digital processor can declare only itself as failed. It is considered imprudent to allow the other processors to declare a particular processor failed. Considering the amount of self-testing that is possible in a digital processor, the authors feel that this is a proper decision.

A typical series of tests conducted by the digital processor to assure its operational status is:

1. A watchdog type monitor in the computer hardware that is of the "dead man" switch type.
2. A PROM memory test that performs a check sum of the memory and compares it against a stored number indicating the correct check sum.
3. A separate computation involving all the microcode instructions of each individual processor. The correct results of the computation(s) is stored and compared to the actual results obtained.
4. A parity check that verifies the validity of data previously stored prior to its use.
5. A pattern check* of the RAM memory.
6. A pattern check* of the digital processor bus system and the registers.
7. A check of the power supply within the control system.

* NOTE:

A pattern check consists of writing all zeros into the desired locations and then reading the information back out to assure that the input and output information is identical. This process is repeated with all ones, alternate zeros and ones, alternate ones and zeros, alternate zeros and ones in pairs, and alternate ones and zeros in pairs.

If a processor can perform all the indicated functions for continuous BIT correctly, the processor can be assumed to be working with a very high degree of confidence. Once the processor is operating satisfactorily, the next step is to test the interfaces from the digital processor to the outside world. These interfaces are:

1. The analog to digital and digital to analog converter
2. The discrete input and output signal interface system
3. The intercommunication systems between digital processors typically called the cross channel data links.
4. The MIL-STD-1553 bus if such a device is part of the system.

3.1.2 Built in Test Applications

The analog to digital (A to D) and digital to analog (D to A) converter continuous built-in test is best described with reference to Figure 3.1. This test uses a dedicated wrap-around test loop and the previously described test patterns that were used in the RAM memory testing. The digital processor uses the test pattern, one at a time, and sends the data out through the D to A converter. The wrap-around test loop is activated and the analog data transfers directly through a buffer amplifier to the analog to digital converter. The analog to digital converter receives the digitized data and verifies that the data is within tolerance. Typical tolerance is ± 2 least significant bits. By using all test patterns and obtaining satisfactory results the normal operation of the A to D and D to A converter is assured.

The discrete signal interface system is best described with reference to Figure 3.2. This system also uses a wrap-around test similar to the analog to digital and digital to analog converter wrap-around loop. The discrete signals coming into the digital processor are stored in a latch buffer and are coming in through the parallel interface. From there the signals are sent to the digital processor via the digital processor bus. The output discretizes are sent via the digital processor bus to the output latch buffer. From the output latch buffer the discrete signals are transferred in the parallel mode to the discrete hold circuits. The hold circuits are the refresher type that require periodic updating for the discretizes to stay in the ON state. This assures that if the digital processor stops functioning the discrete signals will go towards the OFF state typically within two to four samples. The wrap-around loop uses the serial interface of the shift registers. The output latch buffers are wrapped around to the input latch buffers and the testing is similar to the analog to digital converter test. The digital processor uses the test patterns previously discussed and feeds the data to the output latch buffer in the parallel manner. From there the data is transferred to the input latch buffers using the serial link between the input and output latch buffers. The digital processor receives the data from the input latch buffer and compares them for bit identical patterns. Using all test patterns, one pattern at a time, and obtaining good results assures normal operation of the discrete interface. The experience of the authors is that the discrete interface is a much more cumbersome and difficult task for design, test, and validation than the analog signal interface.

The cross channel data link interface is best described with reference to Figure 3.3. This interface is the communication link between the different digital processors of the engine control system. This test uses again the previously discussed test patterns where each of the digital processors sends the test patterns to the other processors, all working time synchronous. The digital processors start their computational cycle, which typically varies from 10 to 20 milliseconds, at the same instant. Since all processors start their task of continuous BIT simultaneously and all processors know what data is to be received since they are also transmitting the same data, each processor can check the data received from the other processors and verify normal operations. Receiving all test patterns from the other processors in the bit identical manner assures normal operation of the cross channel data link.

Many of the digital processor interfaces to the outside world have their own dedicated RAM memory. This RAM memory requires a separate test, identical to the test for the previously discussed processor RAM memory, which was described at the beginning of this section.

The testing of the MIL-STD-1553 bus system is specified as part of MIL-STD-1553. The only recommendations that the authors have is that if the dual bus which is specified in the MIL-STD is used, the system should alternate between the A bus and the B bus. By using the buses in an alternate manner, a failed bus is detected at the time the first failure occurs and a failed system is not carried along without knowing that a failure has occurred. Using the same bus until a failure occurs can result in having had a failure in the bus system not being used and when trying to use the second part finding that the system had previously failed.

3.2 Typical Redundancy Management Concepts for Signals Interfacing With A Quadruplex Digital Control System

3.2.1 Input Sensor Management

This section discusses the failure detection of redundant input sensor signals. In case of engine controls, typical input signals are pressure signals, RPM signals, and temperature signals. Pilot command signals and actuator position signals also fall into this category. The testing is best described with reference to Figure 3.4 which shows the data flow for the testing. One capability found in the digital systems that is not available in analog systems is the use of separate criteria for selecting the signals to be used for computation and detecting if a signal is failed. Assuming the quadruplex channel system, shown in Figure 3.4, each of the four processors receives all four sensor signals, one from its own A to D converter and the other three from the cross channel data link. The typical procedure is to perform a reasonableness test which also produces some filtering on the data.

This filtering is helpful in determining if the local channels sensor signal is within normal operating limits relative to the other channel sensor signals. The standard procedure is that each processor can only declare its own signal failed. The criteria typically used is that each sensor must be different (by more than the failure threshold) from all three sensor signals in the same polarity direction. It is not unusual to allow a signal to indicate that it has failed for several consecutive samples before the sensor is officially declared failed. Such an occurrence is stored as failure indication data. That data should be displayed at the post flight system check for maintenance action. If the signal is not within tolerance, it is not used as part of the signal selection algorithm that determines the sensor signal value to be used in the control computations.

The selected signal for computation is typically the average of the middle values if all four signals of the previous computation have been declared good; it is the middle value if there are three good signals; and it is the lower, nonzero value, if there are only two good signals remaining. The lower nonzero value is chosen to prevent hardover signals and signals failed to zero from being selected.

Discrete signal failures are detected through a time relationship. Each digital processor receives all four signals (either through its own interface of through the cross channel data links) and determines the good/failed status of each signal and the switching action that should take place in the central system.

The closing or opening action of switch contacts on a multipole switch typically do not occur simultaneously. Consequently it is possible that there is a substantial time difference between the status change of the first set of contacts and the last set of contacts on a multipole switch in response to a typical command. The only exception is switches incorporating a mechanical overcenter device that makes the switch assembly into a bistable device. The bistable switch assembly is recommended for use with redundant digital systems.

Assuming that a switching action is initiated by some component within the engine control system, actual switching in the processor occurs after a majority of the good signals have changed state. For example, if there are four good signals and three of the four signals have changed state, the processor assumes that the switching command is correct and will perform the switching indicated. Discrete signals have a designated time limit, a typical number is 150 milliseconds, to complete switching. If the fourth signal in the previous example completes switching prior to the switching time limit expiring, no failure is declared. If the fourth signal does not switch prior to the time limit expiring, the processor will declare the fourth signal failed. The same procedure is used if there are only three or two good signals.

One should note that in discrete signals it is quite possible to have two simultaneous failures, making it impossible for the processor to isolate the failed signals. It is therefore important to define for each type of discrete signal the preferred state to which the processor will revert the system if it is no longer possible to determine what the discrete status is. In some cases, the preferred state is a conditional situation depending on other variables available.

3.2.2 Actuator Management

The remaining failure detection concept is the actuation system failure identification. This is the most difficult concept and also has the most conceptual variation. The authors' preferred concept is the electro-hydraulic servo valve with multiple coils. This concept is shown in Figure 3.5. The advantage of the multiple coil concept is that it allows interface of multiple electrical channels to one actuator without violating the electrical isolation requirements. Most engine control systems use the single-fluid actuator, which is substantially simpler to control and monitor than the dual fluid system typically used in flight controls. Since position accuracies of ± 2 percent are acceptable for engine control systems, the electrical command, mechanical feedback concept is recommended. This, in turn, will lead to the simplified in-line monitoring concept shown in Figure 3.5. The in-line monitoring concept duplicates the drive circuit used on the primary actuator for monitoring and compares the drive currents from the model drive circuit to the actual circuit. Failure identification of each circuit is nearly instantaneous and thereby prevents large failure transients. This actuation system is three-fail operate electrically, but since there is only one EHV and only one fluid system, there is no fail operational capability in the fluidic portion.

There is an additional method of failure detection available for engine control systems. This concept is based on real time modeling of the thermodynamic engine in the onboard digital processor. This concept is shown in Figure 3.6. Such a real time model was developed by General Electric and demonstrated as part of the FADEC program. The results are published in a SAE paper (Reference 1). This paper describes a real time modeling concept of the thermodynamic engine which provides all sensor signals of the engine. This model can be carried in the digital processor controlling the engine and the sensor outputs from the model can be used to control the engine. By using Kalman filtering techniques to make the engine and sensor signals track together, one can determine failures of the sensors and continue to operate the engine in the presence of sensor failures, even if there is only one digital processor in the engine control system.

3.3 Computing Probability of Engine Operation

There are several possible approaches to determine the probability of operation of an engine control system. A typical approach is to determine the for each component and the probability of operation is the sum of all the individual 's. This approach is not suitable for redundant fly-by-wire type control systems. This is mostly due to the fact that there is a high dependency and interrelationship in redundant control systems that defeats the summation technique.

For the probability of operation analysis of redundant electronic control systems the probability block diagram modeling approach is recommended. This requires the construction of a block diagram representing the total control system. The total control system consists of the electrical power circuits, the fluidic power systems, the electronic system segments, the wiring of the control system, the sensors and the actuators of the control system. The basis for such a diagram is shown in Figure 3.7 for a generalized control system. The solid lines represent a single digital processor system. Adding the dotted lines results in a dual digital channel processor system. The fluidic portions are single channel. Each of the components or blocks has its own MTBF and the probability of failure of that block is $1/MTBF$.

By adding the sensors, electrical fuel control circuits and electrical actuator control circuits to the output states defined by Figure 3.7 a probability of operation flow diagram can be constructed, as shown by Figure 3-8. The control system is operational if a continuous path exists from the left side of the flow diagram to the right side of the flow diagram. The individual branches of the flow diagram are connected by AND gates and OR gates.

Figure 3-8 illustrates probability of operation for a nonredundant control system. Being single channel, the system has poor probability of operation with the expected number of failures per million flight hours determined by adding the probability of failure (or) of each block.

Figure 3.9 represents a dual digital channel engine control system with the previously used single channel fluid systems. The outputs of Figure 3.7 are used as inputs to Figure 3.9 as in the previous example. The sensors, electronic fuel control circuits and electronic actuator control circuits are also dualized. The multiple EHV coil interface approach, previously discussed, was assumed. The resulting block diagram, shown in Figure 3.9, is substantially more complex than the previously shown single channel system. By inspection one can determine that the resulting control system can operate after the following failures:

1. One digital processor, or
2. One fuel control channel failure plus one Vane guide actuator channel failure plus one nozzle actuator channel failure, or
3. One electrical power system failure. All other failures lead to loss of engine control.

An additional level of redundancy is shown in Figure 3.10. In this configuration use is made of the Cross Channel Data Link (CCDL) that allows data exchange between the digital processors. The outputs of Figure 3.7 again provide the inputs to Figure 3.10. Even though the hardware complexity increased very little, the complexity of the block diagram in Figure 3.10 increased substantially over the previous two figures. The addition of the CCDL allows the control system to operate the actuation control circuits of channel 1 with the sensors of channel 2 and visa versa. In addition, the exchange of sensor signals allows better testing and failure isolation.

The systems shown in Figures 3.9 and 3.10 can no longer be analyzed by summing up the probability of failure of the individual blocks. In order to obtain the probability of the engine control system failure, one must write the probability equations of the block diagram and compute the probability of loss of engine control based on mission duration. The probability of aircraft loss increases exponentially with mission duration.

There are additional implications to the block diagrams in Figures 3.9 and 3.10. Each of the OR gates indicates redundant systems with either system having the capability of controlling the engine. By computing the probability of operation in the manner indicated, one makes implicitly the assumption that all failures are detected and the control system is reconfigured in a safe manner. In other words, a failure can not cause critical damage to the engine prior to system reconfiguration and the reconfiguration circuits can never fail. This would imply a coverage of 1 which is not possible. However, using good design practices and a good CBIT and IBIT concept, coverages above 0.95 are possible and are also satisfactory.

By using the recommended analysis approach, one can analyze systems of any complexity. Being able to analyze systems allows one to trade off different system redundancies and redundancy management concepts. Based on work performed in flight controls, the authors believe that it is possible to meet the "proposed requirements definition for redundancy management" postulated in Section 3.4.

3.4 Requirements Definition for Redundancy Management

The redundancy and fail-operational capability requirements of a control system must be developed with guidance from probability analysis. The basic design requirements for such an approach is to define the criteria for:

1. Allowable Probability of plant shutdown due to control system failure.
2. Allowable Probability of Mission Abort.
3. Design Mission Duration.
4. Probability of damage to the plant being controlled due to control system failures.

Using again Flight Control Technology as reference, the probability of loss of aircraft due to flight control system failures of a tactical fighter is typically less than 10^{-7} and the probability of mission abort due to flight control system failure must be less than 10^{-3} . Mechanical flight controls cannot be built to meet these criteria, but an electronic flight control system, designed to have the necessary redundancy and fail-operational capability, can meet the specified criteria.

The criteria for engine control system loss can be specified in the same manner as for flight controls. There might be a difference in loss of control criteria for single and multiple engine aircraft. The critical case is the single engine airplane and the criteria development in this paper will use a single engine tactical aircraft example. Considering that loss of engine control typically causes loss of the aircraft, the allowable probability of engine control system failure is less than 10^{-7} .

The second item that must be specified is the probability of mission abort due to engine control system failure. The probability of one mission abort per 10^3 missions used in the flight control system is considered applicable and will be used here.

The third item is the mission duration time. According to MIL-F-9490D, the applicable flight control system specification, the design mission is typically the longest mission that does not require mid-air refueling. The longest mission for a tactical fighter is the ferry mission and the specified mission time for this design is 2 hours.

Fourth, the allowable probability of damage to the engine due to the control system failures must be considered. Due to the high cost of engine repair, no failure within the engine control system is allowed to result in damage to the engine. The probability specified for engine damage must therefore be less than 10^{-7} which is the probability of total loss of engine control. The specified probability is 10^{-8} .

The design requirements specified so far are:

1. Allowable probability of engine failure due to engine control system failures is less than 10^{-7} or one failure in 10^7 flight hours.
2. Allowable probability of mission abort due to engine control system failure is less than 10^{-3} per mission. This can also be expressed as one mission abort every 10^3 missions.
3. The design mission duration is 2 hours.
4. Allowable probability of engine damage due to engine control system failure is less than 10^{-8} .

Based on these requirements and proper use of probability analysis, the redundancy management portion of the engine control system can be specified.

Missions time for probability operations are not necessarily the same functionly as mission time which is from take-off to landing. Mission time for probability computations is the time between complete system validations. This includes all functions rarely used, such as limits, reconfiguration circuits, failure detection circuits, sensors, actuators, etc. To make this definition mathematically complete, a coverage greater than .95 for all systems and component tests is specified as an acceptable system validation. It can be safely stated that a manual system check to 95 percent coverage is not possible. Such a test is typically automated as a combination of continuous BIT and preflight/postflight BIT with some manual tasks such as moving controls and switches done by the pilot. Such a test must be considered from the initial design to meet the specified requirements. One should note that the probability mission time approaches equipment installed time as coverage approaches zero.

4. CONTROL SYSTEM INTEGRATION, DEVELOPMENT TESTING, AND VALIDATION TESTING

A complex digital control system, of the nature discussed in this paper, must be integrated and validated in a closed loop environment. A closed loop environment means, in this case, that the thermodynamic engine characteristics are being modeled by a digital processor and that the input variables (such as flow, fan position, vane guide positions, and exhaust nozzle area) are used as input to the thermodynamic engine model. The outputs of the thermodynamic engine model are the sensor inputs to the engine control system. Having a complete engine control system, consisting of the actual hardware and software, not a simulation of it, integrated with the thermodynamic loop closure as previously described, allows one to simulate engine operation of the total engine in its final configuration. In the

design of the control system (prior to specification release for the individual LRUs), one must make provisions so that all inputs to the thermodynamic engine model are available as electrical signals, and to make provisions so that all inputs from the thermodynamic engine model to the engine control system can be entered in parallel with the actual sensors. An alternate to this approach is a sensor input simulation so that the output of the actual sensor can be used as input to the engine control system.

Testing to be performed on such engine controls test stands consists of the following:

1. Integration of all control system components and their functional evaluation.
2. Engine controllability evaluation over the flight envelope. (It should be pointed out that the controls test bench testing is rather inexpensive and that time can be spent cost effectively verifying all aspects of engine operations.)
3. Validation of the failure modes and effects analysis and the single point failure analysis.
4. Dynamic thrust response evaluation of the simulated engine and engine control system combination.

By performing the testing in a rather rigorous and extensive manner many problems typically discovered in the flight test program can be discovered on the engine test stand. The prerequisite for such a statement is that the thermodynamic engine model, which is contained as mathematical equation in a digital processor, has the necessary fidelity. Such models can be formed as is indicated in Reference 1.

The authors' design for an engine controls test stand is shown as block diagram in Figure 4.1. Assuming that the engine test stand is used by the engine designer, it must have the capability of simulating typical aircraft acceleration and deceleration profiles, typical climb and descent profiles, and operation at any constant Mach and altitude point. At any of these points in the flight envelope, the engine must be able to accel/dec, simulate air starts or perform any other required functions.

It is the authors' opinion that all testing to be done in the engine test cell should also be done first on the engine controls test stand. This assures that the time spent in the test cell is used for maximum benefit since all test procedures have been previously validated. One of the experiences frequently encountered is that the first set of test instructions shows definite weaknesses and have to be corrected prior to use on the actual aircraft.

Another benefit of the controls test stand is the total validation of the continuous and initiated built-in tests. It is extremely difficult to finalize such concepts on the actual engine. The capability of stopping the simulation at any time, investigating the individual conditions existing at that point, and continuing on after all the errors have been corrected and unexpected indications are understood, is of immense value. This type of testing is not only of value at the initial development of the engine control system, but also during engine tests and flight test programs. One can simulate engine abnormalities and thereby gain a better insight into problems occurring in testing that cannot be analyzed in such detail in an engine test cell or in a flight test program.

The authors also envision use of engine control test stands by the airframe manufacturer. Such a test stand will probably become part of the control system development test stand. If a complex fly-by-wire control system or integrated flight and propulsion controls test stand is used on a particular airplane, the interaction between engine, electrical power system, hydraulic power system and the flight control system can be validated. By having all these subsystems functioning together on the same controls test stand, interactions of these subsystems can be analyzed if there are problem areas; corrections can be found and implemented.

An additional area of concern for the airframe contractor is the engine inlet and the airframe interaction. By modeling the inlet in the proper fashion and by using the controls test stand as a flying simulator, the excursions of alpha and beta (which are typically critical to the inlet and airframe integration) can be investigated. In considering such an investigation one must realize that the maximum alpha and beta excursion which are of interest typically occur during failures in the control system or are due to gust disturbances on the aircraft. By obtaining the alpha and beta excursions due to control system failures and simultaneously simulating the effect on engine performance, one has the capability to verify if there is a problem and test possible solutions to the problem. The same is true for disturbances due to gusts.

5. CONCLUSION

This paper has presented the control system design technique used by the authors in the development of electronic flight controls translated to the development of engine controls. Valuable lessons have been learned in the past decade in the development of fault tolerant, flight critical flight controls. Through this experience a process of linear/nonlinear analysis, simulation, and hardware integration testing has developed. Structuring the control system design, documentation and testing as recommended in this paper offers the hope of leading to control systems that function as desired, have no hidden problems, and can be developed in a cost-effective manner.

6. RESUMES

R. Seemann

Mr. Seemann was born in Stuttgart, West Germany and emigrated to the United States in 1955. He has a B.S. in Electronic Engineering from Northrop University and a M.S. in Electronic Engineering from California State University, Long Beach.

He is presently the Director, Technical Staff of the Vice President of Engineering, Northrop Aircraft Division. Associated with Northrop for 17 years, Mr. Seemann has worked on such diverse flight control system applications as the F-5 aircraft, the M2-F2 and HL-10 NASA lifting bodies, the X-14B NASA VSTOL inflight simulator, and the YF-17 and F-18 control systems. Mr. Seemann became associated with engine controls technology when working on advanced aircraft designs.

Technical publications include papers on the X-14B control system, the 4 coil electronic interface concept used on the F/A-18 aircraft and on advanced fly-by-wire actuation systems.

J. L. Lockenour

Mr. Lockenour's career in aerospace began in 1967 upon his graduation from Purdue University with a B.S. in Aerospace Engineering. Initially he worked for the Rockwell Corporation, Columbus Division, in aircraft flying qualities. At Rockwell he was involved in criteria development, piloted simulation studies, and preliminary design. During this period he obtained an M.S. degree from Ohio State University in automatic controls. From 1970 to 1975 he was employed by the USAF Flight Dynamics Laboratory in its Flight Control Division. There he was responsible for the Variable Stability NT-33A research studies and studies to revise the Flying Qualities Specification MIL-F-8785B. Starting in 1975, Mr. Lockenour was Project Manager of the HIMAT Remotely Piloted Research Program at the NASA Dryden Flight Research Center. In 1978 he moved to the Northrop Aircraft Division. Currently, Mr. Lockenour is Manager of the Flight Control Technology Department. In this capacity he is responsible for all research, advanced design, and project activity in the areas of control law development, flying qualities, and flight control system development.

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- (3) French, Michael W., "Development of a Compact Real-Time Turbofan Engine Dynamic Simulation," SAE Technical Paper 821401, General Electric Aircraft Engine Business Group, October 1982.

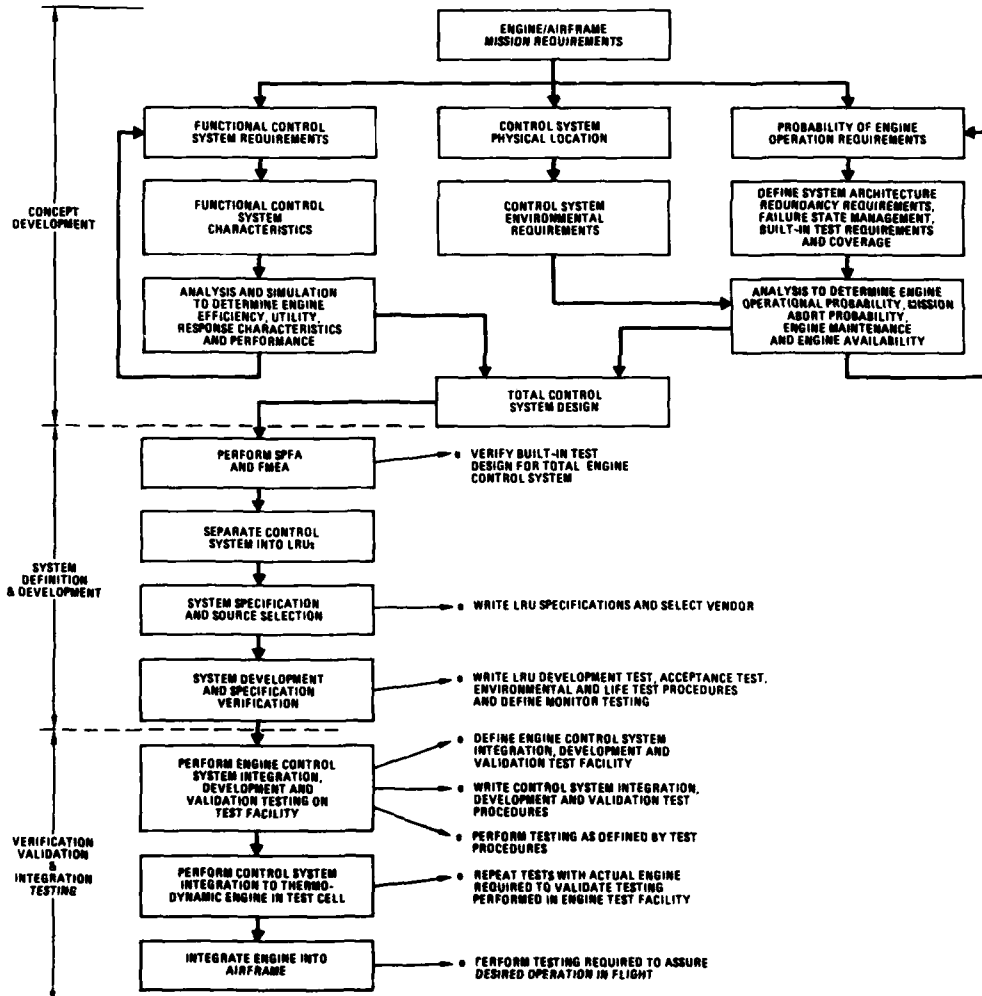


FIGURE 1.1. CONTROL SYSTEM DESIGN, INTEGRATION AND VALIDATION FLOW DIAGRAM

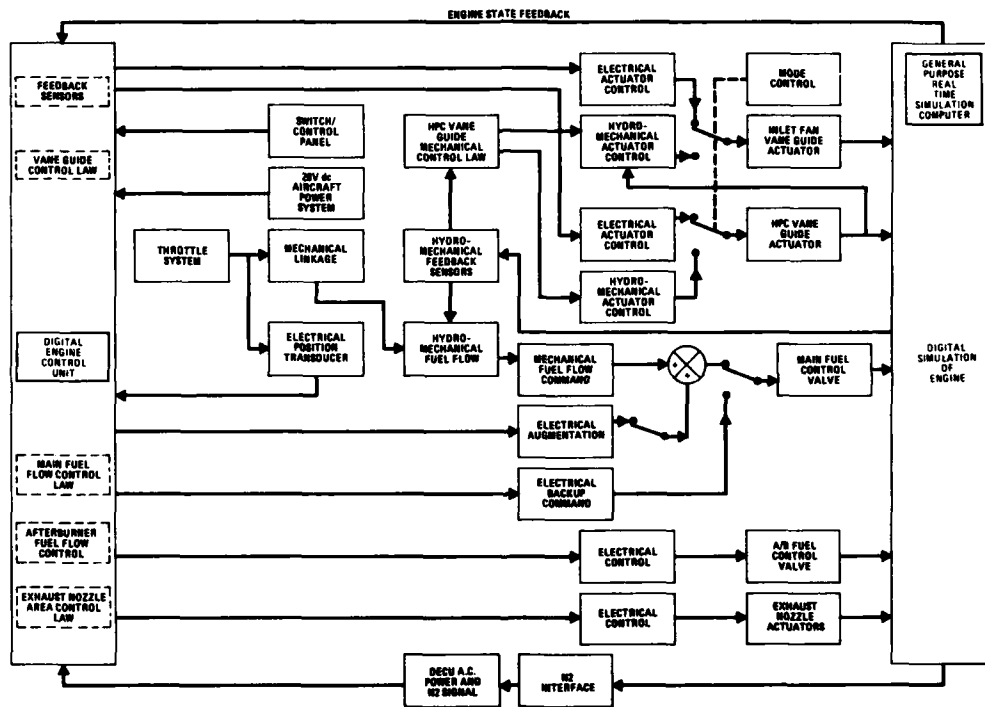


FIGURE 1.2. PRELIMINARY ENGINE TEST STAND

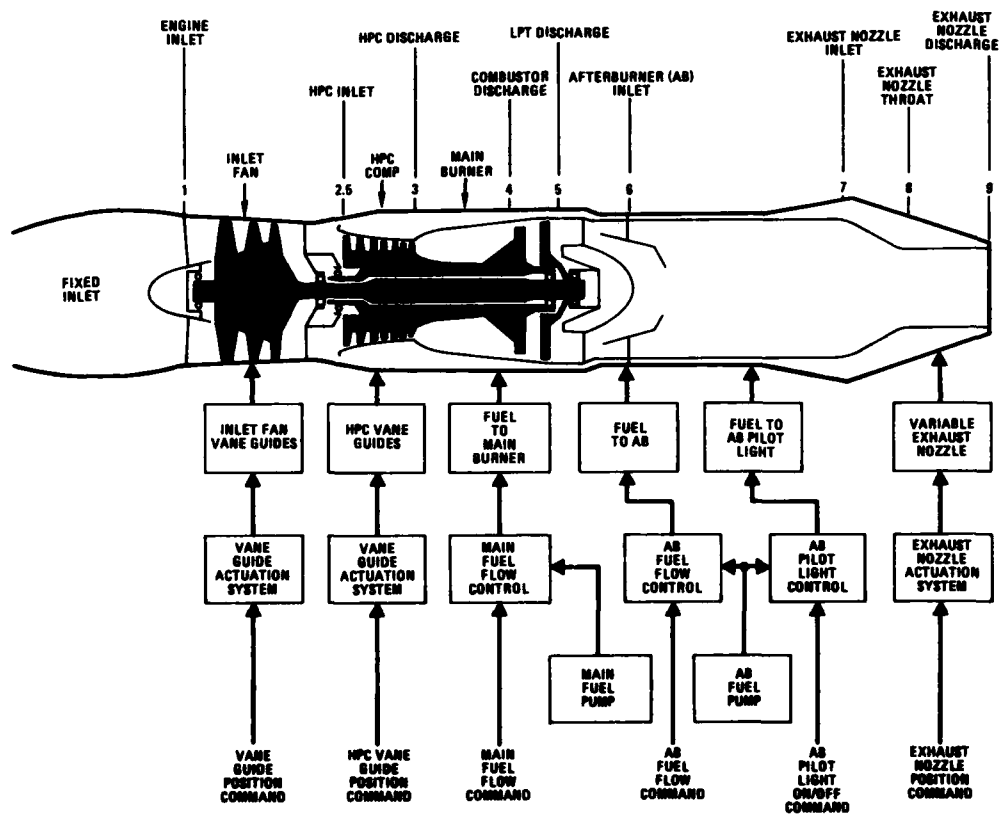


FIGURE 2.1. AFTERBURNING TWO STAGE TURBINE ENGINE WITH CONTROL INPUTS AND SENSORS

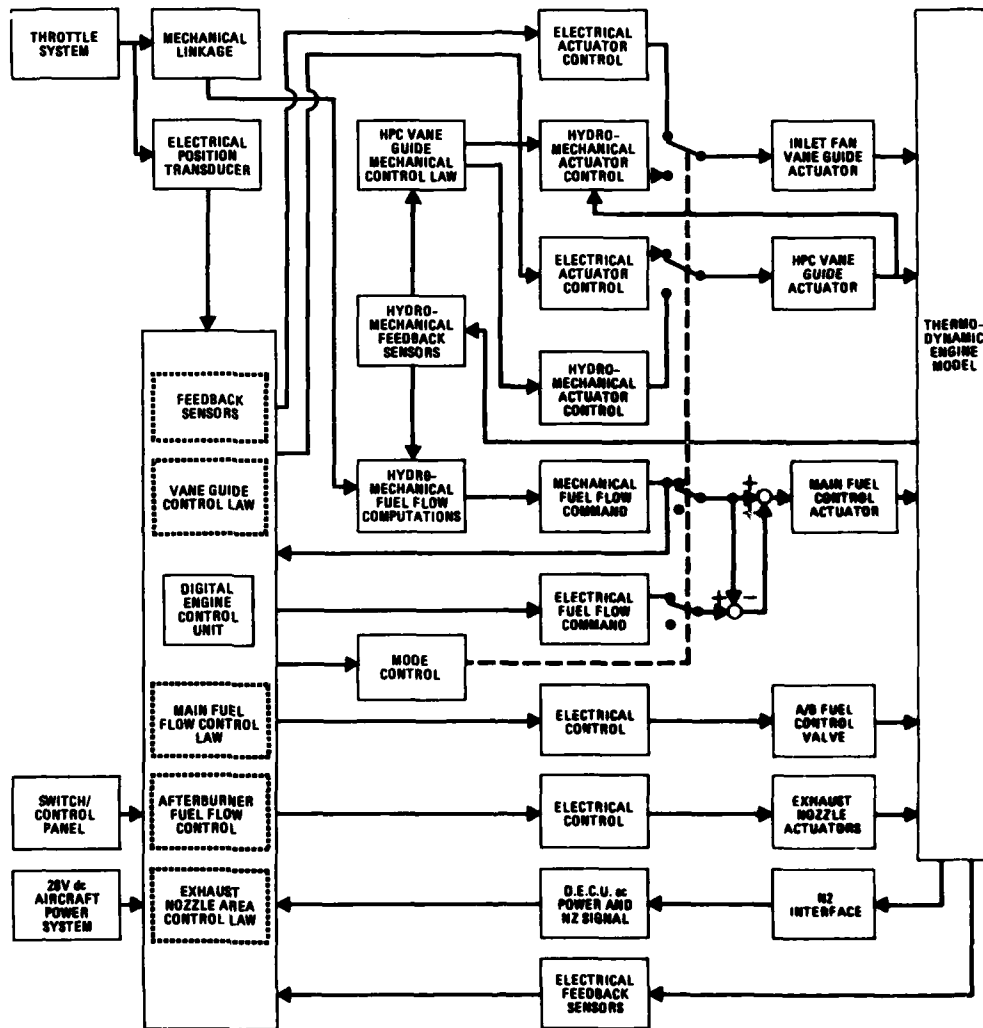


FIGURE 2.2. RECOMMENDED MECHANICAL AND ELECTRICAL CONTROL SYSTEM INTERFACE

1. TURBINE INLET TEMPERATURE LIMIT	20. RICH FUEL FLOW LIMIT FOR AB PILOT LIGHT OPERATION
2. LOW PRESSURE COMPRESSOR/TURBINE RPM LOWER LIMIT	21. LEAN FUEL FLOW LIMIT FOR AB PILOT LIGHT IGNITION
3. LOW PRESSURE COMPRESSOR/TURBINE RPM UPPER LIMIT	22. LEAN FUEL FLOW LIMIT FOR AB PILOT LIGHT OPERATION
4. LOW PRESSURE COMPRESSOR STALL LIMIT	23. RICH FUEL FLOW LIMIT FOR AB IGNITION
5. LOW PRESSURE COMPRESSOR OUTPUT PRESSURE UPPER LIMIT	24. RICH FUEL FLOW LIMIT FOR AB OPERATION
6. LOW PRESSURE COMPRESSOR OUTPUT PRESSURE LOWER LIMIT	25. LEAN FUEL FLOW LIMIT FOR AB IGNITION
7. ENGINE BYPASS RATIO (UPPER) LIMIT	26. LEAN FUEL FLOW LIMIT FOR AB OPERATION
8. ENGINE BYPASS RATIO (LOWER) LIMIT	27. MAX MAIN BURNER TEMPERATURE LIMIT
9. HIGH PRESSURE COMPRESSOR/TURBINE RPM LOWER LIMIT	28. MIN MAIN BURNER TEMPERATURE LIMIT
10. HIGH PRESSURE COMPRESSOR/TURBINE RPM UPPER LIMIT	29. MAX MAIN BURNER PRESSURE LIMIT
11. HIGH PRESSURE COMPRESSOR OUTPUT PRESSURE UPPER LIMIT	30. MIN MAIN BURNER PRESSURE LIMIT
12. HIGH PRESSURE COMPRESSOR OUTPUT PRESSURE LOWER LIMIT	31. MAX AB TEMPERATURE LIMIT
13. HIGH PRESSURE COMPRESSOR OUTPUT TEMPERATURE UPPER LIMIT	32. MIN AB TEMPERATURE LIMIT
14. HIGH PRESSURE COMPRESSOR OUTPUT TEMPERATURE LOWER LIMIT	33. MIN FAN VANE GUIDE ACTUATOR ELECTRICAL TRAVEL LIMIT
15. RICH FUEL FLOW LIMIT FOR MAIN COMBUSTOR IGNITION	34. MAX FAN VANE GUIDE ACTUATOR ELECTRICAL TRAVEL LIMIT
16. RICH FUEL FLOW LIMIT FOR MAIN COMBUSTOR OPERATION	35. MIN HPC VANE ACTUATOR MECHANICAL COMMAND LIMIT
17. LEAN FUEL FLOW LIMIT FOR MAIN COMBUSTOR IGNITION	36. MAX HPC VANE ACTUATOR MECHANICAL COMMAND LIMIT
18. LEAN FUEL FLOW LIMIT FOR MAIN COMBUSTOR OPERATION	37. MIN VEN ACTUATOR ELECTRICAL TRAVEL LIMIT
19. RICH FUEL FLOW LIMIT FOR AB PILOT LIGHT IGNITION	38. MAX VEN ACTUATOR ELECTRICAL TRAVEL LIMIT

TABLE 2.1 ENGINE CONTROL SYSTEM LIMITS

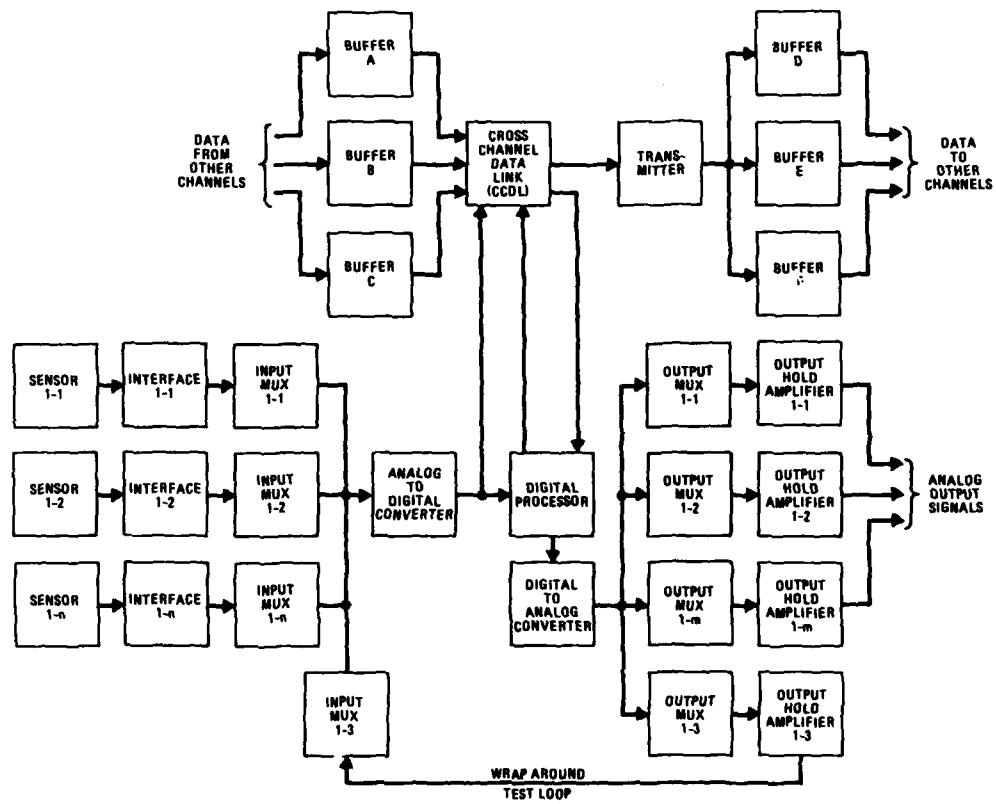


FIGURE 3.1. BLOCK DIAGRAM FOR ANALOG TO DIGITAL AND DIGITAL TO ANALOG CONVERTER SUBSYSTEM

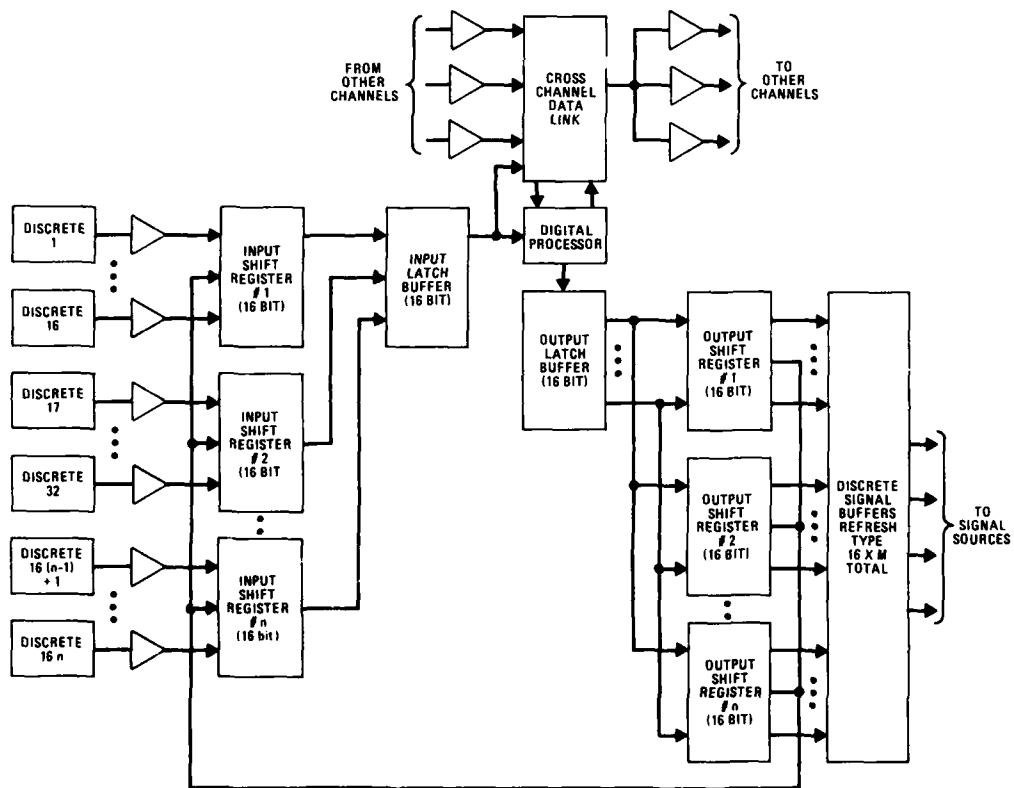


FIGURE 3.2. BLOCK DIAGRAM FOR DISCRETE SIGNAL INTERFACE SYSTEM

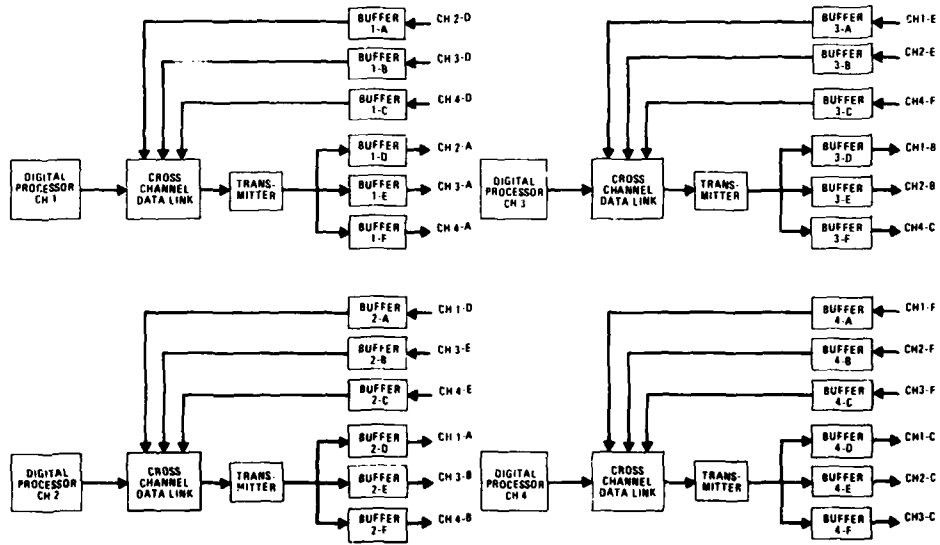


FIGURE 3.3. BLOCK DIAGRAM FOR CROSS CHANNEL DATA LINK SYSTEM

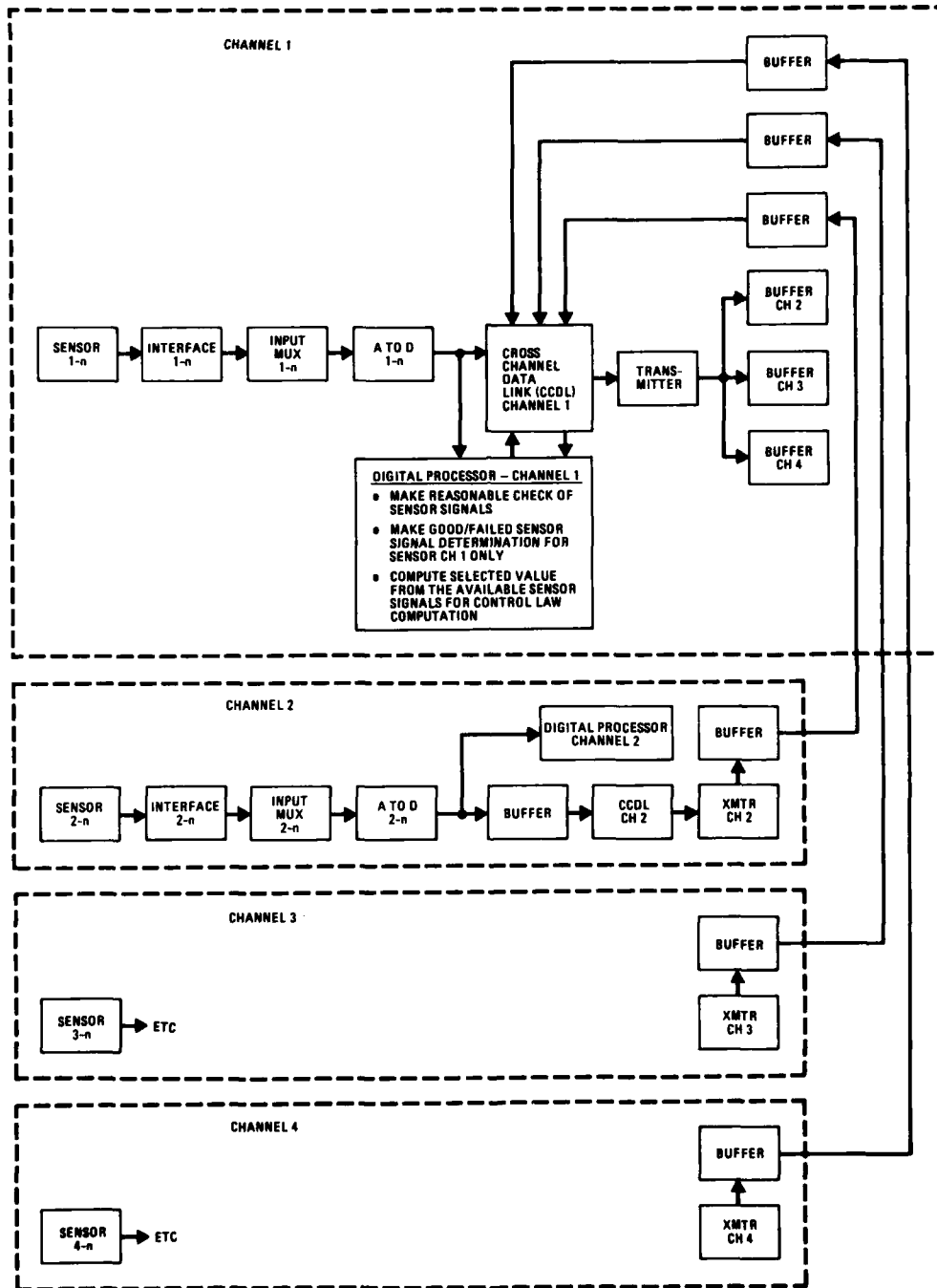


FIGURE 3.4. BLOCK DIAGRAM FOR REDUNDANT SENSOR SIGNAL MANAGEMENT

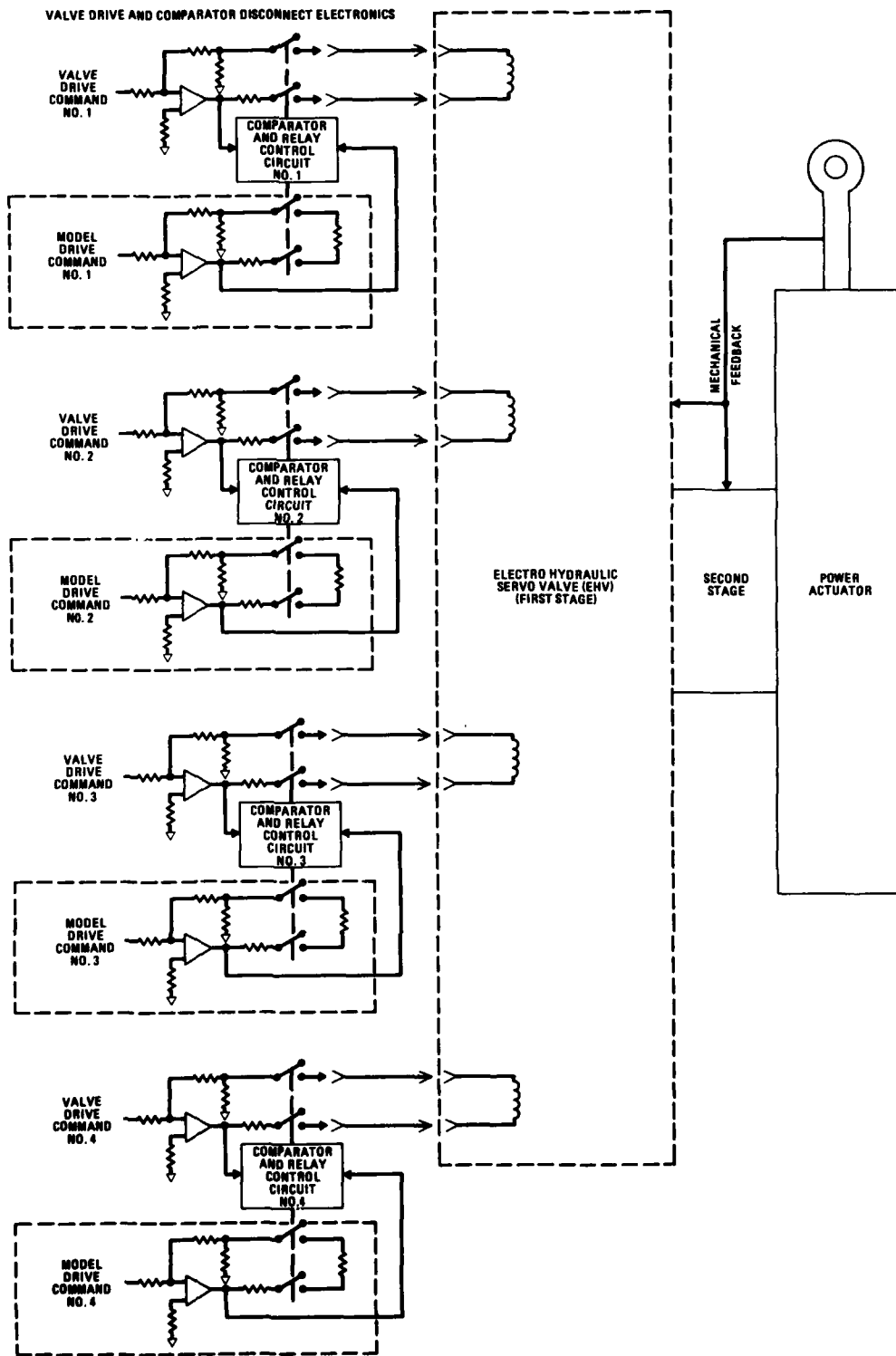


FIGURE 3.5. POSSIBLE ACTUATION SYSTEM CONCEPT

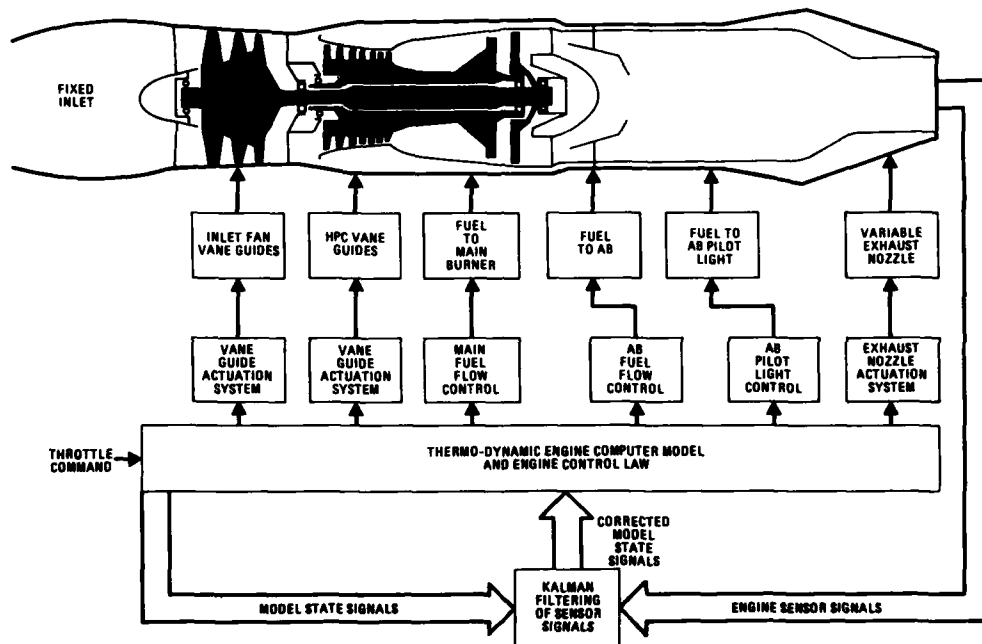


FIGURE 3.6. USE OF ONBOARD COMPUTER MODEL FOR FAILURE DETECTION AND CONTROL

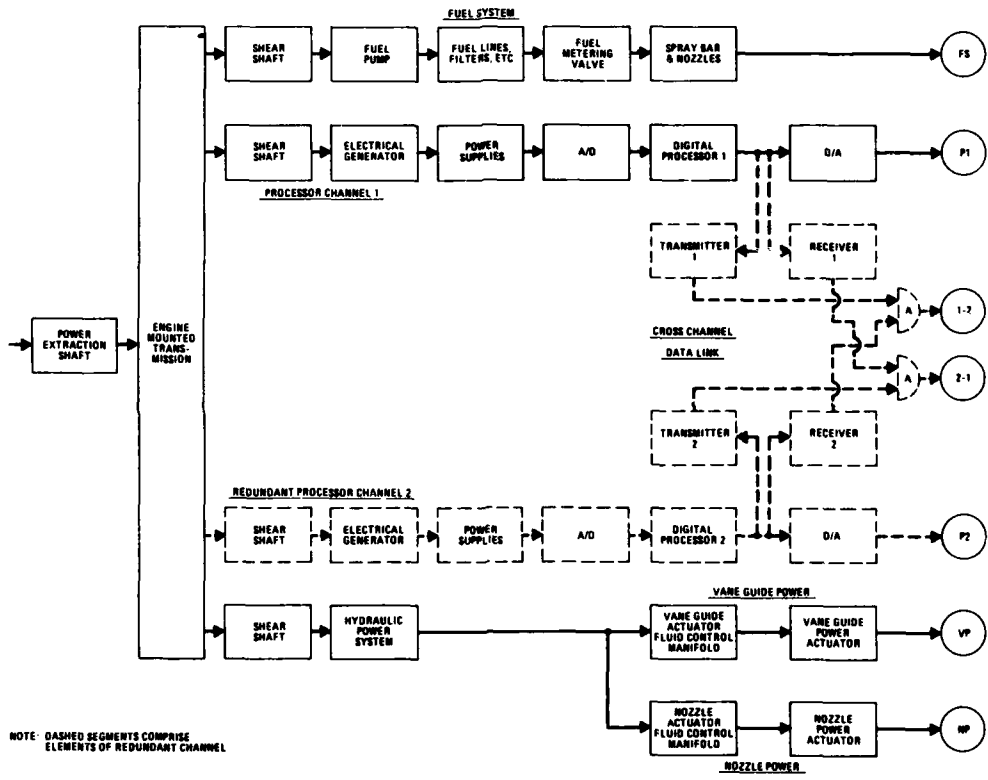


FIGURE 3-7. SIGNAL DEFINITION FOR OPERATION PROBABILITY FLOW GRAPHS

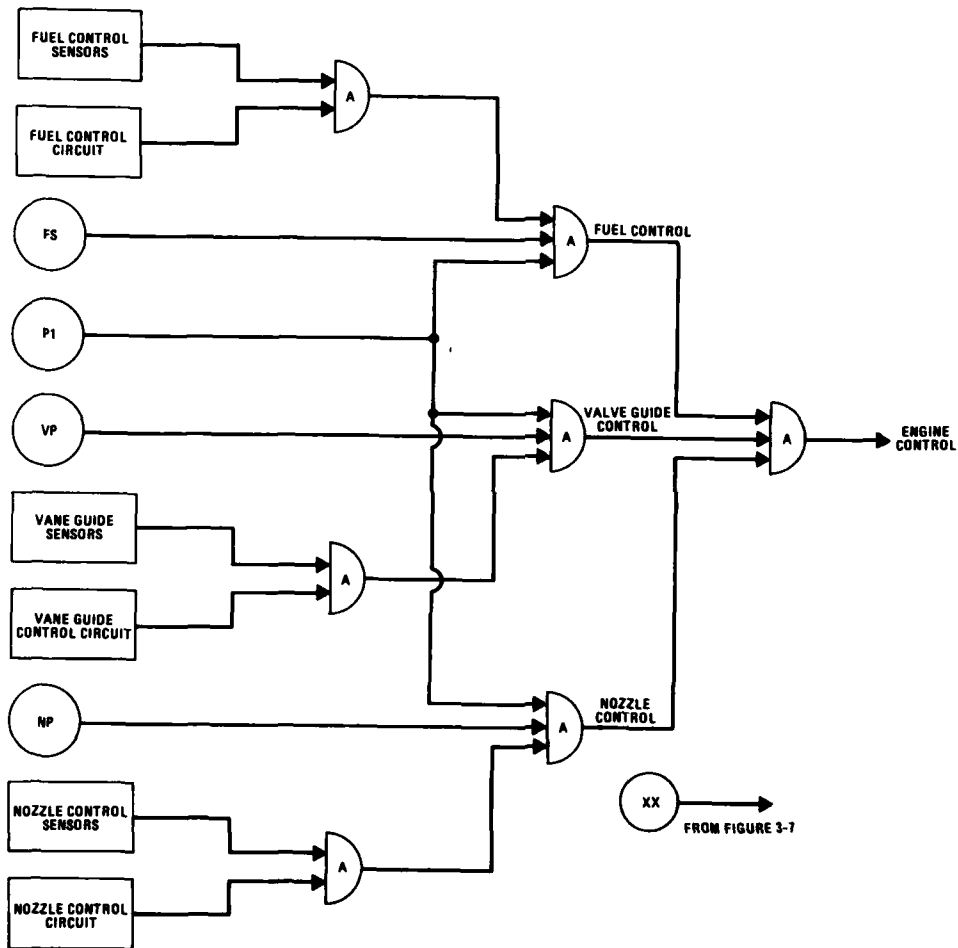


FIGURE 3-8. PROBABILITY OF OPERATION FLOW GRAPH FOR SINGLE CHANNEL SYSTEM

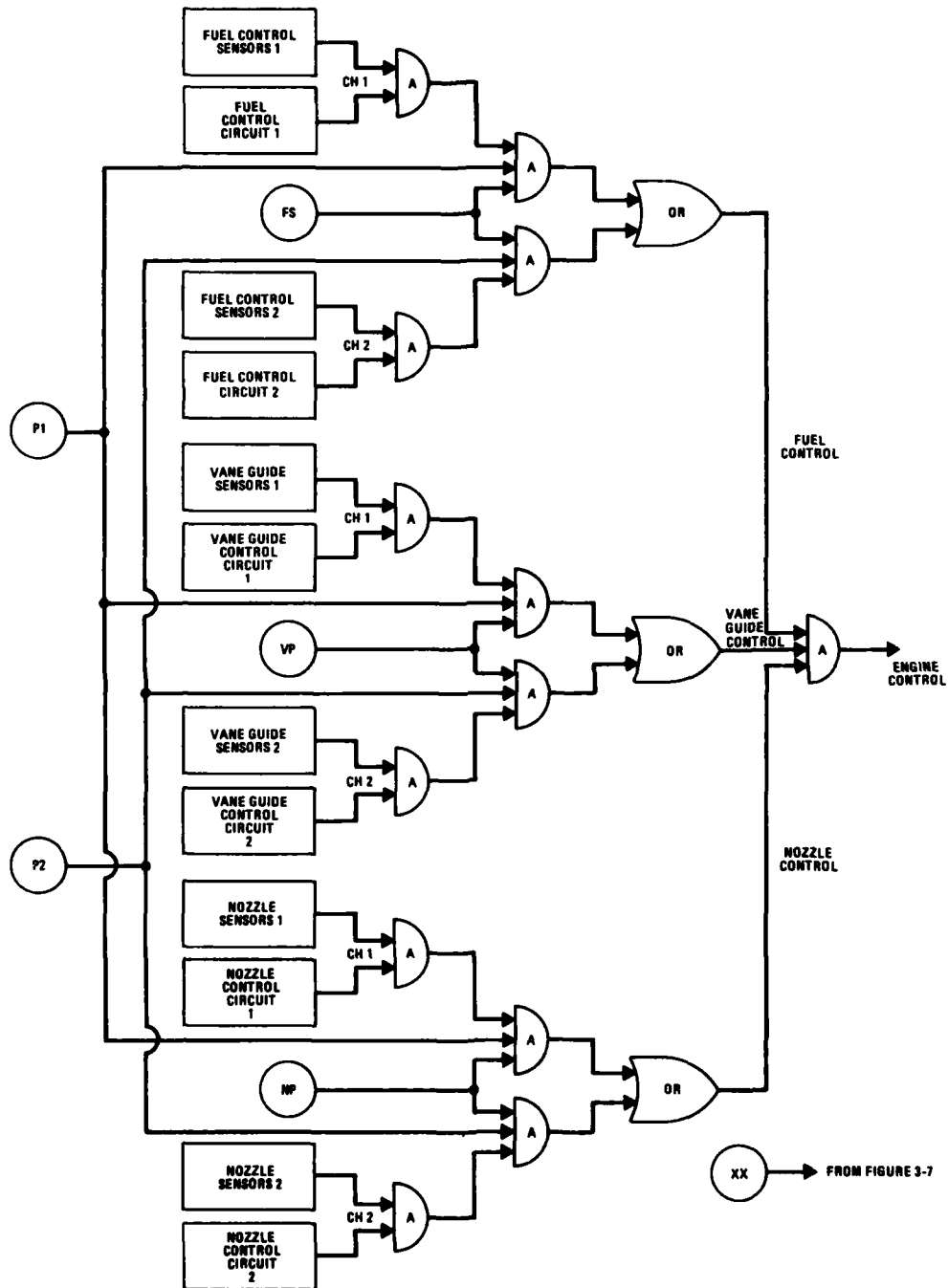


FIGURE 3-8. PROBABILITY OF OPERATION FLOW GRAPH FOR DUAL CHANNEL PROCESSING SYSTEM WITHOUT CROSS CHANNEL DATA LINK

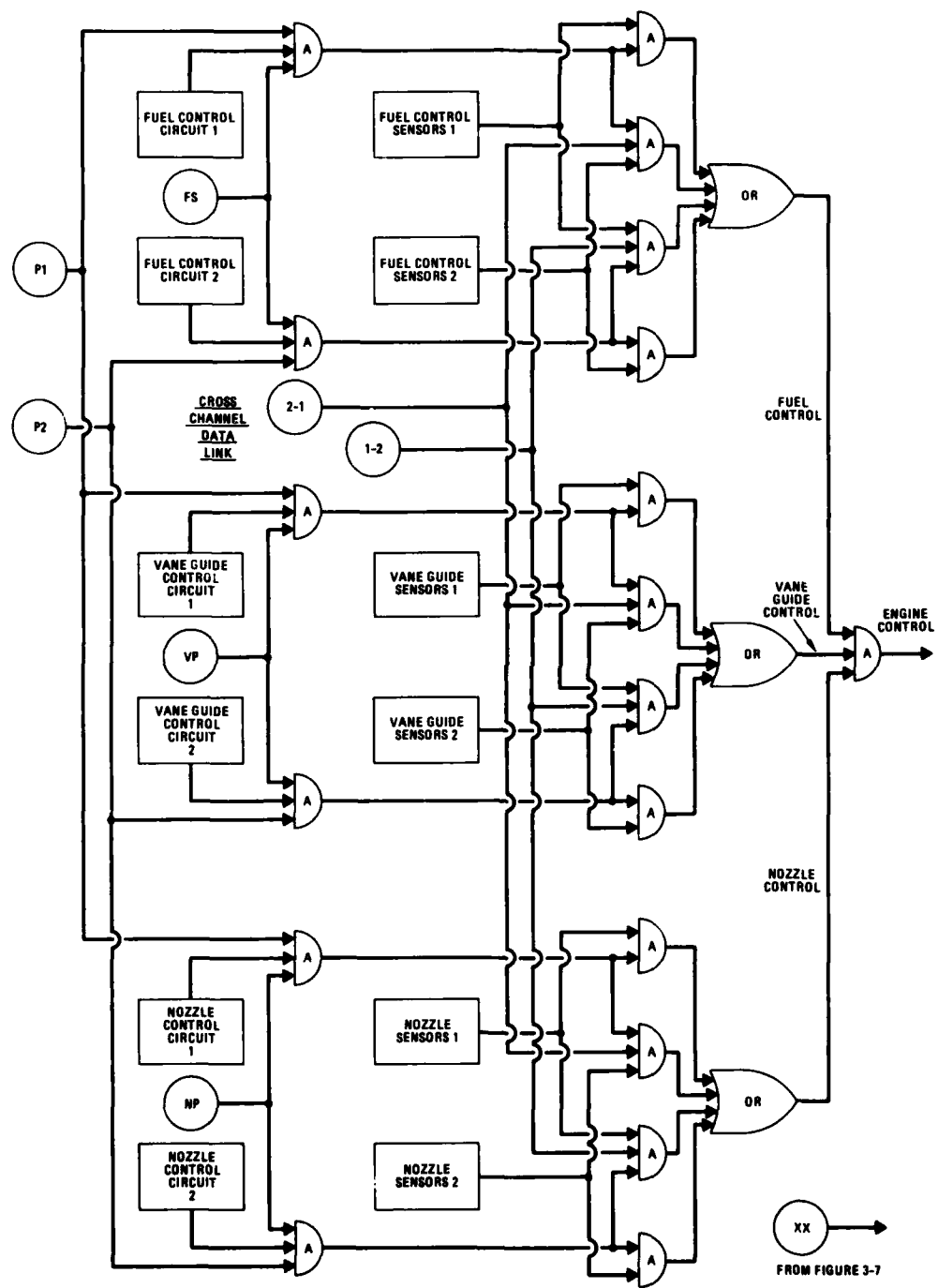


FIGURE 3-10. PROBABILITY OF OPERATION FLOW GRAPH FOR DUAL CHANNEL PROCESSING SYSTEM WITH CROSS CHANNEL DATA LINK

LIENS ENTRE LA REGULATION NUMERIQUE DU MOTEUR
ET LES AUTRES FONCTIONS DE L'AVION

par

D. RAMBACH

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RESUME

La régulation numérique ouvre la porte à une plus grande intégration du moteur à l'avion par le biais de nouveaux dialogues avec le pilote et les autres systèmes avions, soit pour assurer de nouvelles fonctions, soit pour perfectionner les services actuels. Les domaines visés couvrent de meilleures performances, une optimisation plus globale et une plus grande facilité de pilotage. Pour cela, il faut faire dialoguer plusieurs systèmes. Plusieurs structures le permettent. Une méthode de choix de structure est proposée pour conserver à chaque système important son autonomie en cas de difficulté et pour toutefois bénéficier des avantages de l'intégration. Les fonctions sont analysées puis regroupées en pyramide de façon à respecter une non propagation de panne. L'objectif est de conserver au système intégré une organisation compréhensible et maîtrisable. Cette organisation, non optimale, donne la priorité à la sécurité d'une part, et à un partage des responsabilités sans ambiguïté d'autre part.

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I - INTRODUCTION

Dans un moteur traditionnel, les fonctions de régulation moteur, de surveillance moteur et de maintenance, étaient bien distinctes, réalisées par des équipements séparés. La régulation assure le ou les débits carburants et la position des organes mobiles à partir d'équipements hydromécaniques, régulateurs et quelquefois électroniques, calculateurs analogiques. La surveillance se limitait à des indications pilotes - PCA - Tt7 - N2, charge au pilote d'exercer la surveillance proprement dite. La maintenance se faisait au sol, par des moyens spécifiques, mallettes, cabines de test, suivant un calendrier programmé ou en fonction d'anomalies de fonctionnement telles qu'elles pouvaient être perçues par le pilote.

Si les fonctions régulation, surveillance, maintenance, étaient distinctes, à fortiori les autres fonctions de l'avion étaient réalisées quasiment sans lien avec le moteur. L'introduction de la régulation numérique, rendue indispensable par la complication des moteurs, incite à repenser ces liens, soit parce que la réalisation d'une nouvelle fonction peut quelquefois être limitée à une simple modification de logiciel plutôt qu'à l'adjonction d'un nouvel équipement, soit parce qu'un dialogue numérique permet d'envisager des optimisations globales qui auraient été trop pénalisantes en analogique.

A titre d'exemple, une régulation numérique redondante implique un système sophistiqué de détection de panne, pour pouvoir automatiquement et très rapidement commuter d'une voie sur une autre. Cet autotest régulation implique une surveillance moteur qui peut se substituer à celle exercée par le pilote.

Un autre exemple d'interférences entre systèmes distincts rendus possibles par le numérique, pourrait être une modification automatique des lois de régulation en fonction de la charge avion.

II - INTEGRATION MOTEUR AVION

- II - 1. Intérêt d'une intégration fonctionnelle
 - II - 1.1. La régulation du moteur

La régulation, pour ses besoins propres, recueille sur le moteur un ensemble d'informations et les traite. Il serait dommage que ces résultats ne soient pas disponibles pour les dispositifs ayant besoin d'informations moteur. Ceux-ci sont d'une part, la surveil-

lance en vol du moteur et sa maintenance, d'autre part les autres systèmes avions dont en particulier la navigation et l'armement.

Nous examinerons ci-dessous quelques cas où l'intérêt d'une intégration fonctionnelle nous semble manifeste.

La demande du pilote est double. Etre libéré de toute surveillance moteur quand "tout va bien"; il se crée donc une nouvelle fonction de surveillance automatique. Cette fonction s'appuie sur une modélisation temps réel embarquée. L'autre demande est de disposer d'informations précises quand "tout ne va plus bien" et d'un guide de décision (mémento électronique). Ceci suppose qu'après la détection d'une anomalie il y ait une localisation suffisamment précise de la défaillance et, soit une reconfiguration automatique avec information au pilote (perte redondance, par exemple), soit reconfiguration manuelle avec diagnostic en clair au pilote (tube cathodique).

II - 1.2. La maintenance moteur

La maintenance justifie dès aujourd'hui sur des moteurs existants des équipements tels que concentrateur de données (PMUX), compteurs de cycles, etc.... La fonction concentration de données est en grande partie redondante avec celle de régulation. Les traitements de ces informations pour les besoins de la maintenance sont très variables suivant les utilisations, mais ils peuvent tous utiliser des mesures et des calculs effectués par la régulation. Sans définir les niveaux de maintenance on peut distinguer quatre niveaux de "demandeurs" d'informations qui visent à la maintenance du moteur :

- Diagnostic en vol.
- Enregistrement en vol (et dépouillement au sol).
- Tests au sol.
- Gestion centralisée d'un parc de moteurs.

A l'exception peut être du premier niveau, ces opérations font appel à des matériels spécifiques à la maintenance et diffèrent entre eux. Il est par contre possible (et souhaitable) que le moteur présente une interface commune à ces matériels. Les niveaux de maintenance étant de plus en plus informatisés cette interface moteur, elle-même numérique, est un périphérique intelligent des moyens de maintenance.

II - 1.3. Au-delà du moteur

Avec la navigation et l'armement on entre dans le domaine de l'intégration fonctionnelle proprement dite. Déjà aujourd'hui des calculateurs comme l'automanette, le N1 limite, l'appauvrisseur de tir font des liens entre navigation, tir et moteur. Il est bien évident qu'en matière d'optimisation (nombre de liaisons électriques ou économies carburant par exemple) une meilleure répartition doit être étudiée.

Nous prendrons trois exemples :

- Dans un avion supersonique, les entrées d'air ont une géométrie variable pour adapter l'écoulement et obtenir le meilleur rendement d'entrée d'air. Il s'agit en général d'une régulation en boucle ouverte ou la position de l'élément mobile est fonction du Mach. Prendre en compte le paramètre entrée d'air dans le problème de commande multivariable que constitue déjà la régulation ne pourrait qu'améliorer les performances générales. Cette approche pourrait être généralisée aux commandes de vol pour lesquelles la poussée est un paramètre essentiel de la dynamique de vol.

- Un autre exemple où une certaine intégration est déjà un état de fait, est celui du tir. Devant le risque de présence de gaz chaud devant le moteur, le régime de celui-ci est automatiquement réduit par la régulation pour éviter un risque de décrochage. De même, d'autres conditions de mauvaise admission, comme une forte incidence, conduisent à une réduction préventive des gaz. Une meilleure connaissance des conditions réelles d'admission et une action plus modulée, c'est-à-dire moins brutale, pourrait réduire sinon éliminer des chutes de performances qui peuvent être assez gênantes pour le pilote.

- Le troisième exemple s'appuie dans son vocabulaire sur l'aviation commerciale. Certains avions sont aujourd'hui équipés d'un F.M.S. (Flight Management System) connecté à un T.C.C. (Thrust Control Computer) lui-même agissant sur une automanette d'une part, et sur la régulation moteur d'autre part, par l'intermédiaire d'un P.M.C. (Power Management Control). Il y a là une multiplicité de calculateurs et l'on verra sans doute rapidement cette chaîne se réduire à au moins une unité, car le moteur sera à même d'être directement connecté aux autres calculateurs.

II - 2. Modes de régulation intégrés

La régulation a pour fonction de transformer une demande de poussée (manette des gaz) en une poussée effective. Pour cela, elle doit assurer un fonctionnement stabilisé en chaque point du domaine de vol le plus proche possible du point de fonctionnement thermodynamique optimal du moteur, et elle doit assurer les transitoires les plus brillants possible en respectant tout un ensemble de contraintes : surchauffe, survitesse, décrochage riche, décrochage pauvre, Avec une régulation hydromécanique ou même électronique analogique, il n'était réaliste que de trouver un mode de régulation "universel" qui était le meilleur compromis. En numérique, il est beaucoup plus envisageable d'avoir plusieurs modes de régulations.

Certains modes sont des choix internes de régulation du type mode économique, dont le critère sera de réduire la consommation de potentiel moteur ou de carburant, un autre peut être le mode opposé, "super plus", qui donnera un surcroît de performance au détriment

du potentiel, équivalent aujourd'hui de la surcharge. Il y a évidemment également les modes de secours ou modes dégradés qui seront automatiquement enclenchés en cas de panne.

Il peut y avoir également des modes résultant d'une étude d'intégration. Il existe déjà des régulations d'approche ou régulations de Vi, mais d'autres modes peuvent être envisagés. Il y a ceux qui correspondent comme la régulation de Vi, à moduler la poussée pour asservir un paramètre de vol. C'est en sorte un pilote automatique du moteur. D'autres modes pourraient rester à commande manuelle, mais en modifiant l'interprétation de cette commande. Par exemple, par une action particulière sur la manette on pourrait augmenter sa sensibilité autour du point de fonctionnement, effet de loupe, pour faciliter le vol en patrouille.

Enfin, certains modes de régulation, y compris le mode normal, peuvent être personnalisés en fonction de l'utilisateur et des missions de l'avion.

II - 3. Description d'un système propulsif intégré

Nous avons jusqu'ici employé le terme d'intégration fonctionnelle. C'est pour l'opposer à intégration physique. Il nous paraît que ce serait une erreur d'envisager l'intégration comme la réunion de plusieurs petits calculateurs en un seul gros même si l'intégration doit permettre de réduire le nombre de calculateurs. Sans vouloir redémontrer les risques et les effets néfastes d'un seul calculateur central avion, nous avons fait le bilan analogue au niveau du moteur. Refuser que les calculs se fassent au niveau du moteur n'empêche pas qu'il serait nécessaire de conditionner et multiplexer les mesures moteurs. Le boîtier qui en résulterait, surtout s'il doit être muni d'un coupleur numérique pour dialogue avec le calculateur central, sera presque aussi important que le calculateur de régulation autonome et source de beaucoup plus de problèmes, en particulier sur le plan de la sécurité.

Il nous semble donc, qu'à la base et par moteur, il doit y avoir un dispositif (calculateur numérique) capable d'assurer les fonctions :

- de régulation
- de dialogue numérique
- de concentration de données.

Un certain nombre d'autres fonctions sont moins attachées au moteur proprement dit, et couvrent plutôt l'ensemble du système propulsif, en particulier dans le cas d'un multimoteur. Il s'agit d'une sorte de superviseur qui effectue des tâches de traitement et sélection de consignes qui interprète des informations moteurs et les met en forme pour d'autres systèmes avion. C'est au minimum le coupleur des bus avions. Cette fonction peut être physiquement intégrée au calculateur de régulation, mais elle peut aussi être distincte; nous l'appellerons alors : calculateur de gestion du système propulsif. Cette structure (cf. Figure 1) permet de distinguer un dialogue interne numérique entre capteurs numériques (le jour où), calculateurs de régulation, calculateurs de gestion du système propulsif, liaison qui doit être très sûre mais qui peut être lente et le dialogue avion sur le bus rapide choisi par l'avionneur et modifiable selon l'application.

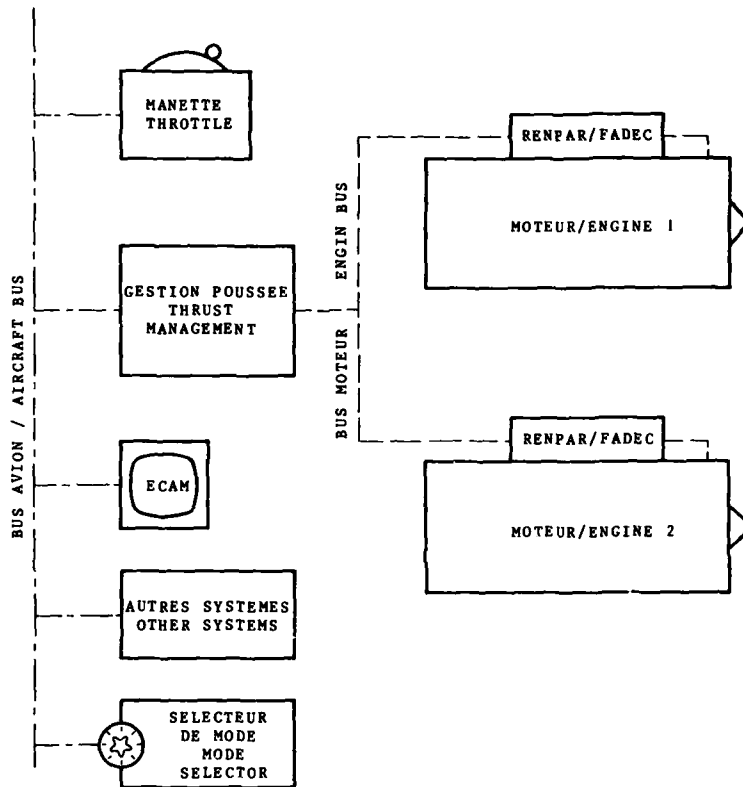
III - LES CHOIX EN MATIERE D'INTEGRATION

III - 1. Différents types d'intégration

Si l'on considère que l'intégration consiste à réunir plusieurs fonctions en une fonction à caractère plus général, il faut distinguer alors plusieurs types d'intégration selon l'architecture des liaisons physiques et celle des liaisons fonctionnelles. Nous citerons comme exemples quelques uns des choix possibles sans chercher à les analyser.

- Intégration centralisée : le maximum de fonctions sont regroupées dans un équipement central (cf. Figure 2.1).
- Intégration "anarchique" : souvent résulte d'apports successifs; les liaisons physiques et fonctionnelles ne coïncident pas forcément et les interactions sont multiples et désordonnées (cf. Figure 2.2).
- Intégration physique structurée mais intégration fonctionnelle anarchique; les liaisons physiques sont réduites et semblent cohérentes mais les dépendances fonctionnelles sont très différentes des liaisons physiques (cf. Figure 2.3).
- Intégration physique et fonctionnelle structurée. Les liaisons physique et fonctionnelle coïncident et le système est tel qu'en cas de perte de certaines fonctions les autres fonctions restent capables de rendre un minimum de services.

La plupart des grands systèmes d'un avion sont en général considérés comme suffisamment importants pour que la perte de la fonction correspondante soit difficile à admettre. Chaque système peut alors être doté de redondances telles que la probabilité de perte de la fonction devienne négligeable. Il est séduisant alors de relier ces systèmes par un (ou deux) bus numérique "sûr" et de s'autoriser alors n'importe quel type d'échange (intégration type, Figure 2.3). Cette démarche est en principe acceptable, mais tout risque sur un système, éventuellement acceptable pour ce système, peut être inacceptable pour un autre. Une telle intégration est donc dangereuse et nous lui préférons une architecture où chaque système réalise une intégration physique et fonctionnelle structurée (type, Figure 2.4), les systèmes étant reliés entre eux par une liaison (type bus) mais n'affectant qu'une partie bien définie de chaque système interconnecté (cf. Figure 2.5).



SYSTEME PROPULSIF INTEGRE

FIGURE 1

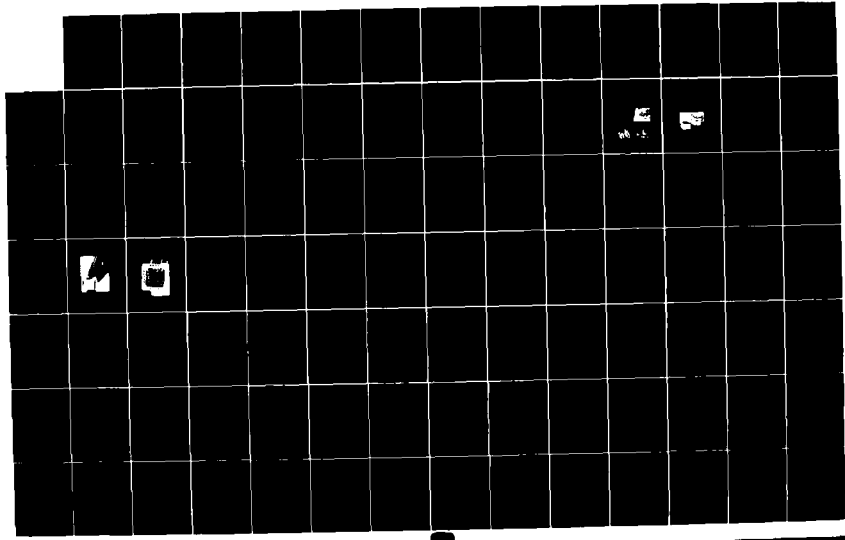
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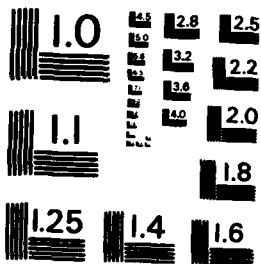
INTEGRATION OF FIRE CONTROL FLIGHT CONTROL AND
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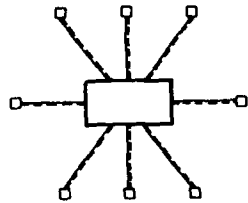
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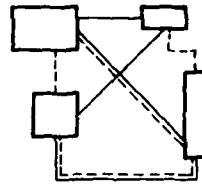


MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS - 1963 - A



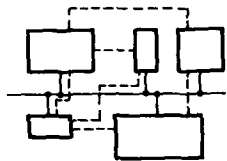
INTEGRATION CENTRALISEE

Figure 2.1



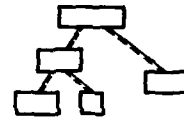
INTEGRATION ANARCHIQUE

Figure 2.2



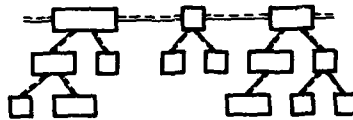
INTEGRATION PHYSIQUE STRUCTUREE
INTEGRATION FONCTIONNELLE ANARCHIQUE

Figure 2.3



INTEGRATION PHYSIQUE ET FONCTIONNELLE
STRUCTUREE

Figure 2.4



ENSEMBLE DE SYSTEMES LIES PAR BUS

Figure 2.5

III - 2. Les limites de l'intégration fonctionnelle

La séparation des fonctions était la conséquence de deux objectifs :

- Une bonne sécurité.
- Des responsabilités techniques et industrielles claires.

Ces objectifs essentiels pourraient se voir remis en cause par une intégration conduite sans précautions.

L'intégration est en fait une façon d'apporter des services supplémentaires mais ces services ne vont pas dans le sens de la diminution du matériel. Plus de matériel veut dire plus de pannes et ces pannes peuvent avoir plus de répercussions. Il doit donc y avoir une analyse de la sécurité pour qu'en cas d'incident d'un dispositif complémentaire (écran cathodique, enregistreur,) on ne perde pas une fonction fondamentale (régulation du moteur,).

La nécessité de responsabilités techniques et industrielles claires est également fondamentale. Il faut que chaque participant à un programme puisse s'engager à satisfaire un objectif sans que cet objectif dépende d'un autre participant dont le propre objectif dépend lui-même du premier participant. Dans un tel partage chaque participant rejettera sur l'autre le maximum de problèmes et en cas de défaut de fonctionnement une polémique sans fin s'engagera. Un cas particulièrement sensible est celui d'un système pour lequel les responsables fonctionnels, matériels et logiciels ne seraient pas confondus.

La limite peut être de l'intégration fonctionnelle est de limiter celle-ci de façon à ce que le maître d'oeuvre, qui est un homme et non pas une machine, puisse conserver la compréhension du fonctionnement de son système, de ce qui fait quoi et de qui fait quoi.

III - 3. Construction d'un système intégré

La suite et la fin de cet article sera consacrée à décrire une méthode de construction d'un système intégré que nous appellerons par simplification pyramide d'intégration.

Cette méthode ne prétend pas découvrir quoi que ce soit de neuf. Elle veut au contraire faire comprendre une approche qui n'est sans doute ou du moins nous l'espérons que l'expression du bon sens.

Elle ne se veut pas non plus une méthode d'optimisation car au contraire, elle impose des contraintes pénalisantes de façon à respecter les limites de l'intégration fonctionnelle telles que décrites ci-dessus.

III - 4. Principe de la pyramide d'intégration

Le principe de la pyramide d'intégration consiste à identifier l'ensemble des fonctions à réaliser : fonctions A, B, C, Z, puis à créer une relation de dépendance entre ces fonctions du type, la fonction D est constituée de la réunion des fonctions A, B et C. Ces fonctions doivent ensuite être associées à leur mode de réalisation ce qui conduira à constater que certaines des fonctions A, B et C peuvent exister seules (A et B par exemple) d'autres non (C par exemple). On dira alors que A et B sont d'un niveau inférieur à C et D et on les représentera comme le montre la Figure 3.1.

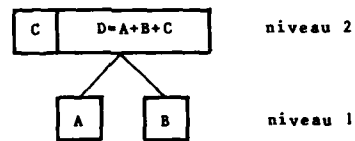
L'ensemble des fonctions sera alors architecturé en créant le nombre de niveau nécessaire et en s'interdisant qu'une fonction d'un certain niveau dépende de plus d'une fonction de niveau immédiatement supérieur. On obtient alors des architectures telles que représentées dans la Figure 3.2.

La pyramide ne sera satisfaisante que lorsqu'il sera possible de démontrer qu'une panne ne peut pas descendre la pyramide, c'est-à-dire que la perte d'une fonction ne peut pas affecter les niveaux inférieurs.

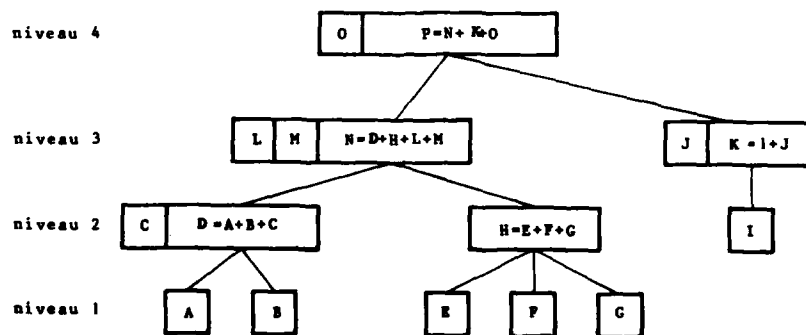
Dans cette décomposition, un mode dégradé est une fonction en tant que tel (A ou B par exemple), et le complément à ce mode dégradé (C par exemple) une autre fonction. Le mode normal (D par exemple) est bien la réunion de ces modes.

III - 5. Exemple d'application

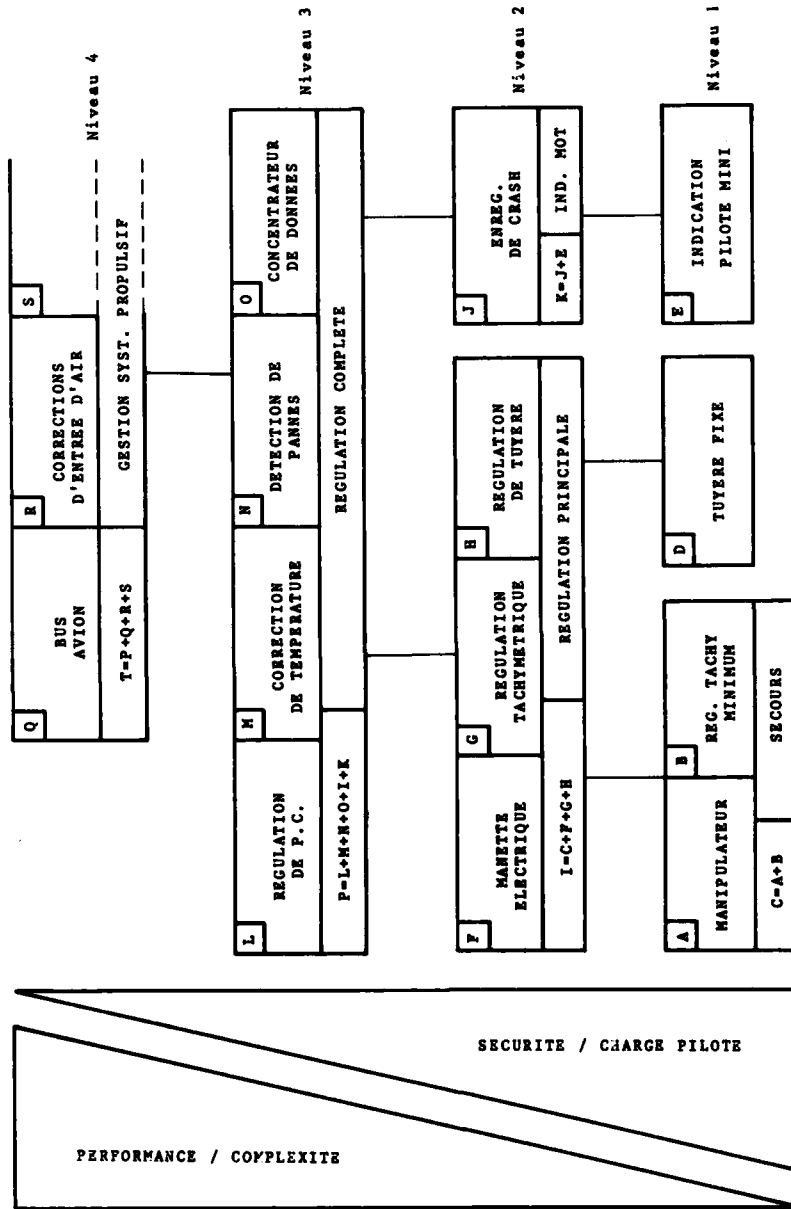
La Figure 4 constitue un exemple plus explicite de la pyramide d'intégration. Examinons le cas de la manette des gaz. La manette des gaz représente la demande de poussée. Celle-ci est aujourd'hui transmise par un système de tringlerie et de flexible lourd et source de nombreuses difficultés. Il serait intéressant de faire plutôt transiter cette demande par un bus avion et cela d'autant plus qu'en cas "d'intégration" cette demande pilote sera corrigée par d'autres systèmes avion. Dans la description de la Figure 4, la liaison numérique (bus avion) intervient au niveau supérieur de la pyramide (niveau 4). L'information "manette digitale" est donc utilisée par le calculateur de gestion de la poussée. En cas de perte de cette information les niveaux inférieurs ne doivent pas être affectés. Leur fonction nécessitant une information manette, il est donc obligatoire qu'une autre information vienne à un niveau inférieur par exemple au niveau 2, manette électrique. Les fonctions spécifiques à la manette digitale ne peuvent donc qu'appartenir au niveau 4. De la même façon il apparaît une fonction au niveau 1 régulation tachymétrique minimum qui doit être assurée même en cas de perte de la manette électrique. Il faut alors créer un troisième niveau par exemple un manipulateur à impulsions ou accepter un régime fixe.



PYRAMIDE ELEMENTAIRE
Figure 3.1



PYRAMIDE D'INTEGRATION
Figure 3.2



EXEMPLE DE PYRAMIDE D'INTEGRATION
Figure 4

III - 6. Avantages et inconvénients de la pyramide

Commençons par les inconvénients : La pyramide d'intégration est très contraignante, car elle interdit le partage d'une même fonction de niveau inférieur par plusieurs fonctions de niveau supérieur. De même, elle interdit, même si elle est assortie d'une très bonne fiabilité, la descente d'une information. C'est une démarche logique et non pas probabiliste. Elle ne conduit ni vers une solution unique ni, à fortiori, optimale.

Par contre, elle se prête bien à une analyse de panne. Elle lie bien les spécifications fonctionnelles et la réalisation matérielle. Elle permet d'aboutir à des partages (sous-traitances) sains, c'est-à-dire où le coopérant est responsable d'un ensemble cohérent.

IV - CONCLUSION

L'utilisation du numérique à bord de l'avion et en particulier dans le cas du moteur, ouvre la porte à de multiples communications qui éviteront certaines redondances et surtout qui permettront d'offrir tout un ensemble de nouveaux services.

Le risque, et le moteur y est particulièrement sensible, est que pour faire un peu mieux, on aboutisse à un système si interdépendant qu'il ne garantisse plus la sécurité. Il nous semble donc qu'au moins pour les grandes fonctions de l'avion, il est indispensable de s'imposer qu'en cas d'incident, quitte à perdre des services supplémentaires, le système revienne automatiquement à des fonctions de base, bien indépendantes sans possibilité de propagation de panne. Ceci veut dire que chaque système est conçu et réalisé comme un système indépendant aux ordres du pilote et que l'intégration fonctionnelle se superpose pour augmenter les performances et diminuer la charge de travail du pilote.

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INTEGRATED POWERPLANT CONTROL SYSTEMS AND POTENTIAL PERFORMANCE BENEFITS

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SUMMARY

Three aspects of integration are examined in which the Powerplant Control System can be integrated with the Aircraft Systems to provide mutual benefits. It is shown how the Powerplant Control System can be progressively integrated into the aircraft systems architecture by means of a distributed computing system for Utility Systems Management. A number of benefits can be expected as a result but, more importantly, the Powerplant Control System is shown to be a candidate for further improvement as a result of the integration and the ready availability of off-take load related data. Finally some of the advantages of integration with Flight Control are discussed.

INTRODUCTION

Work in the area of Utility Systems Management for future aircraft conducted at British Aerospace, Warton has led to the adoption of a control system architecture based on Digital Computing and Multiplex Data Busses. The system evolved has enabled the base mechanical functions of the aircraft: Fuel System, Hydraulic Systems, Environmental Control and Secondary Power Control, etc to be successfully integrated into the Avionic architecture which includes high technology cockpit displays and controls essential for the reduction of pilot workload in complex weapon systems.

The system can be shown to be suitable for allowing the introduction of Powerplant Control Systems into the aircraft in such a way as to retain the required integrity, allow intercommunication with other systems, and, importantly, to allow a gradual increase in technology from current day analogue powerplant control systems to fully integrated digital systems with minimum disturbance to the overall aircraft programme.

In addition the use of sophisticated digital systems allows more data to be made available which can be used to the mutual benefit of the integrated systems. Since the aircraft manufacturer is constantly seeking improvements to the weapon system in order to make his product more competitive integration is seen as an important tool to achieve performance benefits. Once integrated into this structure the Powerplant Control System becomes a candidate for similar potential benefits.

FUTURE AIRCRAFT REQUIREMENTS

For future combat aircraft there is pressure to achieve better manoeuvrability under combat conditions, and the generation of projects aimed at providing Vertical Take Off and Landing (VTOL) or Short Take Off and Vertical Landing (STOVL) capability collectively provide additional challenges to the control system designer. As well as improvement in combat handling there is a need to improve the overall aircraft performance by using lighter materials to reduce airframe weight and by improving engine performance over the aircraft flight envelope.

In addition the increased power load resulting from a large avionic equipment fit and the demands for high rate control surface movements lead to a need for high power offtakes with transient demands. This can result in problems with maintaining optimum continued engine operation throughout the flight envelope.

Aircraft which use thrust to provide manoeuvre control, either for take off and landing or for combat manoeuvre enhancement, demand a degree of integration of control that has hitherto not been considered. In this type of aircraft especially the pilot will expect to achieve carefree handling of both aircraft and engine in all regimes of flight with minimum workload.

ASPECTS OF INTEGRATION

Integration is evident in many areas of the modern aircraft project proposal, particularly in Avionics, where the Data Bus is utilised as a major integrating medium for functional components of the system. The data bus and the emergence of readily available digital computing elements has given impetus to the integration of mechanical

systems into the avionic structure. These mechanical systems include the aircraft Utility Systems and the Powerplant Control System.

Integration in the area of Powerplant Control can take a number of forms :

1. Integration of Powerplant Control with the Powerplant.
2. Integration of Powerplant Control and Flight Control.
3. Integration of Powerplant Control and Aircraft Systems.

Each aspect has its own advantages and disadvantages and in each case the mechanism of integration differs.

In 1. integration of the Powerplant Control system with the Powerplant itself ultimately consists of mounting the control unit on the engine, incorporating the various pneumatic sensors into the package, and simplifying the Hydromechanical components by the use of suitable algorithmic techniques. This method can offer significant cost, weight and complexity reductions depending on the degree of integration that can be tolerated. However, with modern high speed aircraft the means of providing sufficient cooling to obtain the requisite reliability targets poses a significant obstacle to immediate application (1).

In 2. the integration of Flight Control and Powerplant Control is achieved by appropriate exchange of information between the systems. Information exchange can be provided by data bus links or direct connections and further integration can be provided at the man-machine interface by suitable design of controls : Throttle Box and Control Column.

Integration of the systems is seen as a method of achieving benefits in the areas of pilot workload reduction and weapon system performance improvement.

In 3. the powerplant control system and the aircraft systems can be integrated using the data bus as a suitable medium for information exchange, enabling an interaction between systems for mutual benefits.

It is this latter aspect of integration that will be further discussed in this paper. However it is likely that future aircraft will be designed to encompass the advantages of all 3 aspects.

The mechanism for integration of the powerplant and the aircraft will be based upon a system for control of the aircraft Utility Systems which is based upon extensive use of digital data transmission systems (2).

UTILITY SYSTEMS

The Utility Systems of the aircraft have been classified as those relating to the base mechanical functions such as Fuel Gauging and Management, Environmental Control, Hydraulics, Secondary Power, etc. Work conducted at British Aerospace, Warton has led to the proposal of an Integrated Computing System operating on a standard multiplex data bus as shown in Figure 1.

In this diagram the Utility Systems Management Processors perform the tasks of :

- * Data Acquisition from connected sub-system sensors.
- * Performance of control and self test algorithms.
- * Control of Power to connected sub-system loads.

The Processors form a link to other data bus systems on the aircraft and provide a means of interfacing the essentially individual mechanical components and elements of control into an aircraft architecture based on serial digital data transmission and using sophisticated techniques for information presentation in the cockpit.

The interconnection of aircraft data busses enables data to be transferred between major system blocks eg Flight Control, Avionics, Cockpit, Weapons, etc whilst retaining adequate separation of systems to allow the application of well proven methods of procurement and testing.

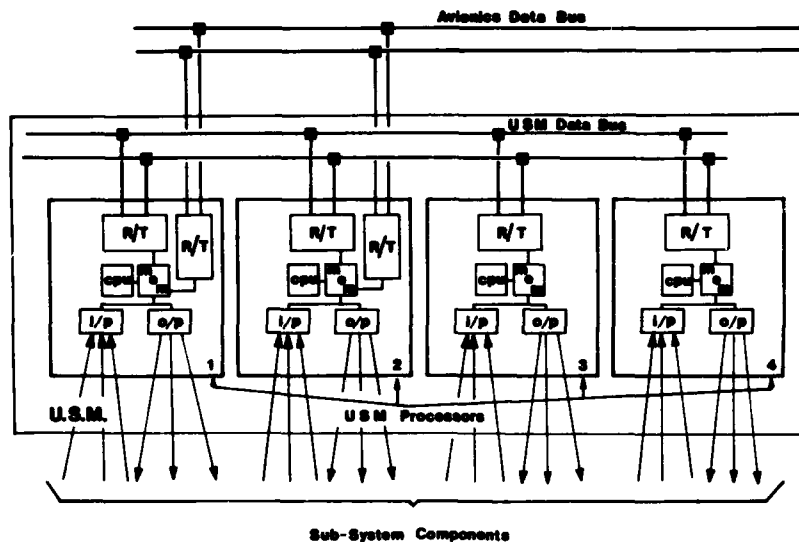


FIG. 1. UTILITY SYSTEMS MANAGEMENT

PROGRESSIVE INTEGRATION OF POWERPLANT CONTROL

Since the majority of the constituent systems of Utility Systems Management are essential to the continued and safe operation of the aircraft Utility Systems Management provides a sound basis for the integration of Powerplant Control into the airframe system architecture in such a way that progressive changes to accommodate technology improvement and increasing integration can be achieved with minimum disturbance to the systems configuration.

In Figure 2a, for example, an analogue Powerplant Control System is shown connected to the Utility Systems Management Processors in such a way as to use the analogue and discrete interfaces to provide a means of integrating with a digital aircraft system.

The provision of spare stub coupling units on the data bus will allow a full digital control system with the appropriate remote terminal hardware to be connected to the utility Systems Data Bus. This digital control unit can be either airframe or engine mounted depending on the size and complexity of the unit and the likely operating environment of the aircraft (1).

In this way the aircraft manufacturer can design a project configuration to minimise development risk by selecting current technology for engine control, whilst allowing the option of progressive updating at minimum incremental cost. For example one can update from airframe mounted analogue to airframe mounted digital to engine mounted digital with minimum change to wiring as illustrated in Figure 2 a, b and c respectively.

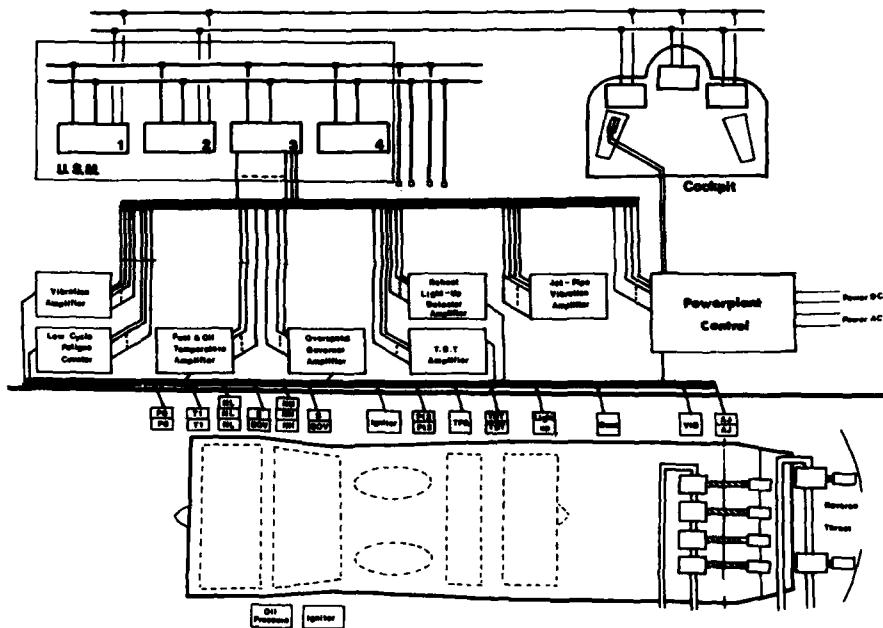


FIG. 2a. INTEGRATION OF AN ANALOGUE POWERPLANT CONTROL SYSTEM

In Figure 2a an analogue engine control system is shown which comprises a Powerplant Control Unit and a number of ancillary units. These items are connected to the engine by a large number of wires. To provide an interface with a data bus type system implementation the items need to be connected to Utility Systems Management with a similarly large number of wires.

Figure 2b shows a system in which the functions of the ancillary units are integrated into a single Powerplant Control Unit.

This retains the analogue, hard wired connections to the engine, but can be connected to Utility Systems Management in conventional analogue fashion (shown in broken lines at "a") or alternatively, and preferably, using the stubs onto the Utility Systems data bus (shown at "b").

In the diagram the distance "d" represents the distance between the powerplant connector interface with the airframe and the Powerplant Control Unit. This wiring represents a weight penalty of approximately 2 Kg/m which can be reduced by locating the Powerplant Control Unit close to the engine. In Figure 2c the Control Unit is shown integrated with the Powerplant, thereby reducing the majority of the data exchanges to the Utility Systems Data Bus connection.

Further integration and the use of digital techniques has been shown to allow a significant simplification of the engine hydromechanical components with consequent reduction in weight and cost (1).

BENEFITS

The benefits of even this limited amount of integration are significant in terms of the flexibility allowed to the airframe designer in his freedom to make early design decisions, whilst allowing for technology improvements with little capital investment at the beginning of the project.

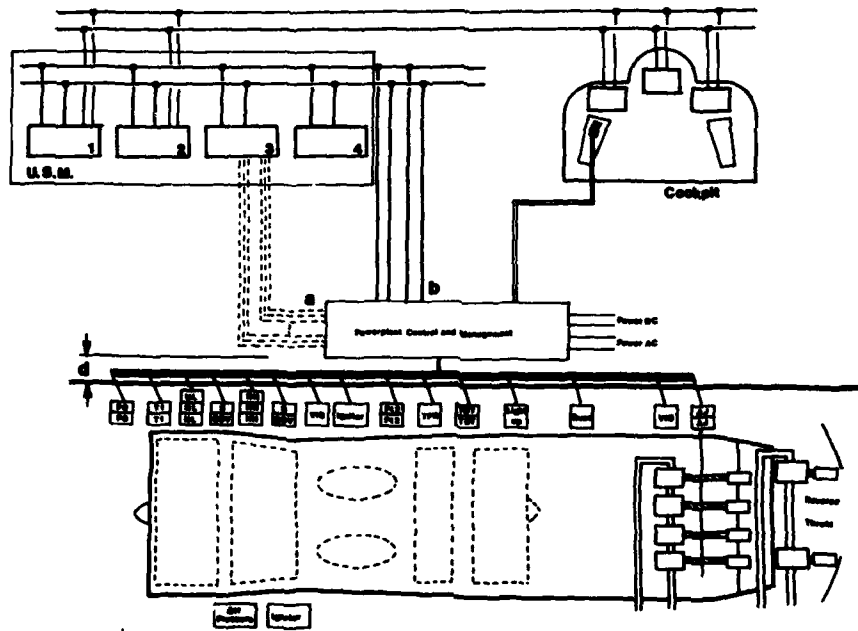


FIG. 2b. INTEGRATION OF A PARTIALLY INTEGRATED POWERPLANT CONTROL SYSTEM.

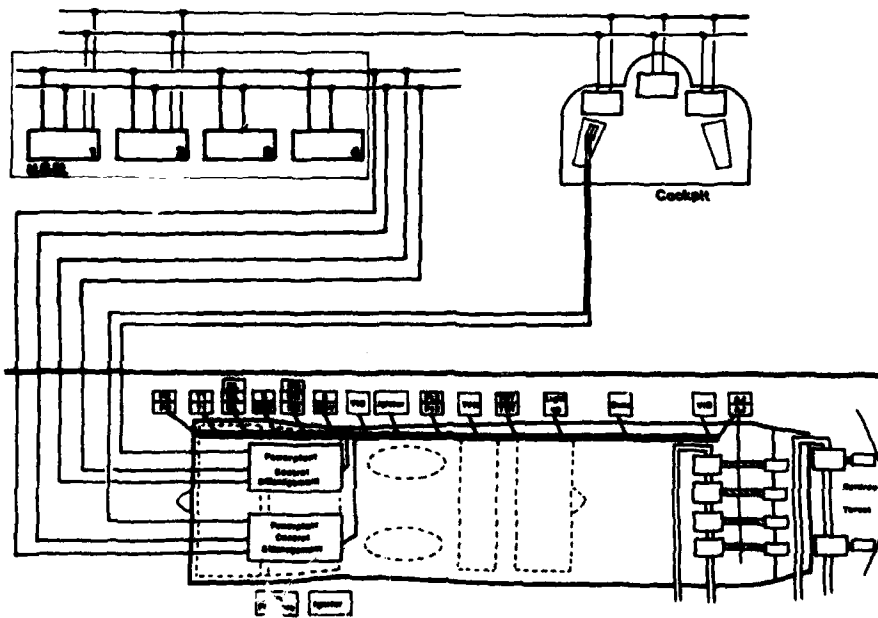


FIG. 2c. INTEGRATION OF A TOTALLY INTEGRATED POWERPLANT CONTROL SYSTEM

Available also are the benefits accruing from integration - those associated with weight, cost and performance configuration (1).

- Weight Reduction - resulting from reduction in wiring and connectors on the airframe and from simplification of hydromechanical components on the engine. This has been estimated to save 170 Kg for one particular configuration (1).
- Wiring Reduction - mainly arising from mounting the electronic control hardware on the engine and continuing the data bus connection across the engine/airframe interface. This has been estimated to reduce the number of individual wires connecting the airframe to the powerplant wiring by a factor of 15 (1).
- Cost Reduction - arising from the simplification of hydromechanical components which currently contain many complex casting and machining operations in their manufacture.
- Maintenance - improvements are expected from the ability to communicate with a sophisticated Maintenance Monitoring System which will reduce diagnostic time. Simplification of fuel system components should result in reduced potential for wear and manufacturing error with consequent reduction of the need for adjustments, a reduction in engine running hours, and, as a direct result of improved reliability a reduction in the need to remove components with long change times.
- Development Flexibility - arising from the ability to gain access to pre-set datum values in the control system by means of data entry devices eg the cockpit keyboards or Maintenance Data Panel Keyboard. This means of changing datum values with the control system hardware in situ can reduce engine change time, particularly if values need to be changed during engine runs.
- Reliability - improvements resulting from a simplification in the engine fuel system components. This combined with improved electronics and sensors has been estimated to provide a 5 fold improvement in MTBF (3).
- Safety - reduced probability of consequential hazardous effects following single engine failures by use of a suitable automatic aircraft load shedding routine.

DEVELOPMENT OF THE INTEGRATED SYSTEM

The integration of the Powerplant and its control system into the aircraft Utility Systems architecture enables the system designer to take advantage of data within the system or to envisage data that can now be made available to improve his overall system design.

For example it is possible to obtain information about the power offtake load taken from the engine by inspection of the pilot's actions, the Utility Sub-systems and the Avionic Systems during the course of a mission.

Monitoring of pilot selections and Avionic Equipment mode selections, together with its own knowledge of internal control functions leading to automatic load selections allows USM to continuously monitor the state of connected electrical loads.

As an example, consider the system shown in Figure 3 which is representative of a typical combat aircraft total system fit - Avionics, Weapon Systems, Countermeasures, Radar, Utilities etc. Each of these sub-system classes will contain loads which vary, either continuously or discretely, thus presenting either smooth variations or step changes in connected load. Examples of this latter mode of operation are De-Icing or Electronic Countermeasures which impose large step electrical loads at certain stages in the mission according to atmospheric conditions or mission requirements.

On current powerplant installations it has not been practical to measure offtake shaft torque, therefore the operation of engine Bleed Valves, for example, is usually scheduled with altitude and airspeed but cannot be modified by offtake power. As a result it is often the case that the bleed valves are opened at a fixed point in the flight envelope, which leads to an unnecessary restriction in engine power by premature air bleed selection. Utility Systems Management, with its access to all systems connected to data busses, can obtain information related to mode selections and pilot selections as well as automatic operations that can be used to determine a sufficiently accurate estimate of total connected load throughout the mission. This can, in turn, be used by the Engine Control System to modify fuel flow or bleed valve schedules as necessary. This transfer of information is shown diagrammatically in Figure 4.

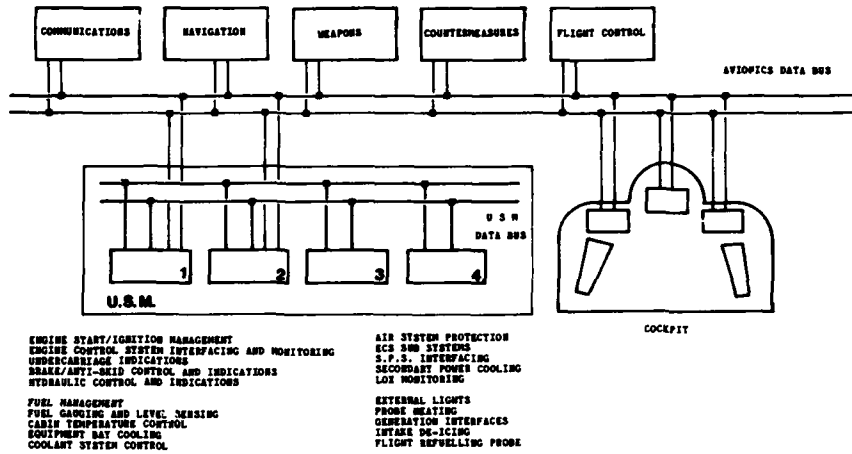


FIG. 3. EXAMPLE OF AN INTEGRATED AVIONIC SYSTEM

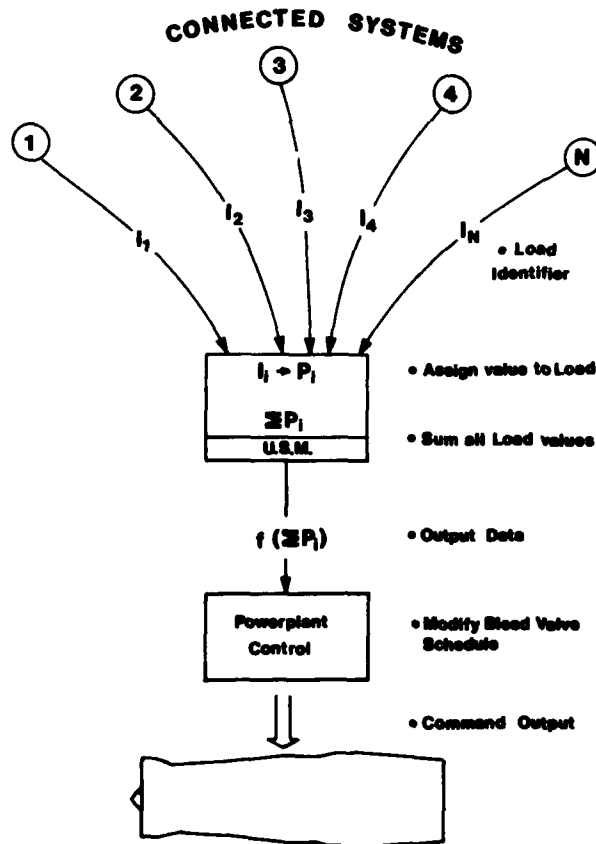


FIG. 4. FLOW DIAGRAM FOR CONNECTED LOAD DATA

The connected systems 1, 2, 3, 4, ..., N are assigned an Identifier I_i ($i = 1, \dots, N$) which is received by Utility Systems Management from each connected equipment when that equipment is considered to be ON or in an active condition. Some equipments may require more than one identifier to indicate which mode they are selected to if mode selection significantly affects their load characteristics. Utility Systems Management assigns, for each I_i , a load value P_i , which is calculated to take into account the nature of the load and its effect on the generation system. This calculation will include Load Power, Reactance, Nature - continuous or intermittent, and may be entered into a look-up table in the Utility Systems Management software.

Utility Systems will then calculate a sum of all connected loads, $\sum P_i$, and output a data value to Powerplant Control which is a function of total connected load - $f(\sum P_i)$. This total can be made to represent air bleed and hydraulic offtake demands as well as electrical loads. The powerplant Control System will then use this data to perform whatever functions are required to maintain efficient engine operating conditions in the presence of the connected loads and other external conditions. This action may involve operation of Bleed valve, scheduling of Inlet Guide Vane position, or modulation of Fuel Flow.

This same information can be used in a "Load Management" function to perform a selective load shedding action in the Single Engine failure case: Rapid, automatic load shedding can reduce the impact of the transfer of offtake load to the remaining engine.

As a result of this action the following additional benefits can be expected:

- * Fuel Savings - resulting from the use of system data to optimise engine control laws both at the design stage and also continuously during engine operation.
- * Performance Benefit - resulting from the ability to design the engine and its control system laws to take advantage of a knowledge of connected offtake loads and to protect the engine from the effects of sudden changes in load and hence maximise the useful flight envelope.

FLIGHT CONTROL INTEGRATION

Aerodynamic and configuration studies carried out at Warton have shown that integration of flight control and powerplant vectored thrust operation is essential in situations where aerodynamic control surfaces have little effect, eg Ultra Short Take Off, Hover, Vertical Landing and, to some extent, in Post Stall Manoeuvring. It is achieved by arranging for the flight control system to demand changes in the magnitude and direction of the thrust vector(s) in a co-ordinated fashion in response to pilot demands utilising stick, pedal and nozzle deflection control with appropriate control functions. Combat manoeuvrability can be similarly enhanced, and engine handling should be improved in poor intake airflow conditions, eg very high incidence.

Independent manual flight and engine control in all flight modes is only practical on aircraft such as the Harrier because of the simple nozzle vectoring system (rotation in one plane). Even in this case the low speed operation is at the expense of a high pilot workload requiring the use of five independent controls. Separation of flight control and powerplant control is expected to result in an unacceptably high pilot workload in future aircraft employing thrust vectoring in more than one plane. Some attractive configurations may well then be brought into practical consideration by judicious blending of automatic and manual control eg twin engine, tilting nacelles with deflecting nozzles operating in two planes.

Preliminary aerodynamic studies and simulation work have shown, however, that even aircraft with these particularly severe control requirements can be confidently proposed (4).

For future projects it is expected that data exchanges between Flight Control and Powerplant Control Systems will allow the aircraft to be handled equally confidently in combat as in the transition from airborne to jet-borne flight.

Work is in progress at British Aerospace in conjunction with Rolls-Royce to further examine the aspects of tightly integrated systems. This work will be supported by extensive simulation activity with models of various engine and airframe configurations aimed particularly at Advanced STOVL aircraft.

The integrity aspects of the Powerplant Control System are receiving attention particularly with regard to providing fault tolerant architectures with emphasis on use in integrated systems (5).

CONCLUSIONS

Data is available in Utility Systems Management which can be used to obtain an accurate indication of total offtake load and can be used to allow prediction of increased demand. This information can be used to allow the engine to operate with more precise control of air bleed valves to obtain better control of thrust throughout the flight envelope.

The current implementation of Utility Systems Management can be used to obtain the requisite information and perform the calculations provided that the information is made available in the connected systems.

The ideal, totally integrated system is shown in Figure 2c in which an engine mounted, twin lane, fault tolerant powerplant control system is connected by data bus to a Utility Systems Management System for intercommunication with aircraft systems and flight control with a directly connected thrust demand unit.

This system configuration allows full interaction between the Powerplant Control System and the airframe systems using the data bus, on which data interactions can be used to the mutual benefit of all systems.

The potential benefits are summarised in Figure 5 which shows from which aspect of integration they are derived.

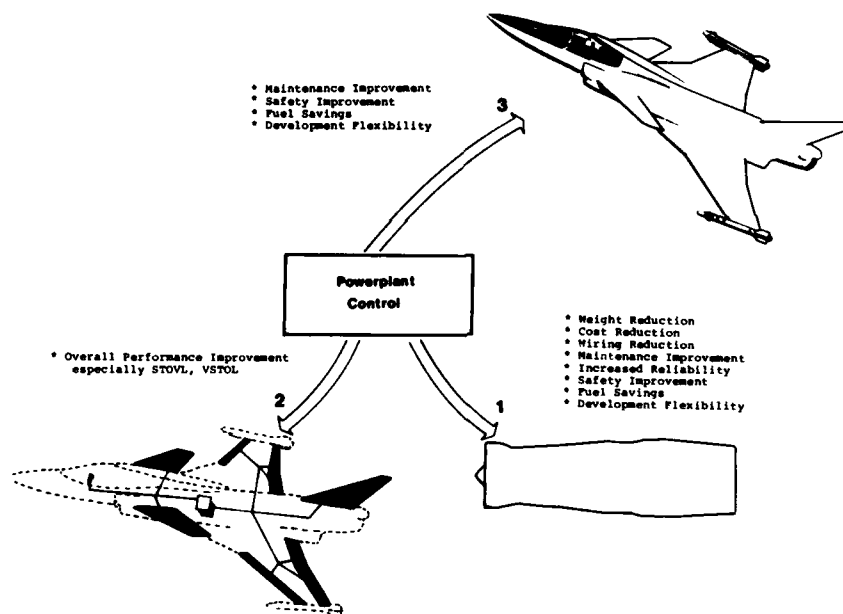


FIG. 5. POTENTIAL BENEFITS OF VARIOUS ASPECTS OF INTEGRATION

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FULL AUTHORITY DIGITAL ELECTRONIC ENGINE CONTROLS AND
THEIR INTEGRATION WITH FLIGHT CONTROL SYSTEMS IN VSTOL AIRCRAFT

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SUMMARY

The paper describes the architecture and construction of the system being applied to the Pegasus engine and aimed at the AV8B application. The system's architecture was particularly determined by the requirements of a single engine VSTOL aircraft. It is essentially a dual-dual system. This architecture was chosen to provide reliable and positive detection of failure with rapid reaction and no degradation of performance. The paper discusses the likely extension of this system to the control of plenum chamber burning (PCB).

The system includes a data bus terminal in each lane of the system. This provides a communication path between the engine control and EMC system in the aircraft. A similar terminal could provide a path through which the operation of the flight control, the weapon control and the engine control systems could be integrated. This integration is required because of interaction between the operation of the engine and the release weapons, the attitude of the aircraft and the mechanics of these interactions and the principles employed in the mitigation of their effects.

1.0 INTRODUCTION

This paper addresses the particular case of VSTOL aircraft using vectored thrust. DSIC has supplied a FADEC for flight evaluations on a Pegasus engine in a single seat, single engine Harrier VTOL aircraft. (1), (2), (3), (4). This flight demonstrator system (5) is described initially followed by the definitive system currently being designed and built. The definitive system is aimed at the AV8B application of the Pegasus engine.

2.0 THE PRESENT SYSTEM

The present engine control system on the Pegasus engine (6) is a hydromechanical control, with an electronic temperature limiter, and has an emergency system fitted. The emergency system is required because any failure in the engine control itself which leads to loss of thrust or engine shut-down would have a catastrophic effect for the aircraft. The emergency system was fitted with the control from the first flight and, as yet, no accident of the aircraft has been attributed to control failure.

The emergency system itself is very simple. The control of the engine is by modulation of a valve coupled to the shut-off valve and directly operated by the pilots lever as shown in Figure 1. A changeover valve by-passes and disconnects the main metering valve in order to engage the emergency system. It also disconnects the limiter by-pass valve and back-flow is prevented by a check valve in the downstream line from the metering valve delivery. In normal operation the shut-off valve is fully open and the manual flow control is therefore set to a flow determined by the lever position. For changeover from the normal operation to emergency operation the manual flow control is returned to the idle stop by means of the pilot's lever and the changeover valve is operated by means of a switched solenoid. Once the changeover valve has moved to the emergency position then the throttle lever can be advanced to an appropriate thrust setting. This procedure is required in order to prevent surge were the changeover valve to operate while the manual flow control was selecting a high flow.

Figure 2 shows the block diagram for the system with the changeover selection from normal operation using the hydromechanical control plus temperature limiter to the manual flow control operation. The electronic limiter has a multiple datum selection and muting switch.

The nozzle setting is commanded from a separate lever in the cockpit and is positioned by an independent follow-up servo.

3.0 DEMONSTRATOR SYSTEM

The demonstrator system is shown in Figure 3. It uses two electronic control lanes, one of full capability and one of reduced capability. The system normally operates using the main digital electronic control but switches automatically to the emergency control when the main electronics fails. The predicted probability of both the electronic controls failing is very much lower than the probability of failure of the hydromechanical control which they replace. Frequency of reversion to the manual flow control should therefore be lower than with the existing system but, for safety, the manual flow control was retained. The main areas of interest in the design were the definition of the capability of the partial control lane and of the production configuration.

The system comprises two major components as shown in Figure 4. The hydromechanical section is adapted from the current system. As shown in Figure 5 it retains the same basic hydromechanical circuit schematic as the existing system.

However, an electro-hydraulic interface is provided with the fuel metering valve which allows either of the two electronic control lanes to position the valve through a stepper motor interface. The electro-hydraulic interface includes a mechanical P3 multiplier and the electrical command to the system is for WF/P3 (fuel flow divided by CDP).

The mechanical reversionary system is unchanged.

3.1 INSTALLATION

The electronic module shown in Figure 4 is engine mounted, being carried, as illustrated in Figure 6, by a frame cantilevered from the hydromechanical module. It is vibration isolated and fuel cooled as a single unit and the electronics are withdrawn as a single block.

The environment in which the control system is located is extremely hostile. Bay temperatures of up to 150°C can occur for short periods after shut-down, while heat is soaking from the engine into the engine bay. In normal operation, the fuel temperature used for cooling is low, but it can peak to temperatures above 80°C for fairly long periods during some types of operation.

The main area of attention in this design is keeping good thermal paths between the components themselves and the coolant. Particular attention has been given to areas of metal contact within the design of the unit and the thermal washers and gaskets are used in order to secure low thermal impedances at critical points in the design.

The method of assembly, illustrated in Figure 7, uses metal frames which carry pairs of circuit boards. Each board has a copper facing which runs under the components and is clamped between the board and the frame. The frame itself is clamped to the case through which fuel circulates. The fuel passage lies immediately below the shoulder on which the frame is clamped. This arrangement provides the minimum number of interfaces and the shortest possible path between individual components and the fuel. The frame also stiffens the whole assembly and improves its mechanical integrity.

The front section of the case forms an enclosure behind the connectors. EMC filters are mounted on a bulkhead at the back of this enclosure.

The power supply is installed in a second enclosed volume on one side of the unit. This arrangement provides two benefits. First, it excludes from the main electronics compartment interference generated by the switching regulators in the power supply. Secondly it allows high dissipation components to be mounted directly on the case.

Figure 8 shows one of the demonstrator units opened and the modules separated to expose all components. The wiring shown is strictly a feature of the prototype construction. Flexible film wiring would be used in production.

3.2 CONTROL LAWS AND PERFORMANCE

Figure 9 shows the control functions provided by the system. Water modulation is an optional feature.

The control laws provide for steady state control of thrust by governing the fan speed. A range of ratings is used to permit short lift and "wet" engine operation in vertical lift operation. Rating selection is made automatically by logic using several state signals defining configuration and operating mode. Similar logic selects one of up to 7 datums for the T6 limiter. Additional protection for the engine is provided by a non-dimensional NH limiter.

The system provides full transient control during acceleration and deceleration to avoid compressor surge or flame-out respectively. The system provides automatic starting and transient reduction of fuel flow during weapon release to avoid gun-gas ingestion.

The system uses conventional low and high error selection gates to ensure that the correct control loop is closed onto the engine at any time.

3.3 SAFETY & MONITORING

The system provides automatic failure detection and response to failure. Where failure is detected the stepper motor drive is inhibited, the output is, therefore, frozen and the signal line to a changeover relay is activated. This relay sets in train the necessary actions to effect reversion to whichever system is next to be used.

The main control lane uses dual microprocessors which are cross-compared to provide very high fault coverage. Most inputs are either dual redundant or, as do resolvers, have dual outputs which can be compared using an algorithm such as $\sin^2 + \cos^2 = 1$. The engine speed signals can be compared by using known relationships between HP and LP shaft speeds. The only input variable not monitorable by comparison is the T6 signal generated by an 8-thermocouple harness. Individual couple failures only result in inaccuracy.

The outputs are monitored by position pick-offs driven by the stepper motor. A failure in either the motor or the pick-off is immediately detectable. In effect, the main lane is a duplex sub-system with very high failure detection probability.

The emergency electronic control uses a single microprocessor, single sensors, and software monitoring. This was considered adequate with pre-flight checks on each flight and its being called into use only after the main lane had failed.

Attitude control, as well as lift in vertical flight depends upon the engine. Furthermore, there is no store of energy, such as the rotor in a helicopter, which can be drawn on in an emergency. Neither is it possible to recover aerodynamic control without great loss of height. It follows that the major design objective is that a first electronic failure be detected with very high confidence and that control be restored rapidly and automatically in response to that detection of failure. This, the system is designed to achieve.

The demonstrator system has been evaluated over the full flight envelope of the Harrier, including tests in hovering flight.

4.0 DEFINITIVE SYSTEM

The demonstrator system for Pegasus uses one and a half lanes of electronics. The complexity of the reversionary lane depends on the amount of capability required to be retained after a failure in the main electronics control. In principle, the inputs to the partial lane can be reduced to two and the monitoring of the partial lane can be eliminated completely. Various increasing levels of complexity and sophistication are possible from this base depending upon the degree of capability required to be retained after the first failure. However, retention of mission capability after a first failure and simplified logistics led to the selection of a system using two identical lanes for the definitive system. (Figures 10 and 11).

It uses the same basic methods of procedure as the Demonstrator system but is adapted to use two extended "main" lane controllers in place of the two dissimilar lanes. It embodies two relevant additions to the Demonstrator system. The first of these is a digital data bus interference for communication with other equipment in the aircraft. The second is the addition of an angle of attack input. This is used to reset the control to allow for the reduced surge margins experienced at high incidence. The construction of the system is similar to that described for the demonstrator.

In the longer term, extensive development of the system can be envisaged to match the expected progress in VSTOL aircraft design and operation. It is expected that supersonic versions of this type of aircraft will be designed and built (7). Such aircraft will probably embody plenum chamber burning (PCB). This corresponds to an augmentor system in conventional turbo-fan and turbo-jet engines and provides a significant increase in thrust. The addition of this feature to the engine, giving supersonic capability to the aircraft, will require extensions of the control capability in the future. The control requirements for PCB are similar to those used by DSIC in the control of P850 and Adour engines (8) and (9).

PCB control is essentially the same as a two gutter augmentor system. The dry engine system will require the additions shown in Figure 12. A variable nozzle is required and two metered fuel flows could be involved. The control laws used would be similar to those for normal augmentors with a pressure ratio control of nozzle area and non-dimensional schedules of fuel flow.

Changes required of the system will, typically, be the addition of one input variable and three output channels to each lane. Complete duplication will be mandatory if PCB is used in jet-borne STOVL flight.

Both the demonstrator system and the definitive system use the existing nozzle control and variable guide-vane control as the standard engine. Both also have the same type of hydromechanical emergency system as the existing fuel control. However, the definitive system uses repackaged hydromechanics with the electronics mounted separately on the fan case. The new configuration uses the same drives and mounting points as the existing system but exhibits a significant reduction in weight and complexity.

5.0 FLIGHT CONTROL/ENGINE CONTROL INTERACTION

There are many manifestations of interaction between the airframe and the power plant operating states. They occur both in conventional wing-borne flight and in jet-powered hover and are both static and dynamic. They all reduce to changes in the net aerodynamic force vector on the aircraft and to changes in its pitching moment.

Eccles et al (10), reporting work on this topic conducted by Dowty Group and Smiths Industries in 1974, identified the following as candidate applications for integration of flight control and engine control in VSTOL aircraft.

- a. matching nozzle angle to flight condition
- b. offsetting changes in static margin following stores release, configuration changes or fuel burn
- c. increase in permitted c.g. range
- d. transient increases in normal force during turning manoeuvres
- e. cancellation of transient lift-loss following positive pitch demands
- f. improved turn performance when entered from speeds well above the optimum and when emergency loss is not important
- g. simplification of pilot controls in the pitch plane through non-interacting control of forward and vertical speed
- h. reduction in trim flows bled from the engine to the reaction controls in hover.

Significant returns in overall performance were predicted for many of these features.

The Pegasus definitive system already provides for three other interactions:-

- a. transient reduction of fuel flow to prevent engine surge following release of missiles or discharge of guns
- b. compensation for reduced engine surge margins caused by inlet distortion at high angles of attack
- c. dynamic compensation for the effects on thrust of the engine bleed required for the reaction control of attitude in the hover.

Each of these three functions is proper to the engine control itself and responds to signals derived from the aircraft state. Some of the other interactions may be located in the flight control system (FCS) rather than the engine control system (ECS) proper.

A clear thrust example of the latter is the determination of the best combinations of thrust and nozzle angle for any particular manoeuvre. These can be defined by mapping techniques entered from the demanded thrust vector determined by FCS. The FCS could then command the engine power and the nozzle in an open loop fashion.

In the absence of a direct measure of thrust, this simple arrangement will be sensitive to changes in engine performance. It may be preferable to place them within a closed loop control of flight path. However, the principle that the FCS independently commands the nozzles and engine power remains true.

The mapping technique is essentially one which is static or quasi-static. For dynamic interaction it may be less easy to identify where best the control should be located.

For instance, in configurations using "hot" and "cold" nozzles, differential nozzle modulation would produce a static pitching moment which could be used to trim the aircraft and reduce the bleed taken to the reaction controls from the engine. It is questionable whether this control should be direct from the ECS, where the other bleed effects are handled, or indirectly via the FCS. A compromise solution would be to permit a limited authority differential nozzle trim and thrust reset to be commanded by the ECS with the prime control from the FCS. The purpose of this discussion is not to advocate the use of differential nozzle operation in order to reduce bleed but to explore the implication of the possibility. The method chosen might also be used to reset static margin following configuration changes, or weapon release.

All jet powered VSTOL aircraft do not benefit from the positive ground effects experienced by helicopters. Indeed, the effects of re-ingestion can result in a negative ground effect with thrust reducing as the aircraft approaches touch-down and the re-ingestion effect is amplified. This effect could be alleviated by a simple addition to the ECS which would increase the demanded thrust before touch down in response to some height determining input. The same function could alternatively be located in the FCS and operate on the engine power demand. The major factors affecting the choice would be the severity of the effect and the general redundancy philosophy of the aircraft. A determining factor would be the need for the function combined with an ability to land the aircraft in some emergency mode with the full FCS inoperative.

Aircraft with positive static margin require an increase in downwards lift on the horizontal stabiliser in order to increase the pitch attitude of the aircraft. This results in a transient "sinking" effect immediately prior to generating the desired pitch rate. This effect could be removed by a combined downwards nozzle deflection and compensating thrust increase which restore the total aerodynamic force to its value prior to the application of the pitch command. This effect is analogous to the use, in a helicopter engine control, of collective pitch as an anticipatory signal to generate an immediate engine response to rotor load and avoid the transient loss of rotor speed which is inevitable with a conventional rotor speed governor.

This now raises the issue of the form of the interface between the FCS and the ECS. It could take the form of an autothrottle. If so, the response of the servo might be too slow satisfactorily to operate in the anticipatory mode. The choice is then either to by-pass the conventional power lever by a fast-acting limited authority trim command generated by the FCS or to feed the elevator signals to the ECS in the same way as is collective pitch in a helicopter engine control. The elevator signal used may originate in the FCS. The issue involved lies in the responsibility for the dynamic response of the arrangement and will probably determine that it lies within the FCS, the elevator signal being shaped to produce best aircraft performance. The ECS would incorporate a protective filter against fault conditions in the command.

A further practical reason for supporting this outcome is the difference in development timescales between engine and airframe. Engine development normally starts well ahead of airframe development and the definition of the relationship between elevator demand and thrust vector response is only likely to occur late in the engine control development history. Changes to engine control software at this stage would be undesirable whereas no such problem would occur were the changes to be implemented in the FCS where control development is probably still in process.

Direct control of thrust will not be possible if the aircraft uses non-interacting control of forward and vertical speed. There will be no place for an autothrottle in this type of operation and an entirely electrically thrust command incorporating the pitch anticipation term would be transmitted from the FCS to the ECS.

At fixed fan speed the operating of PCB in a Pegasus type of engine increases the thrust from the front ("cold") nozzles without changing the thrust from the rear nozzles. The result, unless the nozzle thrust lines all pass through the aircraft c.g., is a change in pitching moment which must be offset either by coordinating setting of PCB and fan speed or by a change in longitudinal trim controls. The trim control will be on the elevator in conventional flight and on reaction nozzle flows in hover. The trim control adjustment may be initiated in one of at least three ways:-

- a. by the command to the PCB control
- b. by the PCB control output
- c. by the aircraft response to the PCB pitching moment.

Considering each in turn it is seen that alternative (c) has slower response than either (a) or (b) while alternative (a) may anticipate a thrust change which does not occur. Alternative (b) introduces a data feed from ECS to FCS with the minimum data content that PCB is lit and hence the data path between the FCS and ECS will be bi-directional and may need a data rate appropriate to the PCB modulation bandwidth.

Other engine state signals will need to be communicated to the FCS. The actual engine thrust may depart from the expectations of the FCS because of the operation on the engine of some protective limiter such as a non-dimensional speed or a turbine blade temperature limit. The engine control ceases to behave in a linear fashion and the FCS can only function correctly if the existence of a fixed thrust condition is signalled to it. There is, therefore, a need to transmit to the FCS a range of information on the operating state and the controlling loop in the ECS.

This "reverse" data flow can be used in other ways. It has been found helpful, on large commercial fan engines, to display to the pilot the power demand to the engine which corresponds to the actual power lever position. The pilot may then position the lever to generate the engine power he is seeking rather than manipulate the power level until the actual engine power coincides with the desired value. The pilot then operates directly on the fast loop leaving the ECS to handle the slower engine response. The removal of the inter-action between the loops results in a significant improvement in control performance. The method is equally valid if the pilot is replaced by the FCS and an autothrottle, or other intermediate control interposed between the FCS and the ECS.

6.0 FLIGHT CONTROL AND ENGINE CONTROL INTEGRATION

"Integration", in the context of this paper has the meaning of making the system complete in concept rather than integrating the hardware into a single entity. The means by which this end is achieved is exchange of data between the various systems to be integrated so that each has available to it all the data it needs properly to coordinate its actions with the others. The data is exchanged over a serial digital data bus network.

Figure 13 shows how the longitudinal control functions could be related in a VSTOL or STOVL aircraft. The only inter-connections shown are between individual systems and the FCS. Figure 14 shows the data flows between the various functions as identified in the previous section. It would appear logical, with an engine mounted control, to combine the command of the nozzle vector actuator with the engine control hardware. Integration of this "Thrust Control System" with the other systems could then take a form similar to that shown in Figure 15.

The integration of flight control and engine control will involve the bi-directional exchange of data between engine control and flight control systems. This may take place over a general data bus or over a dedicated data system. In either event, more will be involved than simple data exchanges and more issues than have been raised in the previous section.

One such issue depends on the levels of redundancy in the FCS and ECS. It can be expected that the FCS will be double failure surviving (say quadruplex) while the engine control is most likely to employ some form of dual redundancy. If the data bus systems are also redundant there is a clear but complex issue to be resolved on the control of the system configuration following progressive failure, the routing of information and the prevention of fault propagation. The previous section indicated how this problem may be mitigated by careful choice of location for a particular control function and by appropriate limitations of control authority. However, the main question cannot be addressed within the limits of this paper.

A second issue is the "testability" of the system and the verification of its operational status. This will be an important consideration in the integration of the systems. The methods outlined will simplify the problem but it will be essential to conduct detailed studies of this aspect of the integration of the two systems for each application.

It is assumed that the FCS and ECS will be digital. A third, and major issue, will be the configuration control of software in two independent but highly interactive development programmes, one of which is controlled by the airframe manufacturer. This, rather than the performance issues discussed earlier, may well be the pivotal consideration in many instances.

7.0 CONCLUSION

This paper has described a digital control system for a VSTOL engine. It has described the interactions between engine and airframe functions and shown how major performance improvements could be secured. It has also indicated how performance, safety and interface responsibilities can affect the partitioning of functions between the PCS and ECS. Finally, it has indicated some of the major considerations which the paper does not address.

8.0 ACKNOWLEDGEMENTS

The Author wishes to acknowledge the contributions of his colleagues to the work which he reports in this paper. He also thanks the directors of DSIC for permission to publish the paper although the views expressed are entirely his own. Some of the work described in this paper was supported by the UK Ministry of Defence.

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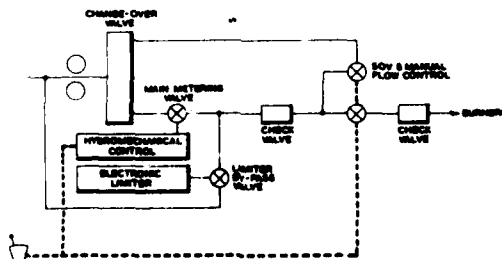


Fig. 1 Hydromechanical circuit diagram

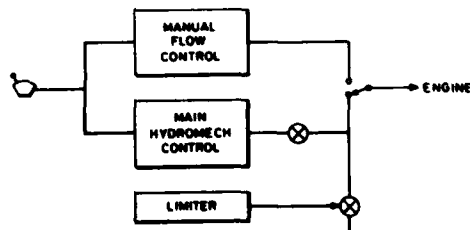


Fig. 2 System block diagram

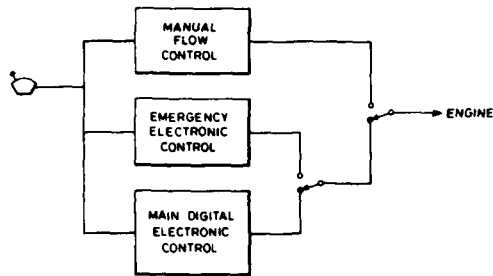


Fig. 3 Phase 1 demonstrator system

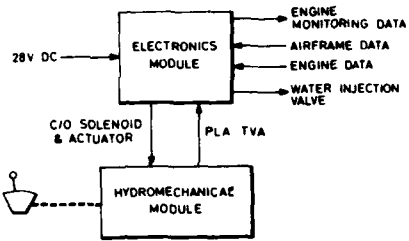


Fig. 4 Pegasus control system components

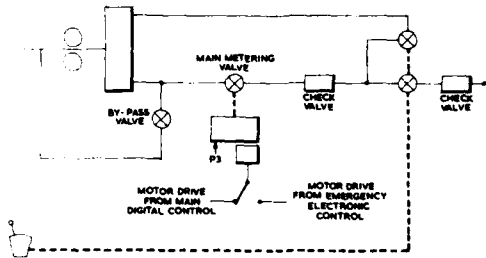


Fig. 5 Electronic automatic change-over

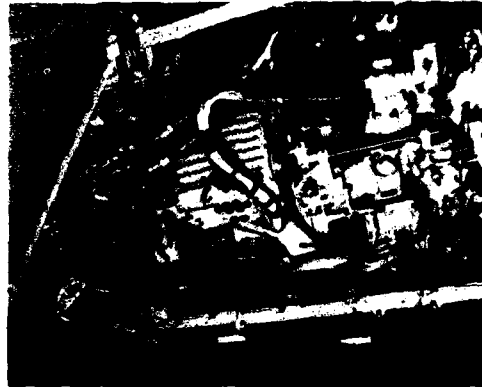


Fig. 6 Engine mounted electronics module

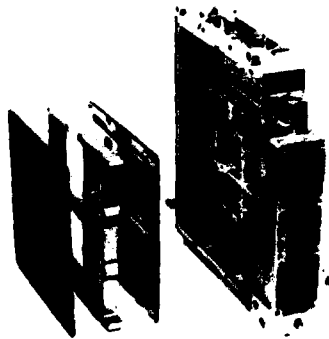


Fig. 7 Reversionary lane construction

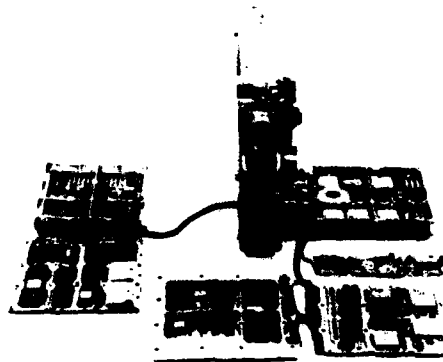


Fig. 8 Main lane internal views

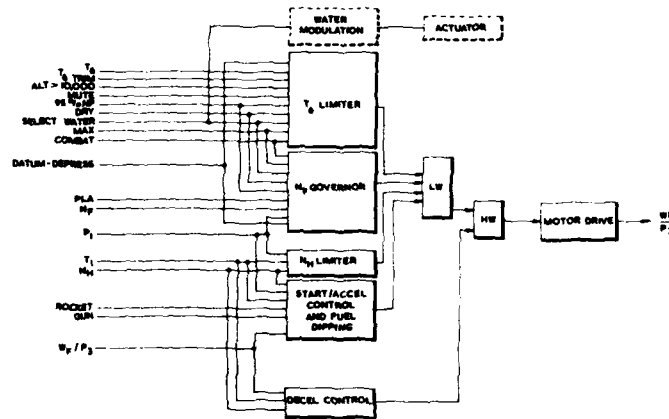


Fig. 9 Digital electronic controller schematic

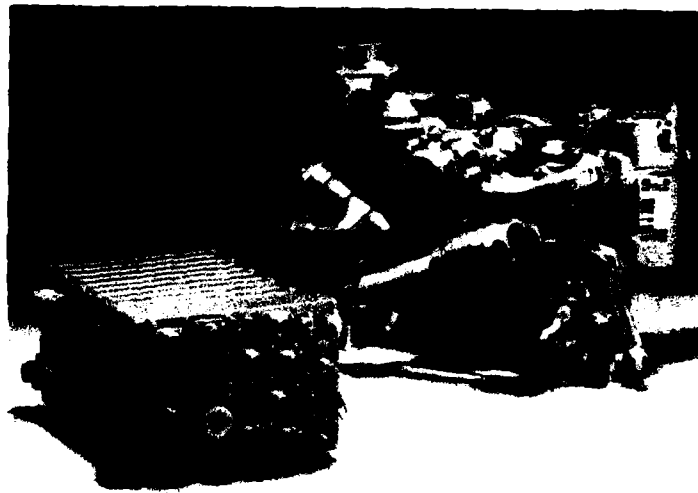


Fig. 10 Definitive system

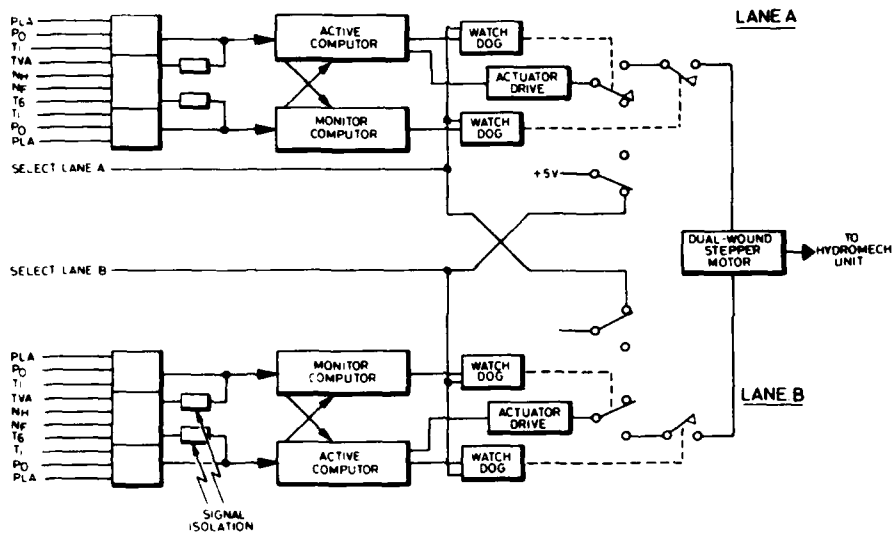


Fig. 11 Two lane electronics configuration

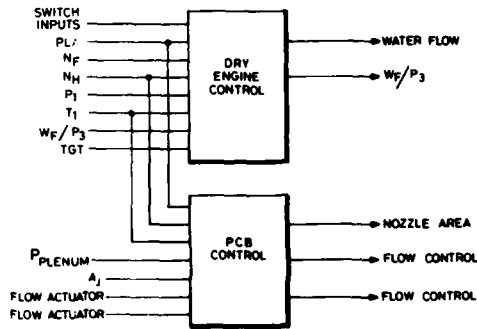


Fig. 12 PCB control schematic

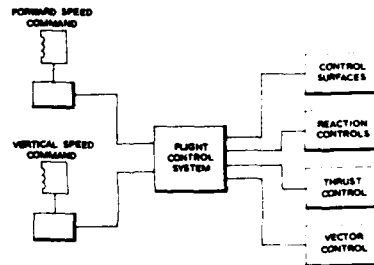


Fig. 13 Possible longitudinal control architecture for VSTOL aircraft

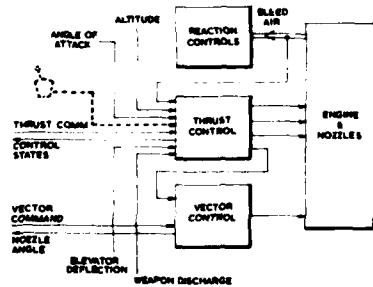


Fig. 14 Data exchanges

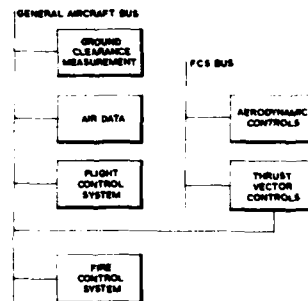


Fig. 15 System interconnection



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DESIGN AND DEVELOPMENT
OF THE
MULTIFUNCTION FLIGHT CONTROL REFERENCE SYSTEM

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SUMMARY

Reliability, redundancy, and survivability are key issues as integrated requirements for flight control, fire control, and propulsion control are developed. These integrated control systems require dependable sources of inertial measurement data. Current inertial sensors, however, are expensive to acquire and maintain, dedicated to specific systems, and are not designed to meet integrated control reliability, redundancy, and survivability requirements. The Multifunction Flight Control Reference System (MFCRS) concept uses a minimum number of inertial sensors in a survivable configuration to provide inertial data for flight control, navigation, weapon delivery, cockpit displays, and sensor stabilization. Because of advantages in survivability, life cycle cost, size, and performance, the MFCRS program was initiated to verify, through flight test, the key issues of redundancy management and flight control. A redundancy management system based on parity equations was designed. Sensor implementation consists of two skewed and dispersed sensor clusters. Each cluster is an orthogonal triad of colocated inertial quality ring laser gyros and accelerometers. Testing showed noise levels were higher than predicted. While the noise had little affect on the navigation performance of the baseline hardware, additional filtering was required for MFCRS to prevent false alarms and high frequency actuator response. This filtering affected the flight control stability and performance and caused the flight control design to be modified. A key lesson learned is that integration of inertial data for fire control, flight control, and propulsion control will require close coupling and coordination between functional groups to resolve performance conflicts and compromises. Testing to date has not shown any basic flaws in the multifunction concept, and flight test is scheduled in late 1983.

1. BACKGROUND

Improvements currently being developed for advanced fighter/attack aircraft include integration of flight control, fire control, and propulsion control systems. Also, flight control is becoming more sophisticated with advances in trajectory control and automatic terrain following/terrain avoidance techniques. As these advanced developments proceed, formerly mission critical functions now become flight critical. The high reliability and redundancy classically associated with flight control must be designed into these integrated systems. Survivability is also a major design factor given the increase of the number and quality of the threat air defense systems. Inertial data is required for all of the advanced flight control techniques and is a key component in many of the integrated systems.

As shown in Figure 1, current operational fighters obtain inertial data from a number of different sources such as the Inertial Navigation System (INS), the Attitude Heading and Reference System (AHRS), the Flight Control Gyros and Accelerometers, and the Fire Control Lead Computing Gyro (LCG). These sensors are dedicated to and optimized for specific functions. Current generation inertial sensors do not, however, meet the reliability and survivability requirements that integrated systems will need. For example, flight control sensors are not considered survivable when the gyros are clustered at a single location near the primary aircraft bending antinode or when the accelerometers are similarly located together at a node. Mission critical sensors such as the INS or LCG are neither redundant nor survivable. This will become an increasingly important problem as the INS outputs are used to generate inputs to the flight control/flight management system to perform maneuvering attack, automatic terrain following/terrain avoidance, or night, all weather control. For these functions, the inertial sensors are in essence part of the flight control/flight management system and should be designed to meet the rigorous flight control safety requirements. Another problem with current inertial sensors is their long reaction time. Current gimballed INS systems require 4 to 6 minutes for warm-up and alignment prior to beginning valid navigation. Current inertial systems are also costly to maintain. This is due in part to the fact that most current systems require complex platform electronics to support the electromechanical gimballed devices. Also, each unique inertial data source requires a dedicated interface which must be maintained through the existence of aircraft intermediate shop support and the training of highly skilled maintenance personnel in each specialty field.

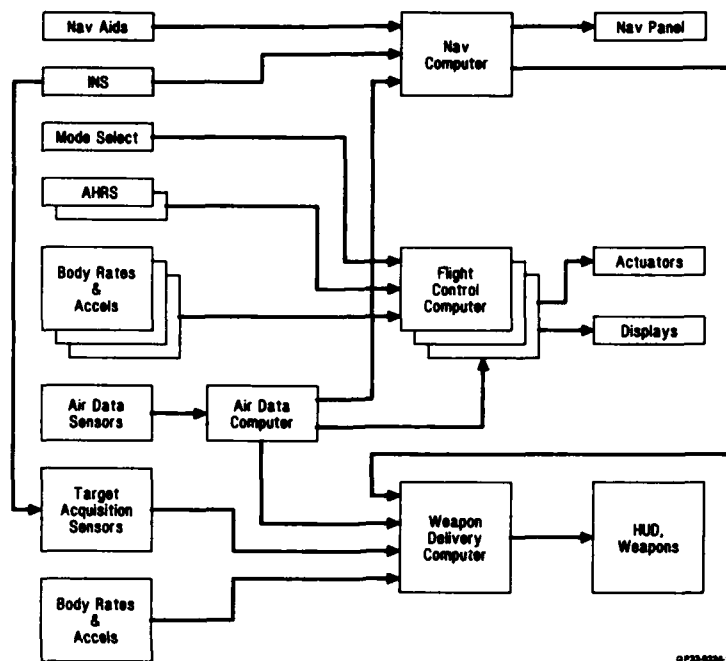


Figure 1. Conventional Approach to Inertial Data Requirements

The Multifunction Flight Control Reference System (MFCRS) concept shown in Figure 2 was developed to solve these problems. (1,2) The MFCRS concept is an innovative approach that uses a minimum number of inertial sensors in a survivable configuration to satisfy the combined inertial data requirements of flight control, navigation, weapon delivery, cockpit displays, and sensor stabilization. One key element of the MFCRS concept is the ring laser gyro (RLG). The RLG in a strapdown configuration provides both the accuracy required for navigation and the dynamic bandwidth required for flight control. The strapdown RLG assembly/cluster is also less complex and more rugged than the current four gimbal inertial platform since the RLG is a solid state device. The other key to the MFCRS is the availability of high speed digital microprocessors. Microprocessors allow the wide variety of functions required of the MFCRS to be calculated and provided in real-time. The processor does fault detection, fault isolation, dynamic reconfiguration, navigation, flight control compensation, and compensation required by the other systems using MFCRS data.

The MFCRS concept was initially investigated by the Multifunction Inertial Reference Assembly (MIRA) program. (1,2) The MIRA program, jointly sponsored by the Air Force Wright Aeronautical Laboratories Flight Dynamics Laboratory (AFWAL/FI) and Avionics Laboratory (AFWAL/AA), identified several potential benefits and payoffs for a projected late 1980's production system. First, reliability and mission success probabilities would be increased for all functions. This is due to the fact the system could provide fail-operate, fail-operate flight control data, and fail-operate navigation and weapon delivery data. The MIRA study also estimated a 21% decrease in life cycle cost. This was based on the mean-time between failures increasing from 240 hours to 1,670 hours and the mean-time to repair decreasing from 2.6 hours to 1.4 hours. Weight would also decrease from 120 pounds to 90 pounds and only 2 line replaceable units would be required instead of 7. The RLG has a faster turn on time than the present gimballed INS and reaction time could be reduced to 1.5 to 3 minutes for gyrocompass alignment and to less than 30 seconds for stored heading alignment. MIRA also predicted a 30% improvement in combat survivability when the sensors were dispersed in two clusters. These benefits and payoffs are summarized in Figure 3. (1,2) These increases in reliability, survivability and redundancy afford the opportunity

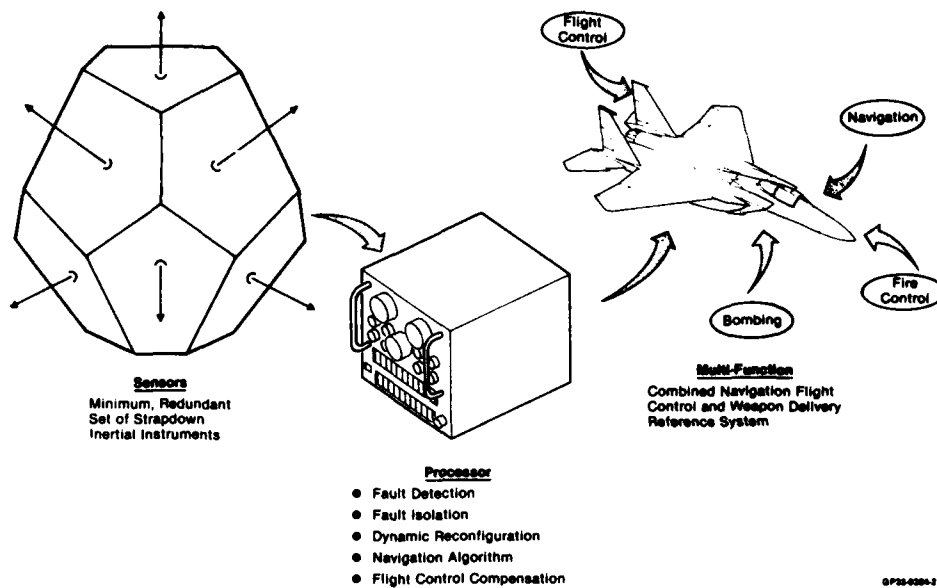


Figure 2. Multifunction Concept

to eliminate aircraft intermediate shop (AIS) maintenance support for inertial systems. They are also required to support the operational readiness of advanced weapon system functions such as integrated flight fire control, terrain following/terrain avoidance, and multi-mode control laws. When the MIRA program was completed in December 1978, several key issues were identified that could not be resolved in MIRA's limited laboratory demonstrations. The MFCRS program was initiated in May 1980 to resolve, through flight test, the key issues of redundancy management of skewed and dispersed sensors and the compensation required to use navigation quality RLG's and accelerometers for flight control reference in an advanced high performance fighter. A contract was awarded by the Flight Dynamics Laboratory's Flight Control Division (AFWAL/FIG) to McDonnell Aircraft Company (MCAIR) to develop and flight test a multifunction unit.

- Increases Reliability and Mission Success Probability for All Functions
 - Two Fail-Operate for Flight Control, Fail-Operate for Navigation and Weapon Delivery from Minimum Sensor Complement
 - 21% Life Cycle Cost Reduction
 - Longer MTBF 240 hr → 1,670 hr
 - Shorter MTTR 2.6 hr → 1.4 hr

Increases Availability and Eliminates AIS
 - Reduces Weight/Volume
 - 120 lb → 90 lb
 - 7 LRUs → 2 LRUs
 - 30% Improvement in Survivability Due to Sensor Dispersion
 - Reaction Time 1.5 - 3 min
 - Supports Advanced Tactical Fighter Sensor Requirements
- GP23-0004-1

Figure 3. Benefits and Payoffs of MFCRS Concept

2. BASELINE DESIGN

The MFCRS program is intended to resolve the two key issues discussed above: redundancy management and flight control compensation. To meet these objectives in a cost effective manner, the MFCRS program modified two existing RLG navigation units built by Honeywell Incorporated for the AV-8B program. The resulting MFCRS design will provide an adequate technology base, validated by flight test, upon which design recommendations for a production multifunction prototype can be made. Figure 4 shows the components of the MFCRS and their location in the F-15 test aircraft. Motion Reference Unit (MRU) A is aligned with the aircraft axes and is located on an avionic shelf in the nose barrel of the aircraft. MRU B is located 2.8 meters aft and the MRU is skewed 60° about its cone axis relative to the aircraft axes. The amount of separation required for survivability was determined in the MIRA program to be at least .76 meters. The limited number of available equipment installation locations on the test aircraft resulted in the large separation for the MFCRS program; however, the compensation developed for this separation will demonstrate a worst case situation. Demonstrating this worst case will provide additional flexibility in locating these components in a new aircraft design. The skewing of MRU B is necessary to provide the redundant inertial information required to perform the two fail-operate redundancy management. Changes to the MRU electronics were required to allow inter MRU communication, communication with the aircraft instrumentation, and communication to the flight control computers. A separate MFCRS unit called the Test Management Panel (TMP) was built to serve as both an interface between the MRU's and the flight control computers, and as a test input/output source for the pilot. Because of safety considerations, a switching unit was also installed to allow selection of either the production aircraft flight control sensors or the MFCRS sensors.

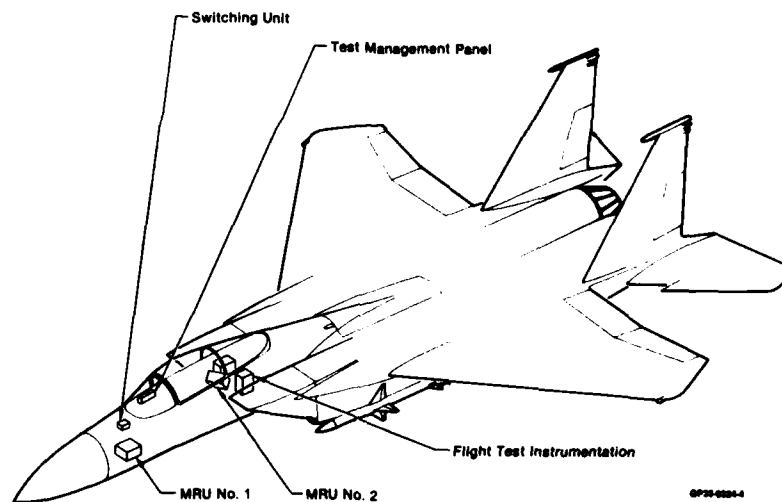
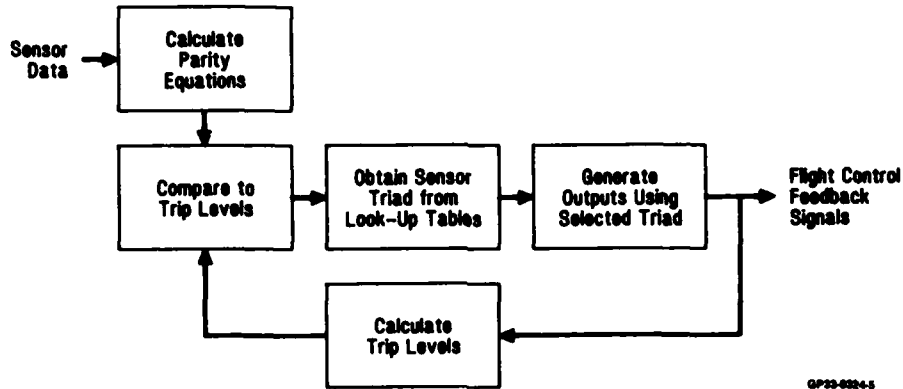


Figure 4. MFCRS Equipment Location

2.1 REDUNDANCY MANAGEMENT BASELINE DESIGN

A redundancy management system that compares the sensor outputs using parity equations was chosen for implementation. (3,4) As shown in Figure 5, the results of the sensor comparison, the parity equation residuals, are compared to trip levels and the results are used as a pointer to select the three best sensors based on stored tables. This approach of parity equations and stored tables has a low processing requirement which allows the existing processor to be used at a 50 Hz processing rate. The two key aspects of the redundancy management system are 1) the compensation to allow sensor outputs to be compared, and 2) the generation of the stored tables. The sensor differences that act as error sources and methods of compensation are shown in Figure 6.



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Figure 5. MFCRS Redundancy Management Logic-Simplified Block Diagram

Error Source	Compensation Method
<ul style="list-style-type: none"> ● Moment Arm Effects ● Static Misalignments ● Dither Noise ● Structural Bending Misalignments ● Bias 	<ul style="list-style-type: none"> ● Moment Arm Compensation ● Electronic Alignment ● Filtering ● Trip Level Scheduling ● Parity Equation Bias Removal

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Figure 6. Redundancy Management Error Compensation

The first error source is the moment arm effect caused by the separation of the accelerometers. The sensed acceleration at MRU A is corrected by a set of deterministic equations to the MRU B location so the outputs can be compared. The second error source, static misalignments, are installation errors that are normally corrected by boresighting. Because of the distance between the two MRU locations and the limited access, boresighting both boxes would not provide the required accuracy and is not feasible. Conventional boresighting is only accurate to ± 6 arcminutes while the MFCRS redundancy management system requires the relative orientation of the boxes to be known to ± 1.5 arcminutes. The MFCRS program uses a unique approach of boresighting just one box and then using the existing navigation alignment algorithms to calculate the exact installation orientation of each MRU. The results of the navigation alignments are compared and used to calculate the relative orientation of the MRU's to within the required ± 1.5 arcminutes. This method of making corrections for static misalignments is faster, cheaper, and more accurate than conventional boresighting and may have future application in determining the relative orientation of remote installation mounts for other systems. The third source of error is noise caused by the mechanical dithering of the RLG's to prevent "lock in" at low angular rates. This dither noise affects both the gyro output, by aliasing into the flight control frequencies, and the accelerometer outputs, by causing motion of the sensor block. Dither noise in the gyro channel also couples into the accelerometer channel through the angular rate and angular acceleration terms in the moment arm equations. The gyro outputs require extensive filtering because the differentiation process to get angular acceleration amplifies the dither noise and the long moment arms add further gain. Gain is also added by the coefficients required to transform the MRU B

outputs to the aircraft axes. A -60 db digital notch prefilter at the dither frequencies was placed in the gyro path and a third order analog lag prefilter was placed in the accelerometer path to attenuate the noise. The next error source is the misalignment between the two MRU's caused by aircraft bending during high g maneuvering. To account for this misalignment, the trip levels that the parity equation residuals are compared against are scheduled as functions of the sensed rates and accelerations. One major objective of the flight test is to validate the trip level equations and the aircraft bending models upon which they are based. Sensor biases are accounted for during turn on and warm-up by calculating the parity equation residuals under static conditions and then using these residuals as bias correction factors. With all of the errors compensated, any mismatches detected by the parity equations should be caused only by incorrect sensor outputs. As mentioned above, the status of the parity equations is used as a pointer in a set of look up tables to pick the best sensor triad. The look up table generation is done by an off line program that relates all possible parity equation states to the sensor that is most likely failed. The equations that relate sensor failures and parity equation states can be either interpreted geometrically, or derived rigorously using the Greatest Likelihood Ratio test. References 3 and 4 contain a detailed discussion of the redundancy management system and the generation of the stored tables.

2.2 FLIGHT CONTROL BASELINE DESIGN

The other key issue to be resolved by MFCRS, the flight control compensation, is shown in Figure 7 and consists of two sets of moment arm compensations and network compensations for the selected gyros and accelerometers. (4,5) The first set of moment arm compensations, as discussed above, corrects the sensed acceleration at MRU A to the MRU B location for redundancy management. The second set of equations corrects the selected accelerometer outputs from the MRU B location to the location of the production accelerometers. This second compensation allows switching between the production and MFCRS accelerometers, and avoids any changes in the F-15 flight control computers. The network compensation also avoids changes in the flight control computer by compensating for dynamic bending effects in the feedback loop of the flight control system. Network compensation is required since the accelerometers are not located at the primary aircraft bending nodes and the gyros are not at the antinodes. Instead, the accelerometers and the gyros are clustered so the system can navigate. In MFCRS this caused flight control system sensitivity to aircraft structural modes resulting in unacceptable stability and handling qualities. Using the baseline F-15 gain and phase margins as design goals, the open loop frequency response was used to determine the network filter requirements to achieve acceptable performance. A 15 db notch filter was required in the pitch rate and normal acceleration paths and a 30 db notch was required in the roll path. The yaw path did not require compensation based on the open loop frequency response. To offset the effect of the computational delay, a 3 db lag-lead network was placed in the pitch rate and yaw rate feedback paths. The 50 Hz computational cycle that was used by the navigation program appeared adequate for both the flight control compensation and the redundancy management.

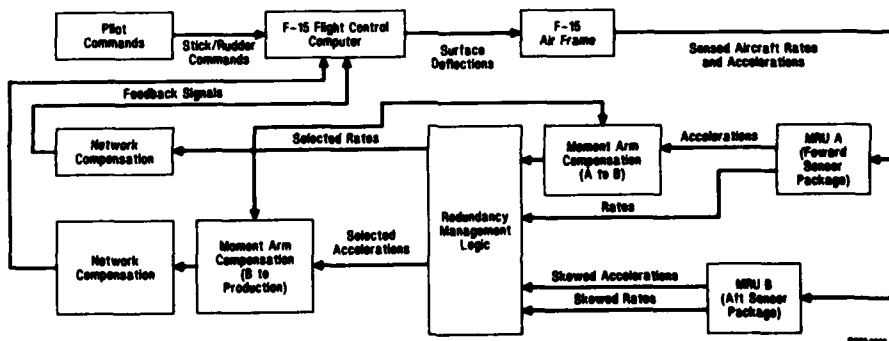


Figure 7. MFCRS Flight Control Compensation-Simplified Block Diagram

3. MODIFICATIONS IDENTIFIED BY TESTS/SIMULATIONS

The baseline system design described above has been subjected to a number of simulations and tests. Based on these results, modifications were required to both the redundancy management and flight control sections. Some of the results suggest modifications that are not within the scope of this program, but should be "lessons learned" for the production prototype design of a multifunction system. The redundancy management test results are discussed first, since some of the required modifications impacted the flight control design.

3.1 REDUNDANCY MANAGEMENT TEST RESULTS

The initial testing of the redundancy management went smoothly. To protect against coding errors, MCAIR developed a test case which injected simulated gyro and accelerometer inputs into a MCAIR computer simulation of the redundancy management system Honeywell had coded. Intermediate results were generated all the way to the output and the inputs and results were provided to Honeywell. Honeywell was then able to compare to the results generated by the flight code when the same simulated inputs were used. This method of testing was instrumental in assuring a very smooth software development. One of the keys to the success of this approach was the fact that the software for the MCAIR simulation and the Honeywell flight software were generated by different individuals. This check and balance situation insured design quality. The other key was the absolute requirement that all software changes must be incorporated in both sets of software, a new test case generated, and the Honeywell and MCAIR results compared to ensure proper coding.

The first indication of problems occurred in the software/hardware integration process. At this point test data indicated that the noise due to RLG dither was present on both the accelerometer and gyro outputs and was significantly larger than predicted analytically. This noise was large enough to cause false alarms in the redundancy management system. The trip levels were raised to the point where additional increases would allow sensor failures to cause aircraft transients before they were detected. After the trip levels were raised, the noise in the MFCRS was measured in an end to end test. Results showed that the noise was still unacceptable, and the sixth order filter was not providing the expected attenuation. Analysis showed that part of the problem was RLG quantization. The RLG measures angular rotation in discrete increments and accumulates these discrete rotations over a set time period to calculate rate. The increment to which the rotation is quantized creates quantization noise that raises the gyro path noise level to a -30 db value versus the -60 db design goal. The RLG resolution can be changed to reduce the quantization level, but the effort is beyond the scope of this program. Any future system must consider the flight control requirements in the design of the basic sensors. This problem does not affect navigation since the outputs are integrated over a long period of time and, unlike flight control, any quantization errors average out.

In addition to the gyro noise coupled into the accelerometer path by the moment arms, the accelerometer path also had more noise than expected with the third order lag prefilter. This noise appears to be related to the hardware implementation method rather than any fundamental phenomena. A careful redesign of the electronics with awareness of the noise sensitivity would probably eliminate the problem. The use of existing hardware and the limited scope of the program did not allow this comprehensive fix. Instead a first order .1 second lag filter was added to the accelerometer path, a 5 db lead lag filter that had been added for flight control was eliminated, and the parity equation moment arm compensation was changed to do the parity equation comparison at a central location. This last change had the effect of shortening the moment arm and reduced the noise in this path. Minor hardware changes are also being made to provide isolation in the accelerometer electronics. These filter changes required the modifications to the flight control system described below. In addition, these changes significantly increased the computational burden to near the processor limit. Any future design should attempt to minimize MRU separation in excess of that required survivability as well as allow sufficient processor sizing. Another solution that may be available to future systems is the selection of a redundancy management system that is less sensitive to noise. This is a difficult goal to achieve since the sensor reconfiguration must be done quickly enough to prevent transients in the flight control outputs. Some form of weighted averaging may allow a longer time period for decision making. These problems, solutions, and recommendations are summarized in Figure 8.

3.2 FLIGHT CONTROL TEST RESULTS

The first major flight control test was a man-in-the-loop simulation. The conclusion of this testing was that the system was stable and controllable for all maneuvers. However, during small amplitude stick raps and rudder kicks, the MFCRS system was slightly less damped than the basic F-15, requiring an extra half cycle of oscillation to damp. This decrease in damping was traced to the 3 db lag-lead network. To increase the damping and maintain desired stability, this filter was replaced with a 5 db, second order, lead-lag network. While this network was adequate to maintain stability and performance, analysis indicated that performance would be better if a computation rate between 80 and 100 hertz was used. This would allow a sharper roll off on the filters. The use of existing hardware with limited processing capability denied this option to the MFCRS program. However, performing flight

Problem	MFCRS Solution	Long Term Solution
Gyro Dither Noise	Raise Trip Levels Add - 60 dB Notch Filter	Reduce Gyro Quantization Level Measure Gyro Dither Stiffen Sensor Block Synchronize Dither and Outputs Design Future Gyros Considering Flight Control Requirements
Accelerometer Dither Noise	Raise Trip Levels Add Analog Prefilter Add 0.1 sec Lag Filter Remove Lead Lag Filter Change Moment Arm Compensation Hardware Changes to Increase Channel Isolation	Reduce Moment Arms Consider Alternate Redundancy Management Design Accelerometer Channel Considering Noise Sensitivity
Processor Computational Burden	Limit Scope of Software Changes	More Powerful Processor

Figure 8. Redundancy Management Problems/Solutions

control at 50 hertz with an existing processor will demonstrate this technology is feasible on current processors.

This modified implementation was given to Honeywell to code into the MRU processor. The same test case methodology described for the redundancy management was used in testing and verifying the flight code for the flight control modules. The development of the flight control software was complicated at this point by the filtering requirements discovered during the redundancy management testing discussed above. One of the results was the addition of a .1 second first order lag filter on the acceleration feedback paths. To maintain stability, the gain in the pitch and yaw control loop had to be scheduled based on dynamic pressure. Analysis also showed the 5 db lead-lag network provided some gain of the accelerometer noise. The use of gain scheduling allowed elimination of this network. To verify stability and performance after these changes, another man-in-the-loop simulation was conducted. The results indicated that this approach will provide acceptable performance and stability.

These changes illustrate the difficulty of integrating multifunction sensors with existing systems. Multifunction systems should not be inserted directly into current flight control systems simply as a sensor replacement. The multifunction system is part of the flight control loop and, to derive maximum benefit, the implications of the sensors should be considered in the complete flight control design. The MFCRS was constrained from modifying the flight control hardware so the flight control system could remain compatible with the existing flight sensors. This constraint should be removed in the design of a prototype system.

3.3 INTEGRATION VS INTERFACING

Taken together, the above flight control and redundancy management problems point to a fundamental problem in the design of integrated systems. The navigation units currently available for the MFCRS test demonstration were designed by navigation specialists who did not have any flight control requirements imposed upon them at that time. The flight control specialists who made the initial design modifications to the MRU's were not fully aware of some of the assumptions made in the basic navigation design. Currently, systems are designed for specific purposes and are made to interface with other systems. True integration, as shown by this program, is more difficult. As the integration of fire control, flight control, and propulsion control continues, similar problems are expected. In order to minimize integration problems and to take advantage of complementary capabilities, future programs need to be strongly influenced by a systems integration group. The objective of this group is to interact with engineers that are specialists in all of the systems being integrated in order to find and resolve conflicting requirements early in the design stage. To carry out this function, excellent communication must exist between the organizational elements responsible for flight control, fire control, and propulsion control, and the coordinated effort must be responsive to the influence exerted by the multifunction integration group. Project organization must facilitate these requirements for communication and coordination.

The foregoing problems, short-term solutions, and long-term recommendations are summarized in Figure 9.

Problem	MFCRS Solution	Long Term Solution
Filter Performance	Higher Order Filter	Increase Computational Rate
Redundancy Management Filter Changes	Flight Control Gain Scheduling	Design Flight Control System in Forward Loop
Lack of Flight Control Consideration in Design of Baseline Navigation Hardware	Add on Changes	Integrate Instead of Interface Design for Multifunction Requirements Use System Integration Group to Interact Strongly in the Design

Figure 9. Flight Control Compensation Problems/Solutions

4. FUTURE PLANS

The testing in the MFCRS program has been instrumental in identifying problem areas and design considerations for future systems. However, there are still questions that cannot be answered by laboratory testing. The MFCRS system will be flown in an F-15 testbed in late 1983 to evaluate aircraft stability and handling qualities, fault coverage, reconfiguration transients, navigation performance, and reliability. This flight test is not meant to be an exhaustive evaluation of a new system but will focus on two key areas: flight control and redundancy management performance. Stability and handling qualities evaluations will answer the question of how well separated sensors at non-optimal locations can be used for fighter flight control. Fault coverage and reconfiguration transient evaluations will hinge on how well the effects of structural bending and differential vibration have been modeled and compensated for in the redundancy management algorithm. The program schedule in Figure 10 shows the tasks that have been accomplished to this point and those remaining, including aircraft integration and flight test.

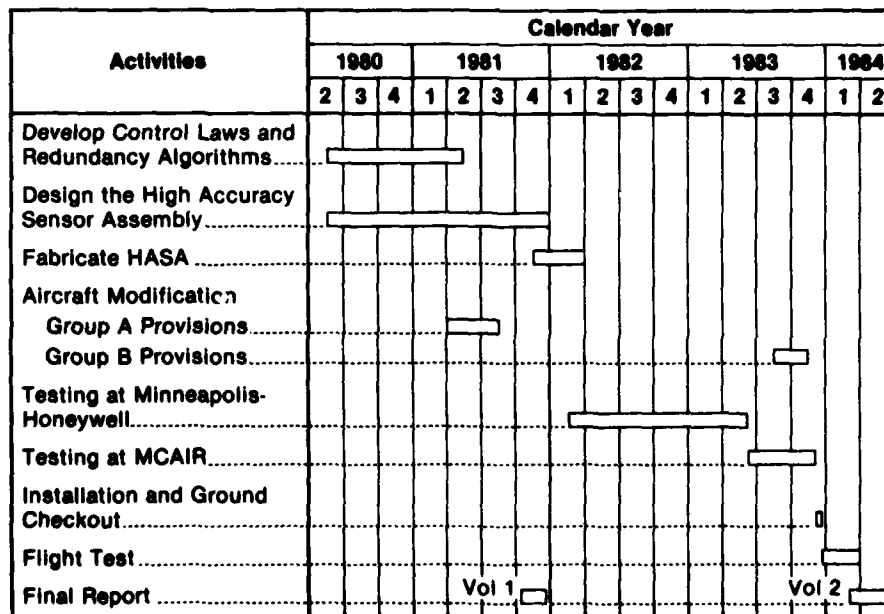


Figure 10. Multifunction Flight Control Reference System Overview Schedule

5. CONCLUSIONS

The Multifunction Flight Control Reference System concept addresses the problems of current systems that require inertial data and provides for the needs of future systems. The MFCRS program uses existing hardware to demonstrate the concept in a cost effective manner.

A redundancy management system using parity equations has been developed for the separated sensors. One key aspect of the redundancy management is the use of an off-line program to relate parity equation status to the most likely failed sensors. This off-line program reduces the real-time processing requirements to allow use of an existing processor. To support the redundancy management system, a one time electronic alignment procedure has been developed and implemented to determine the relative orientation of the two sensor clusters to within ± 1.5 arcminutes. This electronic alignment procedure is faster, cheaper, and more accurate than current optical/mechanical boresighting procedures and has potential for application in other programs.

Flight control and redundancy management compensation has been developed, implemented and simulated for a worst case sensor separation. The software for the redundancy management and flight control was successfully developed and coded. A key to the successful software development was the use of a test case to validate the original code and all subsequent changes.

The results of testing have shown that noise caused by RLG dither is a severe problem for flight control redundancy management. Noise attenuation should be addressed in the basic system design. Testing has also shown that a multifunction system must be considered as part of the flight control loop and the overall flight control system design must consider issues peculiar to the multifunction system.

The suggested management technique to design future integrated systems will be to use a strong systems integration group to interact with and be part of the design team.

Future testing on an F-15 aircraft will provide data so the key areas of flight control and redundancy management performance can be evaluated.

The design, development, and testing of the MFCRS to date has not shown any basic flaws with the multifunction concept, but has provided valuable insights for future designs of integrated systems.

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SOME ASPECTS OF FLIGHT TRAJECTORY CONTROL IN FUTURE AVIONIC SYSTEMS
FOR COMBAT AIRCRAFT

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SUMMARY

This paper considers some of the reasons for increased integration with the emphasis on Flight Profile Control in combat aircraft, largely in the ground attack role. Having examined some of the reasons for further integration involving flight control, it looks at the various phases of flight and highlights particular problems to be considered. Having suggested trends or solutions in the more important specific areas, the paper suggests that future work will consist of developing particular capabilities, concentrating on the relationship between pilot and total system, learning how to control the design of such a closely coupled system and handling the important problems of testing, reliability, maintainability and the attainment of minimum economic configurations which will satisfy all these goals. The need for flexibility in any system which combines automation with a high degree of pilot participation is stressed.

1. INTRODUCTION

For some years the design of avionic systems for combat aircraft has been subject to a number of pressures which have not always been compatible. The need for survivability in an increasingly hostile environment drives such operations towards flight at very low altitudes which in turn tends to raise the pilot's workload and minimise the extent to which he can operate head-down. At the same time a drive to operate at night and in low visibilities involves the carriage of more sophisticated forward looking sensors such as enhanced radars and electro-optical systems which, in their initial form, require considerable pilot interpretation. Future dispenser or guided weapons for use against vehicles or tanks should be highly effective if launched appropriately in conjunction with sensors, but this could complicate the pilot's task in weapon delivery.

All these trends indicate the need to reduce pilot workload by using automation while at the same time improving the accuracy with which the desired flight profile is flown. While past standards of navigation have been adequate as regards position finding there is a need for improved path keeping and conceivably for an improvement in basic accuracy, both of which increase the interest in flight automation.

There is also considerable scope for the application of automation to the aircraft itself for ACT to provide higher efficiency and better handling, and to the handling and monitoring of the basic aircraft systems. With so many possibilities it is important to achieve a picture of the balance which is being sought and also to use integration as a tool to reduce overall complexity and increase testability while reducing cost of ownership.

2. REASONS FOR CLOSER SYSTEM INTEGRATION

The development of computers and better means of data transmission including the data bus have opened the way to further integration provided that the basic disciplines of system design are observed. It is however important not to pursue integration for its own sake but to ensure that any integrated system is designed around a coherent philosophy.

Generally integration can be pursued for reasons connected with system design and management or operational reasons.

The design reasons are largely concerned with economy: there is an optimum way of combining sensors, displays, automatic controls and computing facilities to achieve a given operational capability with a given degree of redundancy which must be justified. It is possible to trade off availability against the rate at which faults will arise and have to be corrected. While the massive amount of data which can be interchanged in modern bus systems has increased the number of solutions available for testing and fault diagnosis, some configurations will still be more maintainable than others. However, in high performance military combat aircraft it is much more likely that the first driver towards integration will be operational.

The areas in which operational integration can be used to solve problems can be distinguished as below. In each case the design objectives will be slightly different, reflecting a different emphasis on such aspects as performance, safety, pilot acceptability, etc.

Automation may be used to off-load the pilot by simply taking over tasks which are better performed thus without being changed in nature. Simple auto pilot facilities

such as height holding or following radio aids are in this category but there may also be highly sophisticated tasks such as automating target recognition through E-O sensors. One advantage of this approach is that relatively piece-meal automation can be carried out without necessarily changing the pilot's task in its essentials. Similarly the safety philosophy need not change.

In some cases automation may actually do a better job than the pilot in terms of performance; e.g. flying a complex flight trajectory which might not easily be flown manually with sufficient accuracy.

There is a sound operational case for any application of automation which can increase survivability, such as any system of automatic terrain following or avoidance which permits very low flight in all weathers.

Once the concept of increased automation of flight path control is accepted it becomes possible to consider its exploitation in changing the shape of a mission or even by developing new kinds of weapon or tactics which could not be used otherwise.

At one time there was an explosive growth in the application of automatic flight control to civil transport aircraft, culminating in the adoption of automatic landing down to Cat. 3 conditions in which, at the time, it was considered human pilots would never be able to operate safely. Such systems, designed on the basis that an automatic solution could be made to work, were sometimes called "ultra-human systems". Subsequently it was found that an appropriate mix of human pilot and automatic or instrumental assistance could do the job more economically, but the idea that there are valid system concepts depending entirely on the use of automatics seems to have been accepted together with the resulting complexity, e.g. active control. A solution which hinges success on automatics and their relationship with the pilot implies a complete commitment and therefore formidable design problems, including those concerned with safety.

3. INTEGRATION IN SINGLE SEAT COMBAT AIRCRAFT

3.1 General

While the integration of fire control/weapon delivery functions and flight control is the primary topic of this Symposium it is necessary to see the operation of a combat aircraft in the context of all the relevant phases of flight. It will be necessary to make transitions between phases both smooth and logical and an integrated approach also demands that wherever such facilities are included for a particular purpose they should also be used to optimise the operation wherever possible.

3.2 Transit to Target

Here the flight trajectory is essentially concerned with navigation, normally in two dimensions but in three if it is expanded to include selecting a vertical profile which is the best compromise between achieving masking from defences and avoiding the ground. The general pressure is to fly at increasingly low altitudes.

There are of course extremely sophisticated single or two seat aircraft which contain a complex equipment fit aimed at both these objectives and at day/night operations. But to consider the system integration problem it is better to start with a relatively simple single seat aircraft in which the exercise is a combination of computers, displays and pilot's eye. An inertial navigator with a corresponding guidance facility can be programmed to steer a horizontal track through a series of pre-planned waypoints. The control loop is completed through the Head Up Display (HUD). Pilots are skilled at both adjusting their altitude by looking ahead by day and making small horizontal adjustments to take advantage of the terrain. A primary objective is survivability and at present this can be increased in several ways apart from flying lower. It is a valid function of automation to reduce workload by removing a task in time. Modern briefing systems (Fig. 1.) can store large amounts of data about optimum flight paths and enemy defences so that the planned trajectory can be more sophisticated, reflecting the best available knowledge, and being loaded into the aircraft's computer before take-off.

In handling the horizontal profile there is a choice between reinforcing the pilot's eye, adding an extremely sophisticated sensor or moving towards some sort of planned horizontal navigation. Various night vision devices or FLIR, perhaps displayed on the HUD, all tend to reinforce the pilot's eye solution but whether they are adequate seems to depend on whether flight at much lower altitudes is contemplated. The freedom to manoeuvre which one would expect most pilots to like implies that blind flying through a sensor must include the capability to turn on an instinct to do so, in which case the sensor must acquire information as to what is "round the corner" instantly and determine the best profile. Such a radar capability was demonstrated by Ferranti some twenty years ago and highly effective systems have been produced since, but they are expensive. Radar and Electro-Optical systems compete for the same real estate around the nose of the aircraft and the system integration approach implies attention to other possibilities.

A theoretical possibility is that an ideal navigation solution can be postulated in which there is a digital database representing the terrain which, coupled with very accurate navigation, enables the generation of control inputs for auto-

matic or manual flight at very low altitudes. Other possibilities include a more realistic reinforcing of such a system with forward sensors which could particularly handle the "round the corner" problem.

Whatever solution is adopted it is most attractive to try to off-load workload through automatics in transit so as to enable the pilot to pay more attention to the tactical situation, enemy defences and the operation of his own counter measures. Clearly any system involving full automatic control at very low altitudes could tend to be "ultra-human" which would mean some degree of dual or triple redundancy, which might be analytical in terms of sensors but could not be in terms of control. It is attractive to consider a mixed manual/automatic solution of which there are obviously many possibilities including:

Fully automatic control of ground clearance at some altitude safe for the system with manual control of heading changes through bank to vary the horizontal profile.

Something similar but "transparent" to enable the pilot to overpower the system without disengaging it so as to go to a lower altitude or turn off.

Some form of blending, more like an auto-stabiliser, with pilot and automatic inputs in parallel, the latter overpowering the former in the event of a hazard, with some suitable warning as to why.

This implies a system, rather like an ACT system with limitation within a given flight envelope, the envelope in this case including the ground. It does seem that a fully automatic system would be unacceptable, apart from its complexity, because of the discontinuity when the pilot had to reject it in order to carry out a perfectly normal manoeuvre. The system could never be programmed for all "normal" manoeuvres. It also seems that the final solution will have to behave "naturally" as seen by pilots rather than as seen by designers. But certainly the sensors (inertial, radio altimetry and possibly radar or laser) and the database or mixing techniques required to implement the variety of solutions are available.

It is also possible that future systems will integrate rather more data about the position of defences, threats, etc. and if this data could be changed in the air from intelligence which was not stale safe changes in trajectory could be generated for presentation to the pilot. They could be fed into an "advisory" type of control system as described above. But in this case, expecting a "natural" response, the pilot would also have to have a suitable display saying why the system wanted to adjust the profile. Present electronic display technologies (Fig. 2.) provide the means to do this but the available content may have to be limited.

3.3 Ground Attack

Papers on the IFFC system (ref. 1.) make some good points about the advantages of a partially automatic trajectory solution in ground attack. The most notable is that weapon release in a curved trajectory makes life more difficult for the defences, and this advantage can be considered in the context of a typical attack.

The most difficult form of attack is likely to be against armoured formations of which the precise location and distribution is a little uncertain. Weapons tailored specifically for this task will have to be released in such a way as to match the release point, the weapon characteristics and the array of vehicles in as optimum a manner as is possible in a very short time. In some cases a pop-up manoeuvre will be employed. In all cases the pilot workload will be high, particularly if he has to interact with his E-O sensors before and after target acquisition. One would expect it to be possible to relieve this load partially by a degree of automatic flight trajectory control, say by flying a planned curved trajectory which also increases survivability. However there could be problems in this area. For example if the location of the target is not known with sufficient precision the pilot will find himself making final corrections in a curve rather than in straight flight, with a consequent effect on the utility of the HUD.

It does seem that in weapon delivery the greatest advantage of a degree of automatic control is in terms of accuracy and rapid settling of aiming provided that the necessary aiming information is available. One would expect the technique to function better in air-to-air situations than in badly structured ground attacks requiring a considerable amount of last minute decision making.

3.4 Air-to-Air Combat

It is instructive to examine the possibility of improving air combat effectiveness if flight and fire control are integrated. In particular the head-on approach is of interest in an air superiority aircraft as is the role of this technology in front hemisphere attacks. Here the problems of weapon aiming are particularly exacting, being accentuated by high range rates, line of sight rates and line of sight accelerations.

In the past the preferred interception weapon has been the guided missile, when optimized for this type of attack. However air-to-air guns can show distinct advantages in terms of reduced weight, reduced profile drag, expense and the ability to carry a large number of projectiles. Weight and drag shorten mission times through an attendant fuel penalty.

For guns to be effective in this mode of attack it is necessary to understand the effects of target acceleration and its measurement and the likely order of magnitude of the variables involved before the necessary improvements in weapon aiming computation can be made. The pilot's task must then be considered, particularly in terms of target acquisition, tracking and ability to handle the system within whatever firing opportunity emerges. It should be possible to enhance tracking by an integration of fire and flight control including some suitable assignment of tasks to pilot and automatics.

The simulation upon which this part of the paper is based was originally undertaken to examine the dynamics of such attacks as part of more comprehensive studies aimed at the definition of future airborne radars. The first part, which is reported here, dealt with the simulation and investigation of typical approaches.

The simulation results shown in Figs. 3 - 7 show the paths followed by the attacking aircraft and its quarry and use the following assumptions:

At time zero the fighter will have position co-ordinates (0, 0) and velocity of 500 ft/second in unspecified direction.

The target's flight is parallel to the Y-axis prior to acceleration, target acceleration being applied smoothly over a one second interval.

Coriolis and gravitational forces are neglected as being trivial or not significant in the present simulation.

Bullet velocity is approximated with a single value.

The target position, velocity and acceleration are known exactly at any given instant.

The cases considered include the following:

Fig. 3: Target flies straight with an initial offset of 500 ft.

Fig. 4: The target applies an acceleration of 8-G after flying straight for about 2.3 seconds or, in Fig. 5, an acceleration in the opposite direction.

In Fig. 6 the target applies acceleration about 1.3 seconds earlier while in Fig. 7 it has a large offset (2,900 feet) and flies straight at all times.

It will be seen that firing opportunities are normally around 4 seconds although in Fig. 6 the opportunity is increased to some undetermined large value; neglecting target acceleration and assuming straight flight the rate of increase of fighter acceleration to maintain lead angle is a function of offset. Acceleration either towards or away from the fighter flight path has little effect in direction but acceleration towards the fighter causes a marginal reduction in firing opportunity.

A detailed scrutiny of the results shows some general characteristics of such an attack. A firing opportunity is taken to involve a time of flight of less than 1.5 seconds and a fighter turning acceleration of less than 8-G. In most cases firing opportunities last for around 4 seconds. Over the first 2 seconds modest turning accelerations of less than 1-G appear but in the latter part of a firing opportunity, at shorter ranges, a very fast wind-up into the turn, say 2.5-G/second, can be encountered.

Considering the pilot's task as a whole when encountering a threat justifying a front sector attack:

The threat will be detected on radar say 30 seconds before the firing opportunity appears and this is the time available to prepare a response.

From 5,000 feet in everything will be over in under 5 seconds and although the demanded turn rates will be modest for the first 2 seconds of the firing opportunity they will wind up rapidly in the last 2 seconds.

In this rapid sequence of events the aircraft will have to be flown extremely accurately in response to weapon delivery information. The problem will be exacerbated in turbulence and the pilot may be interrupted by other factors such as EW or communications.

It follows that integration with flight control should be used to reduce the pilot load as much as possible. Where flight control commands are blended, presumably pilot tracking of the required profile, a major load even without striving for accuracy, can be substantiated by flying automatic tracking with which automatic firing armed by the pilot could be used.

In this case it does appear that as much flight control authority as is possible could be used with advantage provided the pilot were free to over-ride it instantaneously, for example, to break off the attack for some unforeseen reason e.g. the target no longer exists. In any event an immediate reversion to full manual control will be necessary to enter a break-away manoeuvre which is entirely impossible to predict in terms of the demands which would be required of the automatics.

The most important impression gained from this example is that a most useful manoeuvre, the front-hemisphere attack, is extremely demanding in terms of time apart from accuracy and the ability to make such attacks possible will be a major attraction in considering integrated fire-flight control.

4. FUTURE PROBLEMS

There are strong intuitive reasons to feel that pilots of single seat aircraft should have the routine task of flying the aircraft off-loaded as much as possible just as their handling of other aircraft systems and the interpretation of complex data from EW or communication sources should be automated before display. But it also seems clear that a major problem is to decide what is worth doing, or perhaps what not to do, as the technology is so powerful. Clearly future work should concentrate on the pilot relations and on the best way of developing total system design concepts which can be validated in terms of human factors as well as technically.

At the same time, with the advent of more complex avionics and weapons, the designers aim should be to over-kill the workload problem as otherwise pilots will find it difficult to handle the resulting complex systems, particularly under stress.

It is an interesting coincidence that the relationship between pilot and system and between designers and design aids follows the same pattern. At the highest level of abstraction the pilot is concerned with tactics and flexibility and at the lowest with handling equipments and carrying out optimal manoeuvres. A corresponding split of responsibility between pilot and avionics is dictated. In design, at the conceptual level, the problems are multi-dimensional and require an interaction between designers and operators.

The act of automation is essentially to displace a problem in time, in space or both. A highly automated approach to the problems of combat aircraft makes the assumption that, many years ahead, pilots and system designers sitting in conference rooms or laboratories can actually solve problems which will occur years later in the heat of a battlefield. It seems best to assume that as this cannot succeed completely the quest is one for flexibility.

5. CONCLUSION

The paper has reviewed some aspects of automation and therefore the integration of flight control with other functions in combat aircraft.

It has been pointed out that the critical area is between pilot and aircraft system and that even the most successful interaction between operators and designers would probably produce a solution which was not entirely valid in a battlefield. At the same time it is vital to reduce pilot workload.

In detail it has been suggested that some form of advisory automatic control coupled with displays should be used to vary horizontal navigation for survivability and that in the vertical the approach is nearer to a form of flight envelope limitation which includes the ground.

In terms of weapon delivery it seems that the introduction of a degree of automatic flight path control will pay off most in cases in which an improvement in aiming would be beneficial coupled with a more rapid settling to a stable solution. In ground attack where target positions and distributions are uncertain the problem is more difficult and requires further investigation which will have to involve topics concerned with targets and weapon design which are outside the scope of this paper.

In air-to-air combat improved sensors and fire control computers have the potential to make weapons more effective provided that the necessary aiming accuracy can be achieved, possibly in turning flight and where events occur very rapidly. This indicates a favourable direction in which to apply the integration of flight and fire control and the paper has shown, that for example, front hemisphere attacks using guns would certainly fall within this category. At the same time just as automatic and manual control may be blended to carry out difficult manoeuvres, the different phases of an operation will have to blend smoothly into one another. In air-to-air attacks it is important to get the approach right while at the same time the system must cater for quite abrupt discontinuities when the pilot has to abandon a programme manoeuvre and take improvised action drawing on all his tactical and flying skills. On the whole the past history of automatic flight control

has been more successful in terms of flight profile control itself than in handling pilot interactions and changes of mode or sector, particularly when the operation has to go off-programme.

It therefore seems that there should be several broad thrusts in the development of integrated fire-flight control.

Future operations and tactics should be studied carefully to determine precisely how such improvements could be beneficial; e.g. in attacking ground targets will their position be known and if not what problem will confront the pilot?

Practical system development such as that already underway is required to determine optimum flight profiles and control techniques.

It will be necessary to establish whether these new possibilities can be realised with present sensors or whether the sensors themselves require modification or upgrading in performance.

The resulting combinations of sensors, automatics and systems should be exploited as much as possible in all phases of flight and a major effort will be required in the context of a particular system design to determine precisely how it will handle all operational eventualities, particularly those which cannot be foreseen easily but may result from unexpected circumstances.

Finally, having determined a new relationship between pilot and automatics, it will be necessary to ask whether this has particular effects on display and control requirements.

The author is grateful to the Management of Ferranti plc for permission to publish this paper and to his colleagues within Ferranti and elsewhere in the UK for stimulating the ideas contained in it.

Reference 1: Integrated Flight and Fire Control Development and Flight Test on an F.15B Aircraft: Sims, Hillman, Kocher and Green NAECON May 1982 and other related papers.



FIGURE 1. A Computer-based Briefing System with a Data Base and Flight Planning Facilities.

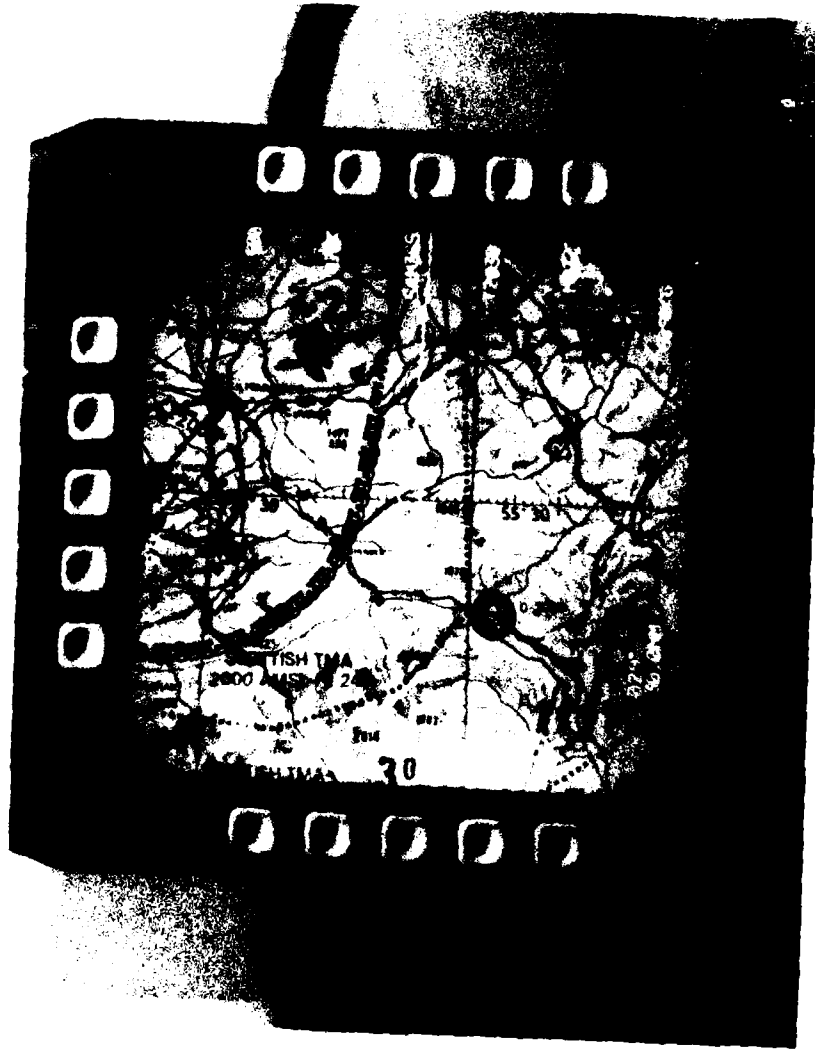


FIGURE 2: Full Colour Topographic Display Capable of Having Electronic Display Formats Super-imposed on it

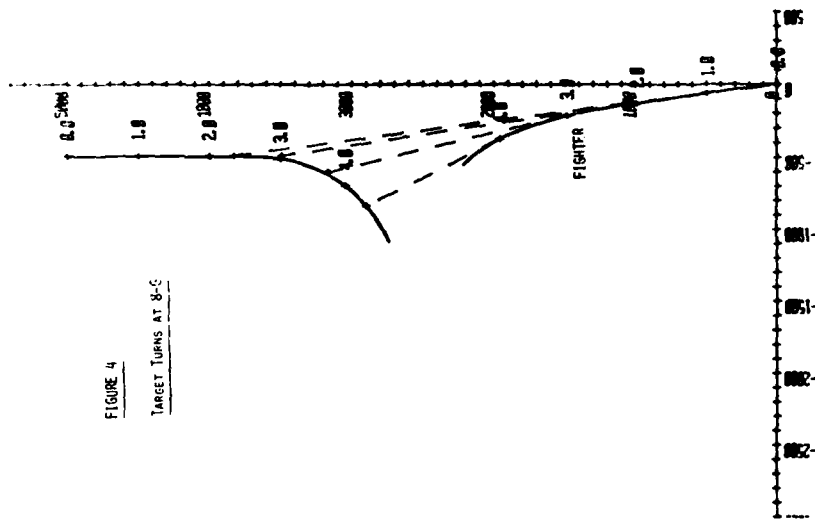


FIGURE 4
TARGET TURNS AT 8-C

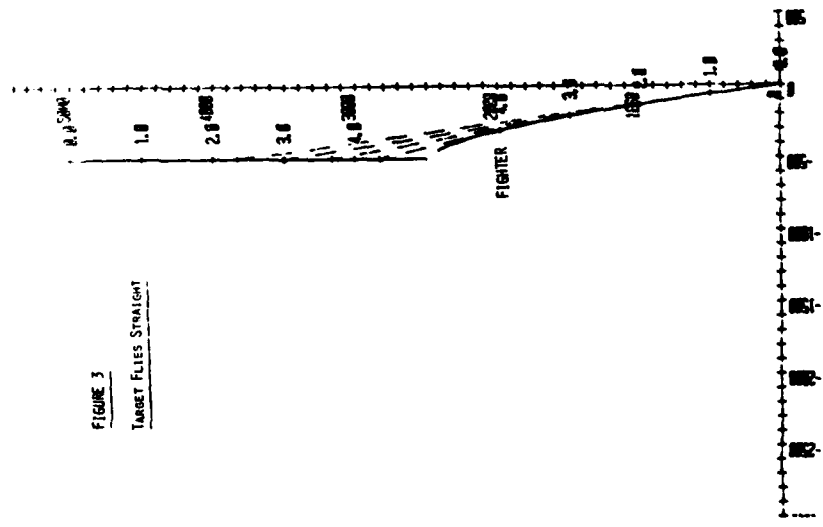


FIGURE 3
TARGET FLIES STRAIGHT

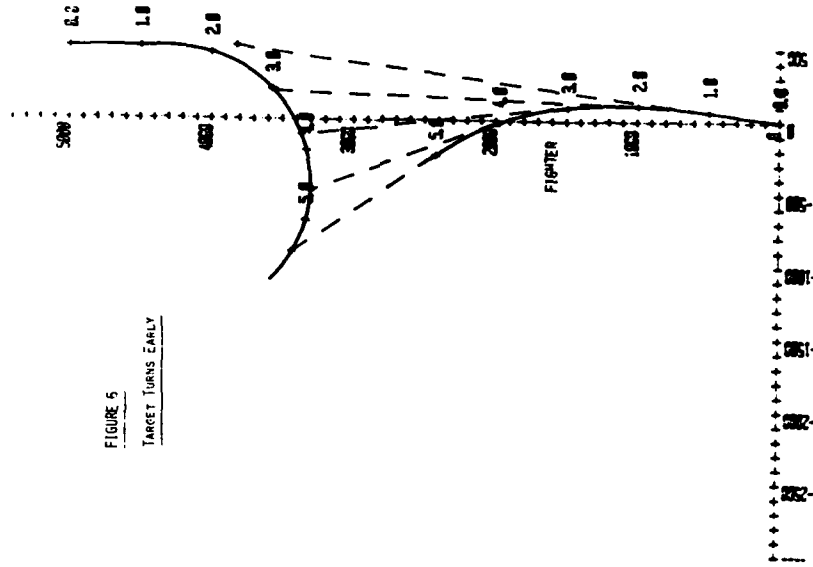


FIGURE 5
TARGET TURNS EARLY

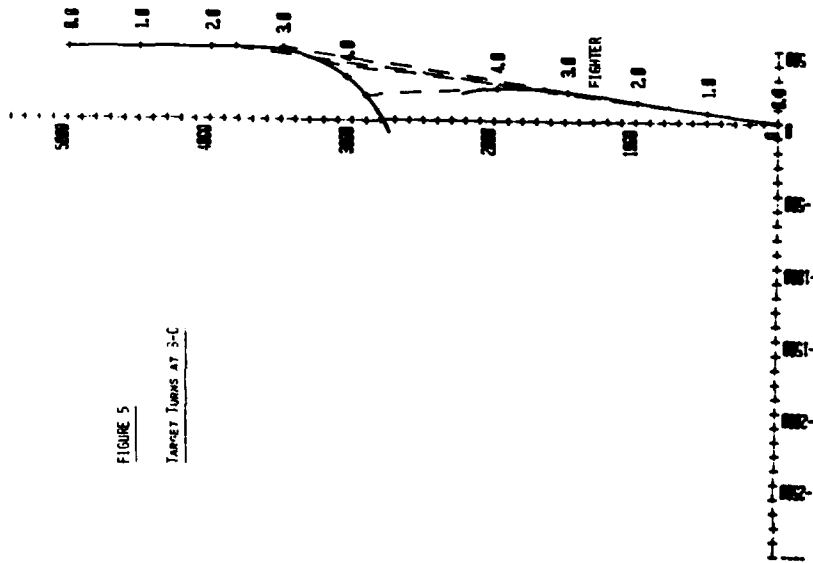
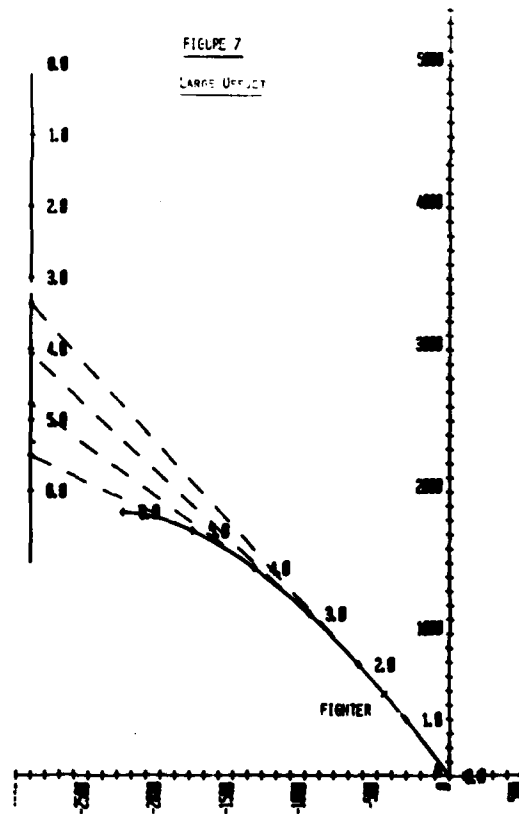


FIGURE 5
TARGET TURNS AT 3-C



A DIAGNOSIS SCHEME FOR SENSORS OF A FLIGHT CONTROL SYSTEM
USING ANALYTIC REDUNDANCY

by

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SUMMARY

A diagnosis scheme for sensors of a flight control system is presented. Based on analytic redundancy, a duplex sensor configuration provides the fail-operational capability of a conventional triplex sensor system. This is achieved by using deterministic observers. The feasibility of the presented concept is demonstrated by flight test results.

1. INTRODUCTION

Reliability plays a vital role in flight control systems of today and in those of the future. Particularly the most attractive control concepts such as artificial stability for the enhancement of maneuverability and flight economy require control systems of extremely high reliability. Traditionally this requirement is fulfilled by using multiple devices in the vital parts of the control systems. For example, it is necessary to triplicate the hardware and to add a majority voting mechanism in order to achieve a fail-operational capability.

However, it is obvious that the conventional hardware redundancy has many disadvantages due to costs, weight, volume, energy consumption, failure rates and maintenance costs. Therefore it is reasonable to look for alternative methods which reduce the necessary efforts in hardware without loss of reliability.

For the sensor part of a flight control system a starting point for the reduction of hardware is the fact that the signals which have to be measured are output signals of one single plant. The plant itself is given by the aircraft motion described by the flight mechanics equations. Hence the plant outputs are not independent from each other but they are internally coupled. These relations given analytically can be used for reliability purposes. Thus the hardware redundancy can be replaced by the so-called analytic redundancy.

The basic tools for utilizing the analytic redundancy are filters and observers. Their algorithms have to be implemented into the signal processing part of the control system. Thus sensor hardware is replaced by computer software.

In recent years several methods have been developed following the described common idea on different ways [1], [2], [3], [4], [5], [6]. In this paper a concept [6] is proposed omitting the third sensor of a triplex sensor system. Nevertheless the fail-operational capability shall be maintained. This is achieved by analytic redundancy performed in deterministic observers. The concept is applied to a complete flight control system similar to those used in the presently forthcoming transport aircrafts. Flight test results will be shown in order to demonstrate how the concept operates in a realistic environment.

2. SYSTEM OVERVIEW

2.1. The closed loop control system

Fig. 1 shows the general structure of the complete system partitioned into the closed loop control system on the one hand and into the failure detection system on the other hand.

The closed loop control system consists of the plant, a duplex sensor configuration, a sensor switching device and the controller. The plant is given as the flight mechanical motion of the aircraft, forced by the control signals and the disturbances such as air turbulence. The output of the plant is defined such that the resulting vector y contains as much information about the plant state as it is needed for the control problem. The output signals are measured by sensors which are put into a duplex sensor configuration in such a way that two identical sensor packages (SP) arise. Hence either SP includes one sensor of every sensor type (ST). As far as no sensor failure (SF) occurs the resulting two measurement vectors z_1 and z_2 are identical.

These two sets of measurement signals are fed to a controller via a sensor switching device. In this device the output signals of a failed sensor is cut off. Output signals of good sensors only can pass. This is shown in fig. 1 for a sensor type i according to the following scheme:

No sensor failure: The sum of both sensor outputs z_{1i} and z_{2i} multiplied with the factor 0.5 is fed to the controller.

Sensor i in SP1 failing: Only the signal z_{2i} of the corresponding good sensor of SP2 is fed to the controller.

Sensor i in SP2 Only the signal z_{i1} of the corresponding good sensor of SP1 is fed to failing: the controller.

This scheme applies to each of the existing sensor types (in fig. 1 shown for ST 1 only). The switch command signal is generated within the failure detection logic as part of the failure detection system.

The controller is supposed to be designed such that the closed loop control system meets the usual requirements, i.e. good response to command inputs and an effective suppression of disturbances over a certain area of the flight envelope.

2.2. The failure detection system

The failure detection part of the complete system shown in fig. 1 consists of two identical deterministic observers and a failure detection logic. The observers operate in the usual way. A mathematical model of the plant is driven by the control signals already mentioned as input signals to the real plant. The output \hat{y} of the model is an estimate of the real plant output y . The difference vector between the measured output z_1 (or z_2 resp.) and the estimated output \hat{y}_1 (or \hat{y}_2 resp.) in this paper called the observer difference is fed back to the plant model via the observer gain matrix. This feedback arrangement offers the opportunity to achieve an optimal estimation.

In common applications observers are used to provide additional information about the plant state. Thus the possibility is given to improve the control performance in a certain optimal way. Differing from those systems in which the observers have to be implemented into the closed loop control system the present concept uses the observers in an open loop manner. As fig. 1 shows the observers are not used for control purposes but they are restricted to the task of failure detection.

This task is accomplished with the aid of the observers differences n_1 and n_2 based on the following facts. As it can be shown easily by the state equations of the observers, the observer differences are forced by

- disturbances
- plant parameter variations
- sensor failures.

As far as the relations between sensor failures and observer differences are concerned it can be seen from state equations and also from the block diagram in fig. 1 that the observer difference vector n_1 of observer 1 is forced by failures in the sensor package 1 only. Equivalently the observer differences n_2 correspond to sensor failures in SP2.

The effect of the disturbances and the plant parameter variations on the observer differences are identical in both observers. Furthermore, they can be assumed to be limited. Hence, the maximum response of the observer differences due to disturbances and parameter variations can be used as thresholds for the sensor diagnosis in the following manner.

Given are the thresholds n_{1T} of each element n_{11} and n_{12} of the vector n_1 and n_2 . When one or more elements n_{11} of observer 1 increase beyond their respective thresholds n_{1T} , it is certain that a sensor failure has occurred in sensor package 1. Equivalently an occurrence of a sensor failure in SP2 can be seen from the response of the observer difference vector n_2 .

The final localization of a failed sensor within the SP1 (or SP2) is accomplished by the simple comparison between the output signals of the corresponding sensors of the same type. This part of the failure detection is identical to that used in the conventional hardware redundancy concepts.

Based on these relations the failure detection logic is defined by the following statements:

- $|z_{11} - z_{12}| \neq 0$: sensor failure in ST 1
- $(|n_{11}| > n_{1T}) \vee (|n_{21}| > n_{2T}) \vee \dots$: sensor failure in SP1
- $(|n_{12}| > n_{2T}) \vee (|n_{22}| > n_{2T}) \vee \dots$: sensor failure in SP2

Using these statements a failed sensor is clearly localized. Then, switch-off actions are taken as described in section 2.1.

3. APPLICATION TO A COMPLETE FLIGHT CONTROL SYSTEM FOR A TRANSPORT AIRCRAFT

3.1. The flight control system

The failure detection concept is applied to a given flight control system for a transport aircraft. A simplified block diagram in fig. 2 shows the structure of the closed loop system. The objectives of the control system are on the one hand a good response to the command signals defined as the command vector

$$y_C = \begin{bmatrix} \text{altitude command} & H_C \\ \text{speed command} & V_C \\ \text{bank angle command} & \phi_C \end{bmatrix}$$

and on the other hand a sufficient suppression of the disturbances w .

A good command response is achieved essentially by a careful design of the feedforward signal path, given a well damped eigen behavior of the closed loop system by choosing reasonable feedback gains. Then, this latter property provides also for a sufficiently low disturbance response.

In order to achieve these design objectives it is necessary, first, to control the plant via the control vector u defined as

$$u = \begin{bmatrix} \text{elevator deflection} & \eta \\ \text{thrust} & F \\ \text{aileron deflection} & \zeta \\ \text{rudder deflection} & \zeta \end{bmatrix}$$

and, secondly, to get the information about the plant state via the measurement vector z , defined as

$$z = \begin{bmatrix} \text{yaw rate} & r \\ \text{roll rate} & p \\ \text{pitch rate} & q \\ \text{bank attitude} & \phi \\ \text{pitch attitude} & \theta \\ \text{vertical speed} & \dot{H} \\ \text{altitude} & H \\ \text{indicated air speed} & V_{IAS} \\ \text{lateral acceleration} & a_y \end{bmatrix}$$

This flight control system is successfully flight tested with the DFVLR experimental aircraft HFB 320. A detailed description is given in [7].

The feedforward control loops are independent of the measurement. Hence, they do not interfere with the sensor failure problem. On the other hand the signal path from the measurements via the feedback law to the control vector u is of particular interest for the sensor failure problem and its solution, since via this loop sensor failures have an effect on the plant.

3.2. The test arrangement for the generation, the detection and the isolation of sensor failures

The sensor diagnosis scheme needs a duplex configuration as represented in fig. 1. But in the experimental flight control system only a simplex sensor system was available. Therefore a slight modification of the original concept became necessary for test purposes. This is shown in fig. 3.

Given is the single sensor package as it was used for the flight control test program. All measurement signals combined into the measurement vector z are splitted up into two separate signal paths. Into either path additive signals can be fed represented by the vectors v_1 and v_2 . Thus failures in sensors of the original two sensor packages are simulated by software. The remainder of the original scheme stays unchanged.

The intrinsic purpose of the experiment can still be achieved, i.e. the indication in which of the two sensor packages a failed sensor is located. Only the indication of the type of the failed sensor operates in an unrealistic condition. But as already mentioned this indication implies no innovative aspects since it arises in the conventional hardware redundancy concepts, too.

Fig. 4 shows in detail the arrangement of the sensors and the internal structure of the observers as it was used in the flight tests. For clarification purposes observer 1 and the signal path of the simulated sensor package 1 is represented only. The representation of observer 2 and the SP2 is identical to that shown in fig. 4.

The aircraft motion model used in the observers is nonlinear. This is done in order to keep the estimated outputs as close as possible to the outputs of the real plant. The nonlinearities consist of

- the coupling between the longitudinal and the lateral motion
- some quadratic terms depending on the dynamic pressure
- the thrust depending on Mach number and static pressure.

The differences between the measurement signals and the estimated outputs are fed back to the plant via the observer gain matrix. Going beyond this concept shown in fig. 1 and described in the original observer/filter literature additional integral terms are fed back, too. This is done in order to cancel possibly stationary deviations in the respective observer differences completely. The implementation of these integral elements into the observer structure does not raise any problems either theoretically or practically. The observer gain matrix itself is chosen as similar to the gains used in the control system. Though this approach is not stringently optimal from the failure detection point of view it offers some practical advantages. Detailed reasons are given in [6].

In fig. 4 additional low pass filters are attached to the observer differences n_x and n_θ . These filters are not part of the closed loop observers because the corresponding filter outputs are free. But they are used for the sensor diagnosis in the same way as the observer differences themselves: When a filter output increases beyond a previously defined threshold the corresponding sensor package is indicated as that one which is including a failed sensor. The purpose of these filters is to imply further indication signals which can be made sensitive to particular failure cases by individually matching the filter characteristics. The effect will be discussed using the flight test results.

4. FLIGHT TEST

Flight tests were conducted using the DFVLR test aircraft HFB 320. The objective was to demonstrate the feasibility of the proposed sensor diagnosis concept under a realistic flight environment. The flight tests are partitioned into two parts according to the outlined concept. During the first part data about the parameter variation and disturbance effect on the observers were collected for the threshold definition problem. In the second part the operation of the failure detection and isolation was demonstrated when sensor failure signals were applied.

4.1. Definition of thresholds

The observer differences deviate from zero due to parameter variations and disturbances. The response to parameter variations becomes high when the control system forces the plant to move dynamically. Therefore, inputs into all 3 command signal paths of fig. 2 are applied successively:

- altitude command with different descent and climb rates
- indicated air speed commands represented as a deceleration procedure
- bank attitude commands.

The test results shown in fig. 5, fig. 6 and fig. 7 are selfexplanatory to a certain extent. Only a few features shall be discussed.

Fig. 5: The descent rate is 800 ft/min, the climb rate is 1500 ft/min. Though during the maneuver the plant moves considerably in altitude H , vertical speed \dot{H} and pitch attitude θ the corresponding observer differences remain small. Only the signal n_θ shows up a certain offset which becomes even clearer in the output $n_{\theta f}$ of the attached low pass filter. This effect may be referred to an unprecise modelling of the actual thrust.

Fig. 6: The deceleration procedure is carried out partly in medium air turbulence. Unexpected is the result that obviously the lateral motion is excited by the deceleration maneuver which theoretically should be a matter of the longitudinal motion only. This effect is due to a peculiar feature of the test aircraft. Some unsymmetries of the aileron system show up when the dynamic pressure is varying. In the stationary case the feedback of the integrated observer difference n_ϕ cancels the unsymmetries effect.

Fig. 7: The bank attitude ϕ moves with a certain rate which is defined within the flight control system. During the transient phases some deviations in the observer differences arise, particularly in n_ϕ .

In all three figures there is a bias in the observer difference n_{ay} . Obviously this is due either to a bias in the lateral accelerometer itself or to a nonhorizontal mounting.

During the test flights an area of heavy air turbulence was sought and found during a descent passing a strong cumulus. The results are shown in fig. 8.

Based on these data the thresholds for the second part of the flight test were fixed as shown in the first column of table 1.

OBSERVER DIFFERENCE THRESHOLDS	SENSOR FAILURE FACTORS
$n_{rT} = 1.0$ °/s	$k_r = 3$ °/s
$n_{pT} = 2.0$ °/s	$k_p = 5$ °/s
$n_{\delta T} = 1.0$ °/s	$k_q = 5$ °/s
$n_{\phi T} = 2.5$ °	$k_\phi = 10$ °
$n_{\theta T} = 1.0$ °	$k_\theta = 3$ °
$n_{HT} = 2.0$ m/s	$k_H^s = 5$ m/s
$n_{HT} = 4.0$ m	$k_H = 50$ m
$n_{VT} = 2.0$ m/s	$k_v = 10$ m/s
$n_{ayT} = 0.5$ m/s ²	$k_{ay} = 5$ m/s ²
$n_{\theta FT} = 0.5$ °	-
$n_{rFT} = 0.25$ °/s	-

TABLE 1: THRESHOLDS AND SENSOR FAILURE FACTORS

4.2. APPLICATION OF SENSOR FAILURES

As represented in fig. 3 and fig. 4 sensor failures are generated by feeding additive signals into one of the two sensor signal paths. This is done successively for each of the sensors of one sensor package. Because the arrangement of the sensor packages is symmetrical with respect to both the control system and the observers, sensor failures are applied to the SP1 only. Consequently the index 1 is omitted in the succeeding representations. The plots of the failure signals are equal but multiplied with an individual factor given in the second column of table 1. The common structure of the failure plots is defined in fig. 9. Since the signal character of the sensor failures is important with respect to their effect on the control system on the one hand and to the failure detectability on the other hand the plot of fig. 9 is partitioned into three different intervals.

Interval "10-60 sek": During this period the failure signal increases on a slight slope, thus simulating a small drift of the physical sensor.

Interval "60-70 sek": A stationary offset is represented.

Interval "70-80 sek": A steep decrease to zero is simulating a high drift rate. Thus a failure signal containing higher frequencies is represented.

The sensor failures were applied during a straight level flight in fairly calm air. Thus the effect of the failures become clear because they are only little disturbed by air turbulence or maneuvers. In fig. 10, fig. 11 and fig. 12 test results are shown respective to failures in the θ -sensor, the VIAS-sensor and the p-sensor. In these figures the plots of the most relevant variables are given showing in the first columns the system responding to the complete failure signal profile without to be interfered by sensor switching. In the respective second columns the operation of the switching logic becomes obvious.

Fig. 10: The additive failure signal in the θ -sensor clearly shows up in the observer difference n_θ and similarly in the filter output $n_{\theta F}$. The aircraft motion itself is almost unaffected by this kind of failure, even during the steep decrease interval. The failed sensor is detected when the filter output $n_{\theta F}$ increases beyond its threshold. At this time the switch command signal appears. At the corresponding moment of fig. 10 b the switching operations take place actually.

Fig. 11: The application of the failure to the VIAS-sensor shows results very much different from those of the previous failure case. Here, the plant variable VIAS follows the faulty measurement in the VIAS-sensor whereas during the low rate drift interval the corresponding observer difference n_v remains close to zero. This means that the observer difference n_v is unusable for the detection of slow drift rate failures. It indicates only higher drift rate failures as demonstrated during the steep decrease interval. But low drift rates have an effect on the observer difference n_θ based on the coupling between speed and pitch attitude within the plant and its model. This relation can be used for the failure indication particularly by the filter output $n_{\theta F}$ since the previously defined disturbance threshold n_{rFT} is small due to the low pass filtering property. Hence, this filter output activates the switching operations in fig. 11 b.

Fig. 12: These plots show an example of a failure in a lateral motion sensor. From a systems theoretical point of view this failure case appears to be similar to that of a failure in the θ -sensor in fig. 10.

Failures of the remaining sensors can be classified either as similar to the failure of the θ -sensor (r-, δ -, β -, a-, sensor) or as similar to the failure of the VIAS-sensor (ϕ -, H-sensor). As far as the low drift in the θ -sensor is concerned, the relation between

bank attitude ϕ and yaw rate r is utilized amplified in the filter output n_{rf} . However, for the H-sensor an equivalent relation is not existing. Hence drift failures in this sensor must be declared as undetectable.

5. FURTHER REMARKS

In the presented concept sensor signals are considered as output signals of individual sensors. But with many flight control systems more than one measurement signal are received from one single hardware unit. For instance, with the presented example the pitch θ and the bank attitude ϕ are obtained at the common inertial platform. Hence, it has to be assumed that a failing platform generates failures in both outputs. But this failure case is also covered because the function of the failure detection is preserved as long as simultaneous failures are located in the same sensor package.

During the flight tests a fixed failure signal profile was used. But in order to assess the performance of the detection concept, worst-case failures have to be applied. A thorough study of this problem is given in [6] where optimal control methods are used in order to define the worst-case failure of any sensor. At this place it can be noticed: The low drift in the VIAS-sensor shown in fig. 11 is of the worst-case kind.

As already mentioned and described in more detail in [6], it is reasonable to choose the observer dynamic as similar to the dynamic of the closed loop control system. But this solution is not optimal in a strict sense. The optimization of the observer gain matrix for the presented failure detection method is still an unsolved problem. But already at this time it can be said that the design as a Kalman filter is not optimal. Neither the performance index should be of a quadratic type nor the measurement noise can be assumed as Gaussian. Actually the optimization has to be done with respect to deterministic, precisely definable sensor failures.

6. CONCLUSION

For the sensor part of a flight control system a diagnosis scheme has been developed. Based on analytic redundancy a duplex sensor configuration supported by deterministic observers achieves the fail-operational capability of a conventional triplex system. The concept was applied to a flight control system of a transport aircraft. Flight tests have shown that in principle the failure detection concept is feasible with commonly used sensors. Only certain failures of the altitude sensor are undetectable due to the fact that the altitude has to be considered as the state of a free output integrator. As far as the operational performance of the detection concept is concerned it can be reported from the flight tests that the performance was at least high enough not to upset the pilots.

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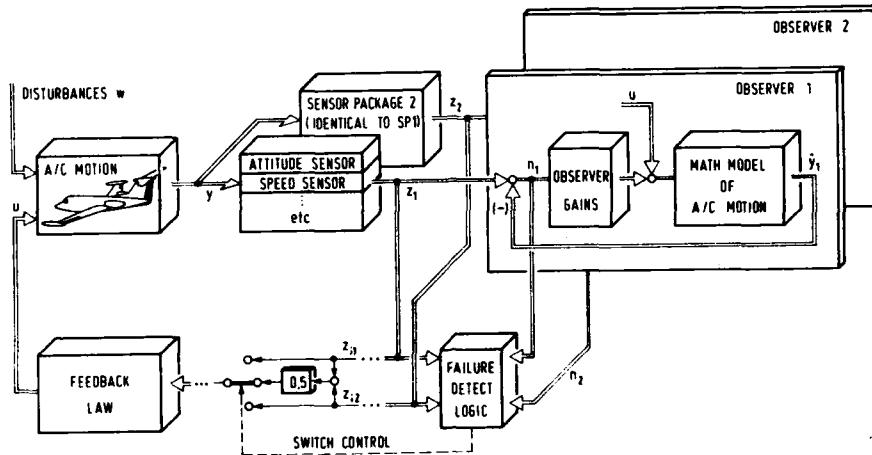


Fig. 1: Flight control system with a fail-op duplex sensor configuration

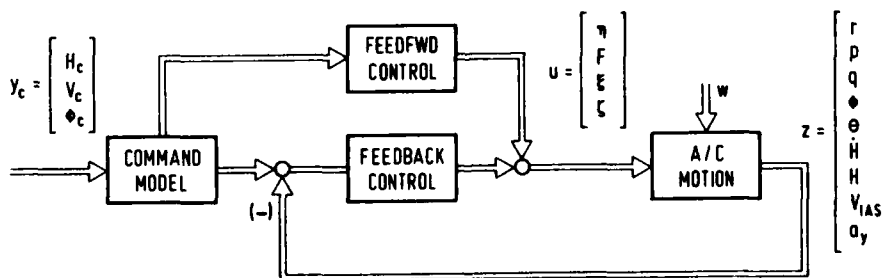
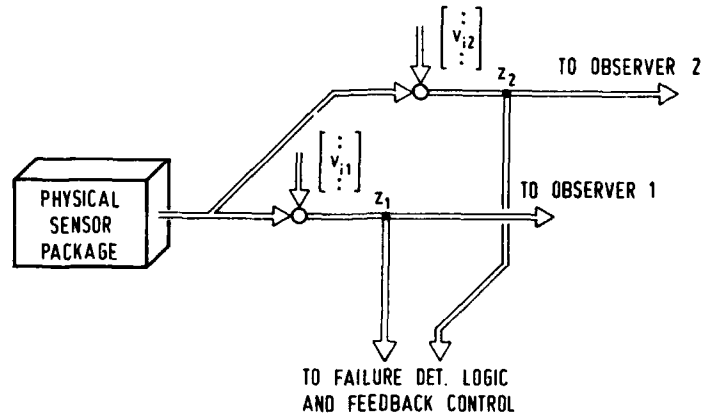


Fig. 2: Flight control system for a transport aircraft



v_{i1} : Additive signal in sensor i of SP1
 v_{i2} : Additive signal in sensor i of SP2

Fig. 3: Software realization of two sensor packages

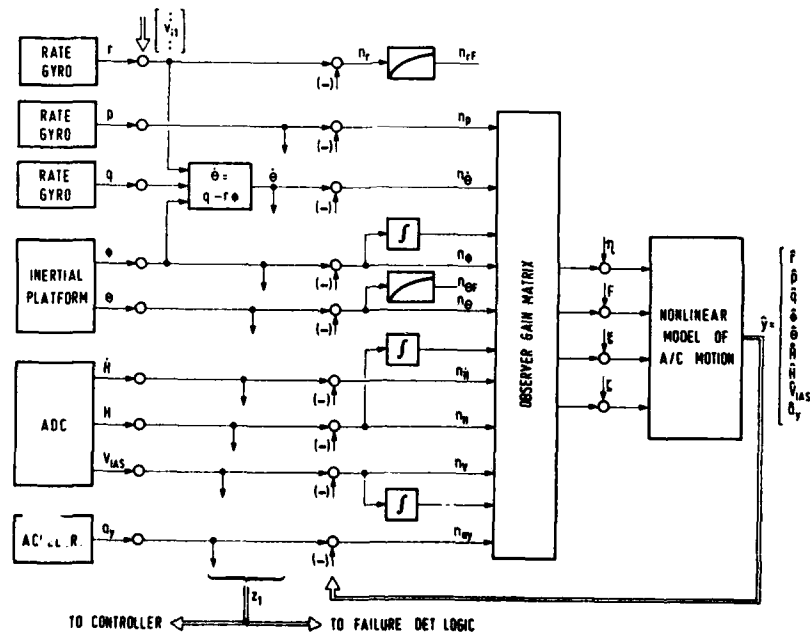


Fig. 4: Experimental sensor and observer arrangement

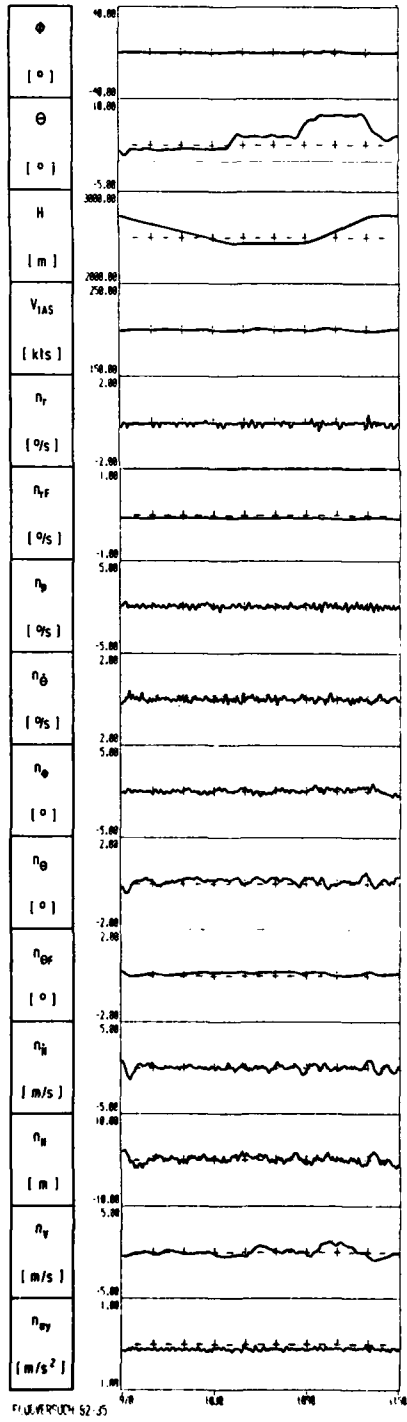


Fig. 5: Response to altitude commands

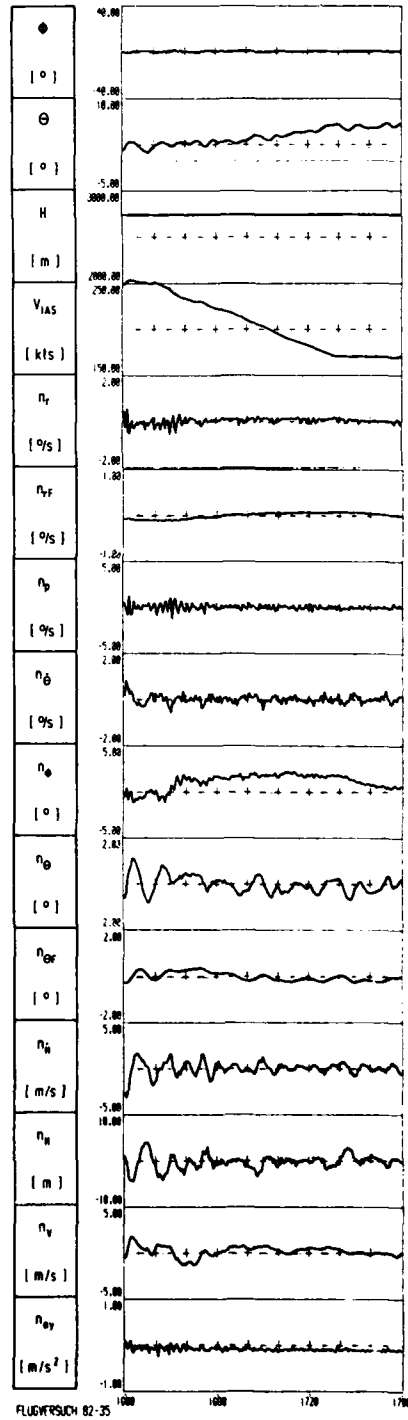


Fig. 6: Response to speed commands

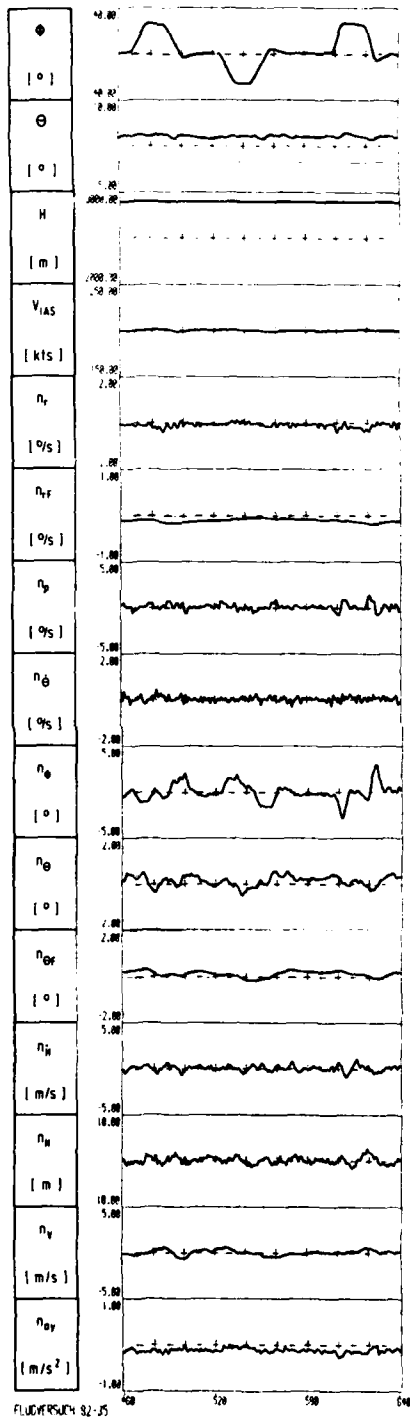


Fig. 7: Response to bank attitude commands

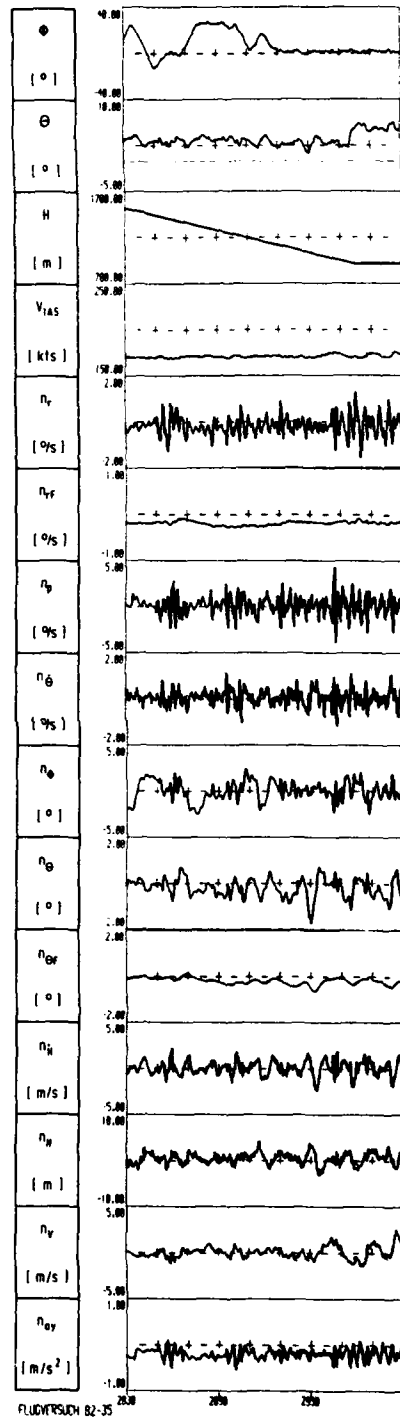


Fig. 8: Response to heavy turbulence during descent including bank attitude commands

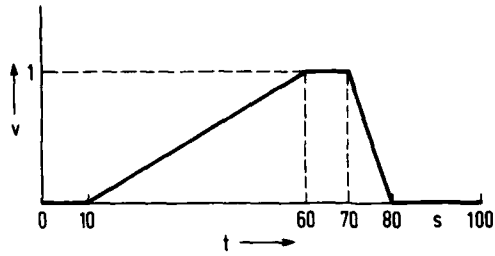
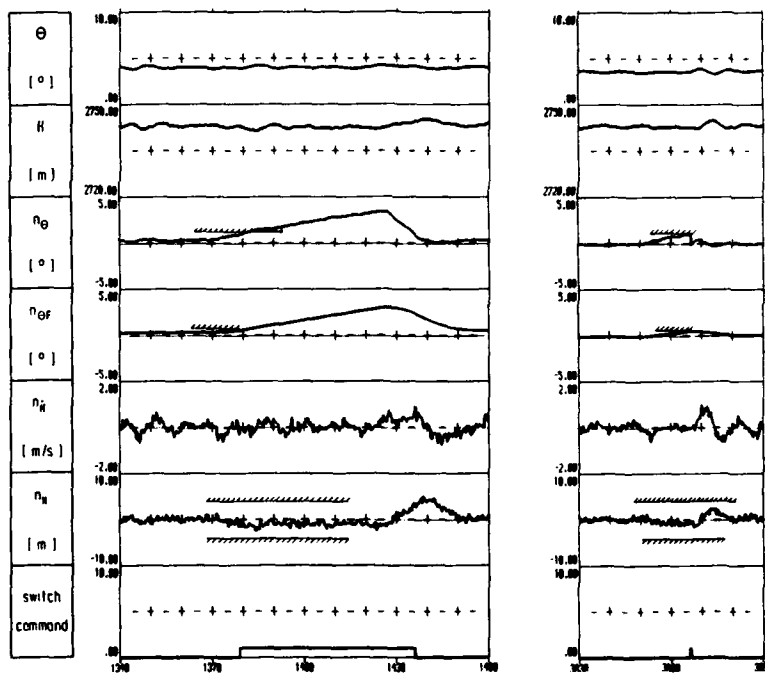


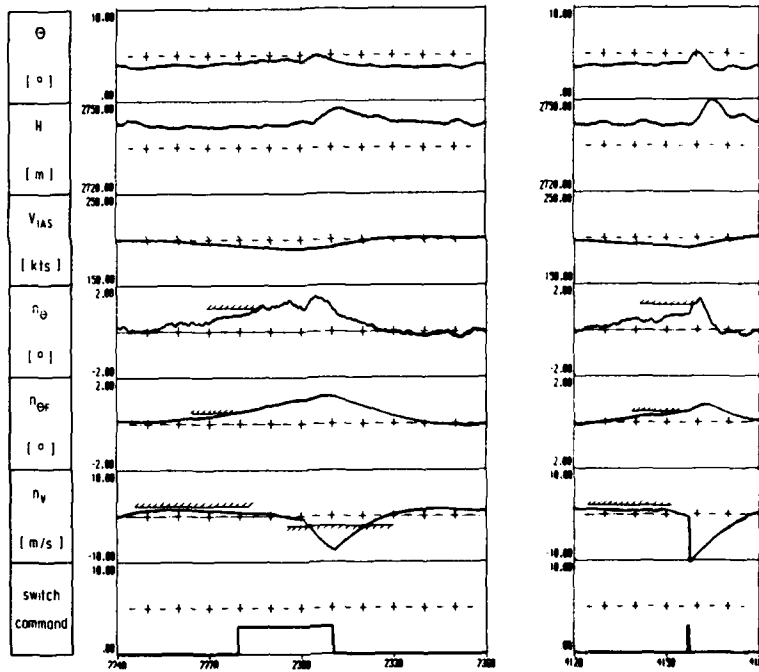
Fig. 9: Common sensor failure profile



a. without switching

b. with switching

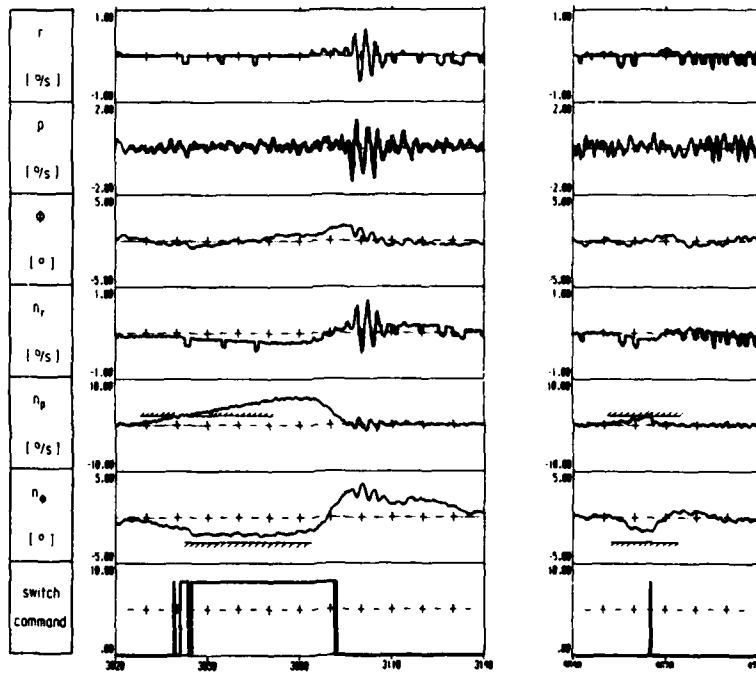
Fig. 10: Response to failures in the θ -sensor



a: without switching

b: with switching

Fig. 10: Response to failures in the V_{IAS} -sensor



a: without switching

b: with switching

Fig. 11: Response to failures in the p-sensor

REPERCUSSIONS DE LA MAINTENANCE INTEGREE SUR LADEFINITION DES EQUIPEMENTS EMBARQUES

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RESUME

La maintenance intégrée a pour but d'assurer la maintenance premier échelon des systèmes d'armes, et de ce fait doit permettre l'identification d'URP (Unités Remplaçables en Piste) en panne et la validation des chaînes fonctionnelles, sans qu'il soit nécessaire de recourir à l'utilisation d'équipements extérieurs au système lui-même. Ce concept est rendu possible par les technologies numériques actuellement utilisées et par une architecture du système d'armes adaptée. En particulier les systèmes développés pour les MIRAGE 2000 se prêtent bien à ce type de maintenance du fait que les équipements sont reliés entre eux par un bus numérique à gestion centralisée, gestion assurée par un seul calculateur dénommé principal ou tactique.

La maintenance intégrée comprend :

- Une maintenance réalisée pendant le fonctionnement opérationnel du système d'armes, basée sur une surveillance permanente du fonctionnement des équipements et l'enregistrement des anomalies pendant le vol.
- Une maintenance réalisée pendant un fonctionnement particulier du système d'armes et permettant des tests plus profonds dans les équipements ainsi que la vérification de tous les échanges d'information s'effectuant par le bus numérique ou par des liaisons point à point analogiques ou digitales. Pendant ce fonctionnement "Maintenance" le système d'armes n'assure plus aucune fonction opérationnelle.

La première partie de l'exposé présente les implications matérielles et logicielles des fonctions de maintenabilité accomplies par les équipements en distinguant :

- les dispositifs exploités pendant le fonctionnement opérationnel
- les dispositifs exploités pendant le fonctionnement maintenance.

La seconde partie de l'exposé, décrit les logiciels de la maintenance intégrée au niveau du calculateur principal du système, gérant le bus numérique.

1 - INTRODUCTION

Dans les systèmes d'armes organisés autour d'un bus numérique multiplexé à gestion centralisée, l'organisation des échanges d'informations est le fait d'un équipement "Maître" (Calculateur principal ou tactique) gérant un flot d'informations circulant sur un support filaire unique. Afin de minimiser le nombre et l'ampleur des équipements de maintenance extérieurs à l'avion, il faut s'efforcer à ce que le moyen privilégié permettant les échanges d'informations "opérationnelles" soit aussi le moyen privilégié d'échange des informations et des procédures de maintenance.

Par ailleurs, et surtout en ce qui concerne les équipements organisés autour d'un opérateur programmé, le test par simulation de fonctionnement doit être remplacé par un test décomposé en plusieurs séquences permettant de vérifier l'intégrité des différentes parties de l'équipement de façon autonome et aussi complètement que possible. D'autre part, le principe de maintenance intégrée doit aboutir à ce que chaque équipement concoure à la maintenance globale du système d'armes.

L'étude de la maintenance intégrée pour un système d'armes donné débute dès la conception des équipements et de l'architecture du système. A cet effet la Société AMD-SA a rédigé un document de spécifications générales de maintenabilité des équipements qui a pour objet d'obtenir une bonne homogénéité des procédures de maintenance s'appliquant à des sous-ensembles fonctionnellement similaires et d'établir un premier schéma de la maintenance générale du système. Ensuite débute une phase d'étude complète de l'application des spécifications équipement par équipement afin d'obtenir une définition détaillée des dispositifs logiciels et/ou matériels de maintenance dans les équipements.

Il est évident que l'ensemble des dispositions de maintenabilité au premier échelon applicables à un équipement conduit nécessairement à des augmentations de matériel et/ou logiciel embarqués. Ces augmentations doivent être contrôlées et réfléchies car elles peuvent entraîner :

- une réduction de fiabilité
- une diminution de la sécurité
- une augmentation des masses et volumes
- une augmentation du prix du matériel.

Il est donc nécessaire dans tous les cas d'examiner ces aspects de façon à réaliser un compromis acceptable entre toutes ces exigences qui sont parfois contradictoires. A titre indicatif nous avons fixé les limites suivantes :

- . Le pourcentage d'instructions directement liées à la maintenance par rapport à l'ensemble des instructions d'un logiciel d'équipement ne doit pas dépasser 15 à 20%.
- . Le pourcentage des éléments matériels directement liés à la maintenance par rapport à l'ensemble des éléments matériels de l'équipement ne doit pas dépasser 10 à 12%.

Ces chiffres sont discutés pour chaque équipement afin d'aboutir au meilleur compromis.

La maintenance intégrée comprend deux parties distinctes :

- Une maintenance réalisée à partir des tests et autotests qui ne perturbent pas le fonctionnement opérationnel du système d'armes et qui est donc effective avion en vol ou au sol. Ces tests et autotests n'ont pas, à eux seuls, un taux de couverture suffisant pour assurer une maintenance premier échelon satisfaisante.
- Une maintenance réalisée à l'aide de procédures perturbantes pour le fonctionnement opérationnel du système d'armes permettant de compléter les tests systèmes et autotests équipements. Cette seconde partie de la maintenance intégrée est obtenue par l'utilisation d'un mode de fonctionnement particulier du système d'armes appelé "fonctionnement maintenance au sol". Dans ce mode, les équipements abandonnent leur fonctionnement opérationnel pour n'effectuer que des tâches strictement maintenance ayant pour objet de compléter la vérification de leurs fonctions internes et de coopérer à la maintenance système : en particulier donner la possibilité de vérifier tous les échanges d'informations effectués soit par le bus multiplexé soit par les liaisons point à point numériques ou analogiques.

Ces deux aspects complémentaires de la maintenance intégrée conduisent à la réalisation de deux familles de dispositifs matériels et logiciels de maintenabilité aussi bien pour les équipements que pour le gérant du bus multiplexé.

2 - MAINTENABILITE DES EQUIPEMENTS

Les équipements d'un système d'armes intégré peuvent être classés en cinq types :

- Les équipements analogiques ou à logique câblée non reliés au bus numérique multiplexé
- Les équipements à processeurs non reliés au bus numérique multiplexé
- Les équipements à logique câblée reliés au bus numérique multiplexé
- Les équipements à processeurs reliés au bus numérique multiplexé
- Les équipements à calculateur universel reliés au bus numérique multiplexé.

La maintenabilité des équipements comprend deux séries de dispositifs :

- des dispositifs qui sont opérants pendant le fonctionnement opérationnel
- des dispositifs qui ne sont opérants que pendant le fonctionnement maintenance.

2.1 - Equipements analogiques ou à logique câblée

Ces équipements sont considérés comme n'étant pas reliés au bus numérique multiplexé et ne sont capables que du seul fonctionnement opérationnel : leur maintenabilité est de type classique.

Ils comportent :

- Un certain nombre de dispositifs matériels surveillant une part plus ou moins importante des fonctions réalisées. Cet ensemble de surveillances doit aboutir à la confection d'une information transmise par un discret vers un autre équipement qui, relié au bus, le transmettra au gérant du système.
- Un dispositif permettant de positionner les informations de sortie de l'équipement
 - . soit à une valeur définie
 - . soit à un écart connu par rapport à une valeur actuelle.
 Ce dispositif doit appliquer les informations de positionnement le plus en amont possible sur les chaînes de traitement. La mise en oeuvre de ce dispositif, perturbant pour le fonctionnement opérationnel de l'équipement peut être déclenché
 - . manuellement par un bouton-poussoir fugitif
 - . électriquement par l'intermédiaire d'un discret provenant d'un équipement connecté au bus numérique.

Le terme "informations de sorties" recouvre aussi bien les informations "électriques" (par exemple valeur de tension) que les informations visualisées au pilote par l'intermédiaire d'aiguilles, de drapeaux etc...

Par ailleurs, il y a lieu de considérer deux types de tests déclenchés :

- Les tests "pilote" dont le déclenchement s'effectue uniquement en cockpit lors du vol ou lorsque l'avion est au sol, sur action du pilote. Pour ce type de test, le retour à la configuration "opérationnelle" de l'équipement à la fin ou à l'interruption du test doit se faire dans un temps inférieur à 1 seconde.
- Les tests "mécanicien" dont le déclenchement s'effectue soit en cockpit soit directement sur l'équipement quel que soit le mode de fonctionnement du système : opérationnel ou maintenance. Pour ce type de test, le retour au fonctionnement hors test s'effectue dans un temps pour lequel il n'y a pas de spécifications particulières.

Pendant le déroulement de la séquence de test, l'information de bon fonctionnement disparaît et ne revient à l'état bon fonctionnement qu'une fois la séquence achevée, c'est-à-dire au moment du positionnement "test" des sorties. Par ailleurs, un discret "test en cours" doit être envoyé vers le système.

2.2 - Equipements à processeur non reliés au bus numérique multiplexé

Ce type d'équipement comporte une partie fonctionnelle de base (fonction capteur) associée à un processeur. La maintenabilité de la fonction de base est assurée de la même manière que pour les équipements analogiques ou à logique câblée.

Les dispositifs de test au niveau du processeur sont les suivants :

- Test de la mémoire programme

L'intégrité du contenu mémoire programme est vérifiée par l'intermédiaire d'une somme de contrôle (check-sum) s'exerçant pendant tout le temps de fonctionnement de l'équipement. Cette somme de contrôle doit être au moins l'addition de toutes les instructions contenues dans la mémoire programme et comparaison du résultat avec une valeur de consigne inscrite dans cette mémoire programme.

- Test des unités arithmétiques et logiques (Unité Centrale)

Ce test est effectué par la réalisation d'un calcul-type ayant pour but de faire fonctionner un maximum de circuits.

- Test du déroulement de programme (Watch dog)

Ce test est effectué sous forme d'une instruction exécutée cycliquement, ayant pour effet de réarmer un dispositif matériel (monostable réarmable). En cas de non-exécution de cette instruction dans la fenêtre de temps de réarmement du dispositif, une information de mauvais fonctionnement est disponible.

- Test des codeurs d'entrée/sortie

Dans le cas où l'équipement possède ces deux dispositifs, un test des codeurs est exécuté suivant la procédure suivante :

- Sur une voie du codeur numérique/analogique : demande de codage en analogique d'une valeur numérique connue par programme.
- Rebouclage de cette information analogique sur une voie du codeur analogique/numérique qui la transforme en numérique.
- Comparaison du résultat avec la valeur d'origine.

Dans le cas où l'équipement ne possède qu'un codeur analogique/numérique le test s'effectue par codage, sur une voie, d'une valeur de tension de référence interne à l'équipement et comparaison avec la valeur numérique témoin correspondante.

L'ensemble de ces autotests s'effectue en tâche de fond programme et ne perturbe donc pas le fonctionnement opérationnel de l'équipement, ils contribuent à l'élaboration d'un signal de bon fonctionnement global.

Traitement du résultat des autotests

Autotests fonctionnels

Ces autotests s'exercent sur les éléments de l'équipement autres que le processeur proprement dit. Pour chacun d'eux, un filtrage peut être effectué par l'équipement lui-même. L'indication de panne vers les équipements périphériques doit être cohérente avec le filtrage, c'est-à-dire que ce n'est qu'à l'issue du filtrage que l'on positionnera les informations d'état à bon ou à mauvais fonctionnement.

Autotests Processeur

On classe dans cette catégorie les autotests s'exerçant uniquement sur le processeur :

- Tests UAL ou UT
- Somme de contrôle
- Test de parité

Test de la zone mémoire de travail si celui-ci s'exerce en permanence.

Tous ces autotests, en cas de déclenchement ont pour effet l'arrêt du programme sans qu'aucun filtrage ni temporisation ne soit effectué, et l'indication de panne doit être immédiate.

Le chien de garde peut déclencher de deux manières :

- d'une part, suite à un arrêt programme occasionné par le déclenchement d'un des autotests précédents.
- d'autre part, par la non exécution de l'instruction de réarmement du chien de garde dans le cas où les autotests précédents ne déclenchent pas.

Ainsi, le chien de garde est, indirectement, le OU logique des résultats des autotests du processeur.

La retombée du chien de garde doit entraîner l'arrêt du programme. Lorsqu'un autotest est déclenché entraînant l'arrêt du programme, plusieurs cas sont possibles :

- Lorsque le programme est enregistré dans une mémoire fonctionnant en cycles "lecture-écriture" (tores ...) l'arrêt doit être définitif jusqu'à la mise hors tension du système d'armes.
- Lorsque le programme est enregistré sur une mémoire "morte" (ROM, PROM, REPRON, etc.) l'arrêt programme
 - doit être définitif jusqu'à la mise hors tension du système d'armes pour les équipements ayant besoin du "passé" pour continuer de fonctionner correctement.
 - peut être temporaire pour les équipements travaillant en temps réel et n'ayant pas besoin du "passé" pour travailler correctement : dans ce cas, une séquence de réinitialisation l'équipement peut être entreprise et réalisée une ou plusieurs fois.
 - Si, au terme d'un certain nombre de tentatives de réinitialisation il est impossible pour le processeur de travailler correctement, l'arrêt programme sera définitif jusqu'à la mise hors tension du système d'armes.
 - Si la procédure de réinitialisation du processeur aboutit à un fonctionnement correct l'équipement est réinitialisé normalement avec disparition de l'indication de panne.
- Il faut remarquer qu'une coupure réseau sera interprétée par les équipements comme une remise sous tension après coupure du système d'armes.
- En cas d'arrêt programme, les sorties analogiques (hors bus) de l'équipement sont positionnées dans un état non dangereux ou préférentiel
 - pour la sécurité de vol
 - pour les équipements périphériques.

Tests à l'initialisation du processeur

A l'initialisation du processeur, l'équipement, en tâche principale, effectue les autotests suivants :

- Test des Unités Arithmétiques et Logiques
- Somme de contrôle de la mémoire programme
- Test de la zone mémoire de travail.

L'ordre dans lequel s'effectuent ces autotests importe peu. Lorsque le processeur a satisfait à ces contrôles, le programme est lancé et le chien de garde est armé pour la première fois.

Ensuite, les autotests processeur sont exécutés en tâche de fond (à l'exception du test de la zone mémoire de travail dans la plupart des cas).

2.3 - Equipements à logique câblée reliés au bus numérique multiplexé

Le couplage au bus numérique multiplexé s'effectue par l'intermédiaire de deux éléments :

- Le coupleur standard du bus (COS) qui permet de recevoir et d'émettre les informations sur le bus. Ce dispositif est commun à tous les équipements reliés au bus.
- Le coupleur sous-système (CSS) qui permet l'organisation des échanges entre le COS et l'équipement lui-même, et contient en particulier les tables d'échanges bus.

La maintenabilité des équipements à logique câblée reliés au bus est la même que celle décrite au paragraphe 2.1 mais le fait d'être relié au bus permet la réalisation de dispositifs complémentaires de test utilisés pendant le fonctionnement opérationnel ou le fonctionnement maintenance du système d'armes.

Maintenabilité pendant le fonctionnement opérationnel

En plus des dispositifs décrits au paragraphe 2.1 ce type d'équipement possède les possibilités suivantes :

- Surveillance des niveaux d'alimentation pour l'alimentation du COS. Les alimentations du COS sont surveillées de telle manière que lorsqu'ils n'ont plus les tolérances permettant d'assurer un fonctionnement correct du coupleur bus, ils inhibent l'émission sur le bus.
- Test de connexion : Ce test a pour objet de reboucler une information reçue par le COS sur ses circuits d'émission et retransmission sur le bus. Ce test est piloté par le gérant et n'a pour rôle que de vérifier le bon fonctionnement des circuits du coupleur de bus. Ce test est cyclique.
- Mot de validité et d'état
Chaque équipement relié au bus doit élaborer et transmettre sur le bus à destination du gérant et des périphériques un ou des mots de validité et d'état. Les informations de validité déterminent la validité des informations qui sont élaborées par l'équipement. Les informations d'état sont des informations de panne. Il existe donc une relation entre les informations de validité et d'état. Un équipement constitué d'une seule URP doit envoyer un mot d'état dont les bits représentent le résultat de chaque autotest permanent dont il est capable et un bit de synthèse de panne. Ces bits seront traités ultérieurement au niveau calculateur principal ou gérant. Un équipement constitué de plusieurs URP envoie un ou plusieurs mots d'états qui doivent être le compte rendu du bon fonctionnement ou de la panne de chaque URP de l'équipement, la logique de traitement étant effectuée dans l'équipement lui-même.
• peuvent éventuellement être le compte rendu du résultat des autotests permanents des équipements.

Les bits non utilisés des mots d'état sont forcés à l'état "Bon Fonctionnement"

Maintenabilité pendant le fonctionnement maintenance

Le fonctionnement maintenance est réalisé principalement par une trame d'échange bus adaptée aux besoins de la maintenance. Les équipements à logique câblée ne peuvent pas changer de mode de fonctionnement. Cependant, un certain nombre de dispositifs sont prévus.

- Test des entrées analogiques
Par analogique, il faut comprendre : l'ensemble des informations qui ne sont pas échangées par le bus numérique multiplexé. L'équipement doit être capable de coder toutes les informations qu'il reçoit en dehors du bus (discrets, tension-synchros, analogiques etc..) et de les retransmettre sur le bus. Les informations qui ne sont pas envoyées sur le bus lors de la trame opérationnelle d'échanges d'informations le seront sur la trame "maintenance".
- Test des sorties analogiques
Les équipements doivent avoir la possibilité de positionner leurs sorties analogiques à certaines valeurs à partir de commandes parvenant par le bus. Le positionnement peut avoir pour origine :
 - un mot bus reçu positionnant une ou plusieurs sorties
 - un mot bus déclenchant un ou plusieurs dispositifs de positionnement dans le cas où il n'y a pas de recopie directe d'informations reçues par le bus. Les positionnements se font :
 - soit à une valeur définie
 - soit à un écart par rapport à une valeur actuelle.

Pour les équipements ayant une face parlante, les éléments sur lesquels on peut faire une lecture visuelle sont considérés comme sortie analogique.

- Rebouclage des sorties analogiques

Pour améliorer la qualité du diagnostic d'URP en panne, il est demandé que l'équipement soit capable de reboucler sur le digibus à travers les codeurs, toutes les informations de sortie analogiques et le rebouclage doit être fait le plus en aval possible.

2.4 - Équipements à processeur reliés au bus numérique multiplexé

Ces équipements sont capables d'un fonctionnement maintenance différent du fonctionnement opérationnel et constituent la majorité des équipements des systèmes d'armes modernes.

Maintenabilité pendant le fonctionnement opérationnel

Ces équipements peuvent disposer de tests déclenchés globaux, ils s'exécutent dans les mêmes conditions que décrit au paragraphe 2.1 pour les équipements analogiques ou à logique câblée. L'information test en cours doit être inscrite, de plus, dans le mot de validité et d'état. Un certain nombre de dispositifs d'autotest sont développés pour surveiller le fonctionnement hors processeur de ces équipements. De plus, la surveillance des alimentations du COS doivent aboutir, en cas de défaut détecté, à l'inhibition de la fonction émission sur le bus.

Test du processeur

Ces tests sont les mêmes que ceux décrits au paragraphe 2.2, à savoir :

- . Test de la mémoire programme.
- . Test des limites arithmétiques et logiques.
- . Test du déroulement de programme.
- . Test des codeurs d'entrée/sortie.

La sanction de ces autotests doit aboutir non seulement à l'arrêt programme en cas de détection d'erreur mais aussi à l'inhibition du dialogue bus en émission. Les possibilités de réutiliser un équipement défaillant décrites au paragraphe 2.2 sont toujours valables ici et en cas de réinitialisation correcte, l'inhibition de l'émission bus doit être levée. En fait, dans la plupart des cas, c'est le chien de garde qui inhibe ou désinhibe le dialogue bus suivant qu'il indique "panne" ou "bon fonctionnement".

Test à l'initialisation du processeur

Ces tests sont les mêmes que ceux décrits au paragraphe 2.2. Le dialogue bus ne sera autorisé que si le chien de garde indique "bon fonctionnement".

Deux cas sont possibles :

- . Le chien de garde est forcé à "Bo. fonctionnement" pendant la phase d'initialisation et donc le dialogue digibus est possible mais aléatoire. Dans ce cas, pendant la phase d'initialisation, l'équipement doit écrire dans son MVE une information "initialisation en cours" qui disparaîtra quand le programme sera lancé.
- . Le chien de garde n'est armé qu'après lancement du programme. Dans ce cas, le dialogue digibus de l'équipement sera toujours valide.

Tests bus

Ces équipements sont capables

- . du test de connexion (voir paragraphe 2.3).
 - . du test conversationnel sur une position d'entrée/sortie.
- Ce test consiste à effectuer les opérations suivantes :
- Réception d'un mot de test lors d'un cycle d'échange et rangement dans une mémoire "Réception".
 - Transfert de ce mot, sans traitement ni calcul, dans une mémoire "Emission".
 - Émission de ce mot sur le bus lors du cycle d'échange bus suivant.
- La valeur du mot test change à chaque réception dans l'équipement. Par ce moyen, on complète le test du COS et on donne la possibilité de vérifier que le processeur de l'équipement établit correctement le dialogue en réception comme en émission. Le gérant du système pilote ce test et en déduit des pannes éventuelles.
- . Mot de validité et d'état.
- Le mot de validité et d'état est élaboré et émis cycliquement sur le bus numérique dans les mêmes conditions que décrit dans le paragraphe 2.3

Maintenabilité pendant le fonctionnement maintenance

Le fonctionnement maintenance de ce type d'équipement est déclenché par une commande délivrée par le gérant du bus, valable pour tout le système d'armes. A la réception de cette commande, les équipements abandonnent leur fonctionnement opérationnel pour n'effectuer que des tâches purement maintenance mises en oeuvre à l'aide de la trame d'échange appelée "Trame maintenance au sol".

Le fonctionnement maintenance est divisé en deux parties :

- . Les fonctions logicielles maintenance de base dont l'exécution est déclenchée dès la commande délivrée par le gérant du bus.
- . Les fonctions logicielles déclenchées par l'opérateur en fonction de ses besoins.

Fonctions logicielles de base : elles permettent

- . Le codage et l'émission sur le bus multiplexé des informations analogiques d'entrée (par analogique il faut entendre toutes les informations échangées en dehors du bus).
- . Positionnement des sorties "analogiques" de l'équipement à partir d'ordres provenant du bus.
- . Rebouclage des sorties "analogiques" permettant le lever de doute entre deux URP lors d'une recherche de panne.

Fonctions logicielles déclenchées

- . Tests déclenchés. Ces tests ont leurs algorithmes résidents dans la mémoire programme de l'équipement et sont déclenchés par une commande contenue dans un mot parvenant par le bus.
- . Chargement de logiciel de test. Les équipements sont capables de recevoir par le bus un logiciel qui est chargé dans la zone mémoire vive, et est exécuté par l'unité centrale sur commande parvenant via le bus.

- Edition du contenu mémoire programme. Cette possibilité est demandée pour tous les équipements et est surtout utilisée au 2ème échelon de maintenance pour le contrôle bit à bit de l'intégrité du contenu mémoire programme. Les équipements ayant une mémoire à tores ou EEPROM doivent pouvoir non seulement éditer leur contenu mémoire par le bus mais aussi pouvoir être chargés à partir du bus. Ce dispositif de lecture-écriture peut être utilisé au 1er échelon.
- Test conversationnel complet. Ce test consiste à :
 - Recevoir un certain nombre de messages permettant de remplir complètement toutes les mémoires "Réception" des équipements. Les informations ou les messages sont reçus avec les adresses correspondant à l'ensemble des adresses réception que l'équipement utilise. Le contenu est vérifié par l'équipement à l'aide d'une somme de contrôle.
 - Emettre des informations sous forme de messages-tests en utilisant toutes les adresses émission de l'équipement concerné. Le gérant du bus vérifiera, par une somme de contrôle que tous les messages ont correctement été émis par l'équipement sous test.
- Test du chien de garde. C'est un test ayant pour but de faire déclencher le chien de garde et donc d'inhiber le dialogue bus en émission de l'équipement.

2.5 Equipements à calculateur universel reliés au digibus

Ce type d'équipement comporte un calculateur travaillant avec une mémoire de masse extérieure, généralement constituée par un disque ou une bande magnétique. La particularité de cette structure est que les zones "mémoire programme" et "mémoire de travail" ne sont plus fixes mais variables et imbriquées entre elles. Il s'en suit que les spécifications exposées au paragraphe précédent ne s'appliquent plus de la même manière au niveau processeur.

- Test de la mémoire programme résident

Il est assuré par une somme de contrôle s'exerçant en fond de tâche programme.

- Test de la mémoire programme exécutable

Il est effectué par une somme de contrôle

- d'une part à la fin du chargement dans la mémoire interne du calculateur du programme à exécuter. Cette somme de contrôle doit s'effectuer en tâche principale avant tout lancement du programme.
- d'autre part, pendant tout le temps de fonctionnement de l'équipement une fois le programme lancé (en fond de tâche).

Une fois le programme lancé, on peut confondre les deux sommes de contrôle et n'en effectuer qu'une seule en additionnant les instructions :

- du programme résident
- du programme exécutable

et en comparant le résultat obtenu avec la somme des deux valeurs de consigne.

- Test du dialogue avec la mémoire de masse

La mémoire de masse est utilisée dans deux buts :

- Incrire dans la mémoire interne du calculateur un programme à exécuter.
- Incrire dans la mémoire de masse un certain nombre d'informations qui seront traitées ultérieurement.

Le coupleur mémoire de masse a donc une importance essentielle et il faut s'assurer en permanence que les échanges d'informations se font correctement dans les deux sens. Le test de ce coupleur est réalisé par l'inscription d'informations test sur la mémoire de masse puis relecture avec vérification. Ce test est effectué en permanence pendant le fonctionnement opérationnel de l'équipement. La même règle est appliquée lorsque l'équipement est relié directement à une unité de gestion d'entrée-sorties autonome (unité de gestion bus par exemple) ou un périphérique standard (télé-imprimeur).

Aux différences décrites ci-dessus, les spécifications décrites au paragraphe précédent s'appliquent entièrement à ce type d'équipement.

3 - LOGICIELS DE MAINTENANCE INTEGREE DU GERANT BUS SYSTEME

3.1 - Logiciels exécutés pendant le fonctionnement opérationnel

3.1.1 - Test de connexion

Ce test est destiné à vérifier que les coupleurs standards de bus (COS) fonctionnent correctement. Le gérant bus système vérifie cycliquement que tous les équipements reliés au bus numérique multiplexé envoient un écho-test correct.

3.1.2 - Test conversationnel sur une position

Ce test est destiné à vérifier que les équipements reliés au bus numérique multiplexé dialoguent correctement en renvoyant une information-test. Le résultat de ce test n'est pris en compte, pour un équipement, que dans la mesure où son test de connexion n'indique pas de panne.

3.1.2 - Comptes rendus de maintenance

Les Comptes Rendus de Maintenance (CRM) permettent, en exploitant les résultats des autotests permanents non perturbants des équipements, de connaître l'état du système d'armes. L'intérêt de ce dispositif est d'autant plus grand que le taux de couverture des autotests des équipements est élevé. Le logiciel des Comptes Rendus de Maintenance a deux modes de fonctionnement :

- Les Comptes Rendus de Vol (CRV) : c'est la mémorisation et la datation de tous les événements de panne détectés par les autotests entre le moment du décollage et l'instant de l'atterrissage.

Ils permettent d'obtenir, au sol :

- . la visualisation des équipements ou des URP qui sont tombés en panne pendant le vol, et qui ne sont pas revenus à l'état de Bon Fonctionnement au moment du toucher des roues (atterrissage).
 - . la visualisation de l'historique de vol, qui permet de connaître dans quel ordre et à quelles dates (en temps de vol) les équipements sont passés à l'état de panne.
- Les Comptes Rendus Sol (CRS) : c'est la possibilité de présenter au mécanicien ou au pilote l'état instantané du système d'armes du point de vue des pannes, avion au sol. Il n'y a aucun enregistrement dans la mémoire du gérant et présentation :
- . soit des équipements ou des URP en panne
 - . soit des mots d'information de panne de chacun des équipements suivant une sélection particulière

en temps réel.

L'exploitation des Comptes Rendus de Maintenance ne peut s'effectuer qu'au sol.

3.1.3.1 - Constitution des Comptes Rendus de Maintenance

Les Comptes Rendus de Maintenance comportent plusieurs informations :

- un Mot d'Etat et de Panne (MEP)
- un code Equipement
- un code de type de panne
- un mot de datation.

3.1.3.1.1 - Mot d'Etat et de Panne

Les mots d'Etat et de Panne (MEP) sont constitués à partir

- des Mots de Validité et d'Etat (MVE)
- des mots de mode
- des surveillances exercées par le gérant sur le dialogue digibus des équipements
- des surveillances particulières.

. Mot de Validité et d'Etat (MVE)

Un MVE est émis par chaque équipement connecté au digibus.

Ce mot comprend deux parties :

- Une partie "validité" qui indique aux équipements utilisateurs des informations générées sur le digibus que ces informations sont utilisables ou non. Cette partie du MVE n'est pas prise en compte pour l'élaboration d'un Mot d'Etat et de Panne (MEP).
- Une partie "état" (sous-entendu de pannes). Cette partie contient le compte rendu du résultat des autotests que l'équipement déroule en permanence et éventuellement un bit de synthèse de panne par URP pour les équipements multi-URP.

. Mot de Modes (MM)

Ce mot indique le mode dans lequel l'équipement fonctionne. C'est dans ce mot que l'on trouve l'information "Test en Cours". Le mot de mode, ou une partie de ce mot, est pris en compte pour l'élaboration d'un MEP.

. Surveillance Digibus

Le dialogue Digibus de tous les équipements qui y sont connectés est surveillé par le gérant du système qui détermine des pannes dialogue éventuelles.

. Surveillances Particulières

Ces surveillances sont de deux ordres :

- Les équipements non connectés au digibus ont un certain nombre d'autotests qui renseignent sur leur état. Ces autotests concourent à l'élaboration d'un ou de plusieurs discrets de bon fonctionnement qui sont transmis
 - . soit vers le gérant du système
 - . soit vers un équipement qui, relié au Digibus, le transmettra vers le gérant.
 Le gérant a donc la possibilité de confectionner un MEP pour l'équipement émetteur des discrets de bon fonctionnement en question.
 - Les équipements envoient soit vers le gérant soit vers d'autres équipements un certain nombre d'informations analogiques ou Digibus. Les équipements récepteurs peuvent faire des tests de vraisemblance sur les informations qu'ils reçoivent, et peuvent donc déterminer des pannes sur les "émetteurs".
- Les résultats de ces surveillances sont :
- . soit directement élaborés par le gérant lorsqu'il effectue lui-même ces contrôles
 - . soit disponibles dans un équipement relié au digibus qui les retransmettra vers le gérant.

Le MEP est donc le rassemblement de toutes ces informations sur 2 mots de 16 bits dans lesquels on trouve :

- des bits qui donnent le résultat des autotests d'un équipement
- des bits qui donnent le résultat des surveillances effectuées sur un équipement par ses périphériques ou le gérant du système.

3.1.3.1.2 - Code Equipement

Chaque équipement faisant l'objet d'un CRV possède un code qui n'a de valeur que pour le gérant du système. Les équipements sont ainsi hiérarchisés implicitement dans la table des Comptes Rendus de Maintenance. Ce code équipement comprend 8 bits.

3.1.3.1.3 - Code de type de panne

4 types de panne peuvent être déterminés. Le code de type de panne comprend 2 bits.

Pannes de type 1 : Ce sont les pannes déterminées par les autotests des équipements, ou par le gérant sur leurs tests dialogue. Ce type de panne permet de considérer que la maintenance est réalisée par l'échange de l'URP fautive sans qu'il soit nécessaire d'effectuer des recherches complémentaires.

Pannes de type 2 : Ce sont toutes les pannes autres que celles déclarées par les équipements eux-mêmes par l'intermédiaire de leurs autotests ou le gérant sur la qualité du dialogue. Ce sont donc des pannes détectées par les "surveillances particulières". Dans ce cas, il y a peut-être lieu d'effectuer des recherches complémentaires avant de déposer l'URP.

Pannes de type 0 : Dans le cas où il y a panne puis retour à bon fonctionnement, il y a indication de type de panne 0 au moment du retour à l'état bon fonctionnement.

Pannes de type S : Dans le cas où une URP a fait l'objet de dix enregistrements de panne pendant le vol, elle est présentée avec indication de type de panne "S" pour "SATURANTE", quel que soit le type de panne du dernier enregistrement : "1", "2" ou "0".

3.1.3.1.4 - Mot de Datation

En vol, chaque compte rendu de maintenance est enregistré en même temps que l'heure à laquelle la panne correspondante a eu lieu. Cette heure est exprimée en nombre de cycles longs d'échange. Le LSB de ce mot a une valeur de 160ms dans le cas du MIRAGE 2000.

3.1.3.2 - Constitution des Comptes Rendus de Vol (CRV)

Les Comptes Rendus de Vol sont la mémorisation des Comptes Rendus de Maintenance entre le moment du décollage et le moment de l'atterrissage, dans la mémoire du calculateur gérant. Cette table est mémorisée dans une zone RAM ou RAX protégée.

3.1.3.2.1 - Datation

Le compteur servant à dater les CRV est mis à zéro une première fois lors du passage sur la position "Navigation" du Sélecteur de Mode de Navigation. Lors du décollage, matérialisé par l'information "Train détendu" le logiciel des CRV doit enregistrer la valeur de ce compteur. Ceci permet de connaître l'heure de décollage par rapport au passage en "Navigation". Dans le même temps, ce compteur sera remis à zéro afin de pouvoir dater les CRV par rapport au moment du décollage.

3.1.3.2.2 - Remise à zéro de la table des CRV

La table des CRV est remise à zéro lors de la première information "train détendu" consécutive à la mise sous tension du Système d'Armes.

3.1.3.2.3 - Enregistrement des CRV

Au moment du décollage, (train détendu) il y a :

- Remise à zéro de la table des CRV (du vol précédent)
 - Enregistrement de l'heure de décollage (par rapport au passage en NAV)
 - Remise à zéro du compteur horaire
 - Comparaison des MEP constitués dans le premier cycle long suivant le décollage et comparaison avec le profil "Bon Fonctionnement" de chacun d'eux.
- Ensuite, il y a constitution et mémorisation d'un CRV :
- lors du passage d'un MEP de l'état Bon Fonctionnement à l'état panne
 - lors du passage d'un MEP d'un état de panne à un autre état de panne
 - lors du passage d'un MEP d'un état de panne à l'état de bon fonctionnement.
- Chaque MEP fait l'objet d'une comparaison par cycle long d'échange (160ms). La mémorisation des CRV s'arrête lorsque le contact de train indique "Sol" (Train écrasé).

3.1.3.2.4 - Limitations

- Un MEP donné ne peut être enregistré que 10 fois au maximum au cours d'un vol
- Le nombre total de mémorisations de CRV est limité à 128 pour un vol.

3.1.3.2.5 - Voyant magnétique

Tout enregistrement dans la table des CRV fait basculer un voyant magnétique sur le tableau mécanicien. L'enregistrement de l'heure de décollage ne fait pas basculer ce voyant qui est réarmable manuellement au sol.

3.1.3.3 - Constitution des Comptes Rendus Sol (CRS)

Les Comptes Rendus Sol sont constitués par une table non sauvegardée dans le gérant du Digibus. Cette table est constituée par les comptes rendus de maintenance instantanés de tous les équipements du Système d'Armes.

Dans le CRS :

- La datation est forcée à zéro
- Il n'existe pas de panne de type "SATURANTE".

3.1.3.4 - Procédure de visualisation des Comptes Rendus de Vol

La visualisation des Comptes Rendus de Vol s'effectue sur la Tête Basse, au sol.

3.1.3.4.1 - Visualisation des pannes existant à la fin du vol

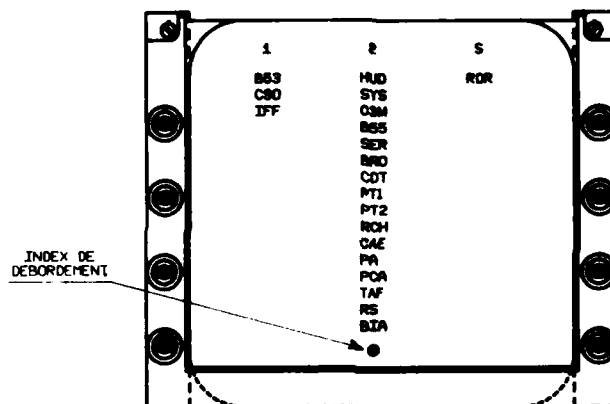
La liste des URP en panne au moment de l'atterrissage est présentée en tête basse.

Il apparait :

- Une 1ère colonne de mnémoniques de couleur verte, surmontée du chiffre 1 tracé en vert. Les mnémoniques de cette colonne correspondent aux URP en panne de type 1.
- Une 2ème colonne de mnémoniques de couleur ambre, surmontée d'un 2 ambre. Ces mnémoniques indiquent les URP en panne de type 2.
- Une 3ème colonne de mnémoniques de couleur rouge surmontée d'un S rouge. Ces mnémoniques indiquent les URP en panne saturante (type S).

Chaque colonne comporte au plus 16 mnémoniques. Lorsque le gérant demande la visualisation de plus de 16 mnémoniques dans une colonne, la tête basse ne visualise que les 16 premiers et présente un astérisque en bas de colonne. L'opérateur doit alors se reporter à l'historique de vol pour connaître la liste complète des URP ou équipements en panne.

Si aucune URP ou équipement n'est en panne, seuls apparaissent le 1 vert, le 2 ambre et le S rouge.



VISUALISATION DES PANNES EXISTANT EN FIN DE VOL

3.1.3.4.2 - Visualisation de l'historique de vol

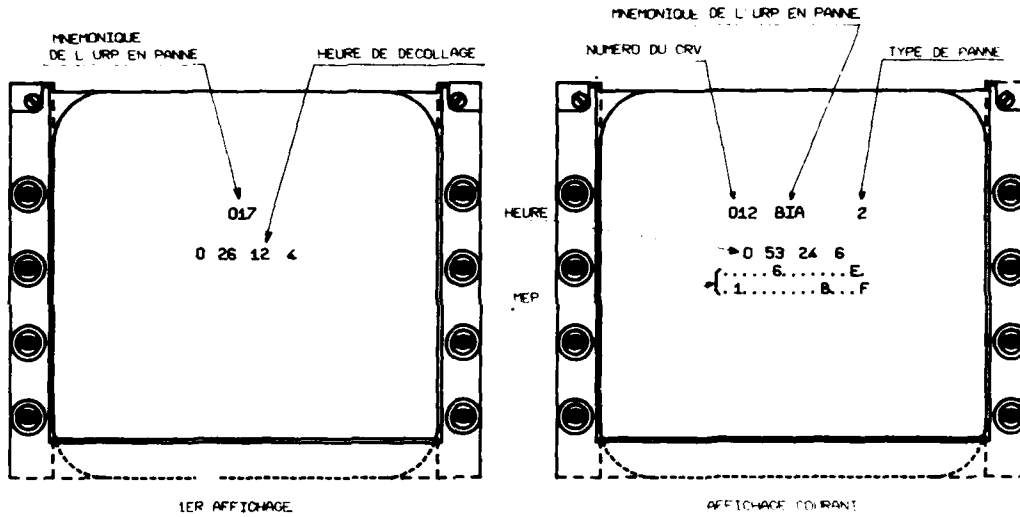
L'historique de vol est constitué par l'ensemble des CRV qui ont été enregistrés pendant le vol dans l'ordre chronologique de leur mémorisation. Il est présenté en tête basse. Dans ce mode de visualisation le premier affichage est celui du nombre de mémorisations de CRV effectuées en vol, suivi de l'heure de décollage exprimée en heures, minutes, secondes, dixièmes de seconde.

Le second affichage est celui du 1er CRV enregistré, sous la forme de 4 lignes superposées :

- La première ligne indique le numéro d'ordre du CRV, l'équipement concerné et le type de panne (0, 1, 2 ou S)
- La seconde ligne indique l'heure de mémorisation en heures, minutes, secondes, dixièmes de seconde.
- Les 3e et 4e lignes de 16 caractères chacune reproduisent le MEP correspondant. Pour tout bit à 1 dans le MEP on visualise un point et pour tout bit à 0 (c'est-à-dire significatif de panne), le rang du bit dans le mot du MEP.

On peut ensuite dérouler tous les CRV d'un vol dans le sens croissant ou décroissant.

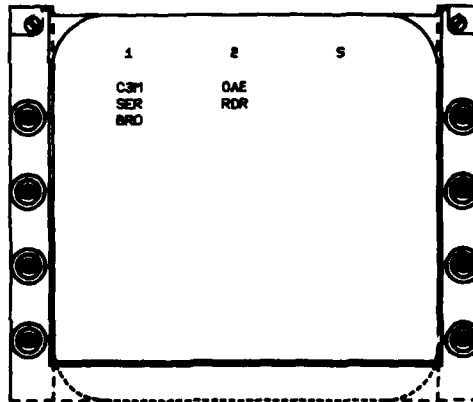
L'heure de décollage est toujours présentée à l'appel de la procédure.
 Si aucune mémorisation de CRV n'a eu lieu en cours de vol, il y aura au moins l'heure de décollage inscrite sur la tête basse.



3.5 - Procédure de visualisation des Comptes Rendus Sol
 Les Comptes Rendus Sol ont pour but de présenter l'état instantané du système du point de vue des pannes. La visualisation s'effectue sur la Tête Basse, au sol.

3.5.1 - Visualisation des Equipements en panne actuelle
 La liste des URP en panne actuelle est présentée sur la Tête Basse.

La visualisation présentée est semblable à la liste des URP en panne des Comptes Rendus de Vol. La colonne des pannes saturantes est vide car une panne actuelle ne peut pas être saturante.



VISUALISATION DES EQUIPEMENTS EN PANNE ACTUELLE

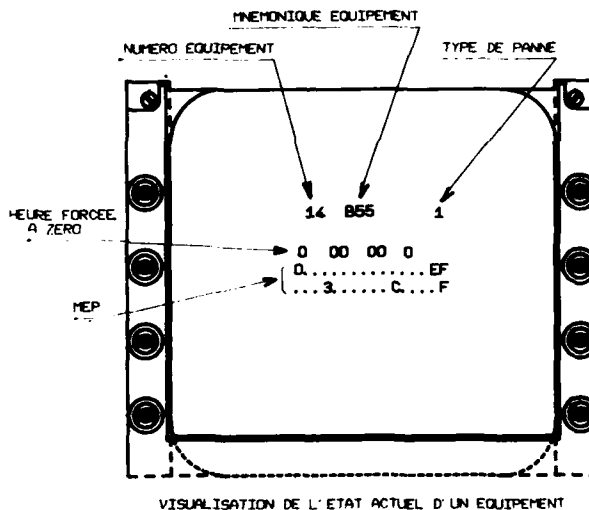
3.1.3.5.2 - Visualisation de l'état actuel d'un Equipement

L'état actuel d'une URP est présenté sur la Tête Basse.

Pour visualiser l'état actuel d'une URP, on frappe au PCN un nombre de 2 chiffres, qui est le numéro dictionnaire de l'équipement auquel l'URP est rattachée. Apparaît alors en tête basse un Compte Rendu Sol sous forme d'un ensemble de 4 lignes.

- . 1ère ligne : numéro de l'équipement, nom de l'équipement, type de panne (1, 2, 0)
- . 2e ligne : date, forcée à 0
- . 3e et 4e lignes : MEP de l'équipement, présenté comme en visualisation de l'historique de vol.

La frappe d'un autre numéro fait apparaître le Compte Rendu Sol correspondant en lieu et place du précédent.



3.1.3.6 - Panne d'un équipement utilisé pour les Comptes Rendus de Maintenance

En cas de panne du calculateur gérant, celle-ci est indiquée par les visualisations opérationnelles. Les mémorisations de CRV déjà effectuées sont perdues. Aucun nouveau traitement ne peut être effectué et il est donc impossible de présenter un compte rendu quelconque. En visualisation de Compte Rendu de Maintenance, le gérant secours provoquera la visualisation des réticules 1, 2 ou 5 avec le mnémonique CP en colonne 1.

En ce qui concerne les autres équipements nécessaires à la visualisation des comptes rendus de maintenance, les pannes éventuelles seront détectées par les autotests déclenchés, plus performants que les autotests permanents.

3.2 - Maintenance complémentaire au sol

Les autotests permanents des équipements et du système ont une certaine limite du fait qu'ils doivent se dérouler sans perturber le fonctionnement opérationnel. Il s'en suit que la profondeur des tests n'est pas toujours suffisante pour détecter et localiser les pannes avec la performance attendue d'un premier échelon.

La maintenance complémentaire au sol doit donc permettre de compléter les autotests permanents en rendant possible la vérification de tous les éléments qui ne peuvent l'être qu'en perturbant le fonctionnement opérationnel du système d'armes. Les principales fonctions à assurer sont les suivantes

- Validation des transmissions d'informations analogiques entre les différents équipements et avec les armes.
- Permettre des tests plus profonds dans les équipements en utilisant des tests déclenchés spécifiques dont les algorithmes sont soit résidents dans les équipements, soit "chargeables" à partir d'une mémoire de masse.
- Qualification exhaustive du dialogue digibus des équipements.

L'exécution de ces différentes fonctions est rendue possible par les logiciels d'aide à la maintenance sol. Ces logiciels sont répartis en deux groupes :

- Les logiciels de base
- Les logiciels complémentaires.

3.2.1 - Logiciels de base

Ces logiciels permettent de donner au mécanicien tous les moyens de dialoguer avec le système afin qu'il puisse mener à bien toutes les opérations de vérification qui lui semblent nécessaires afin de localiser une panne ou de valider une chaîne fonctionnelle.

3.2.1.1 - Logiciel Trame Sol

En configuration maintenance au sol, le système fonctionne suivant un mode particulier.

Il faut considérer deux types d'équipements :

- Les équipements analogiques pour lesquels les opérations de maintenance ne peuvent se faire qu'en utilisant le mode de fonctionnement opérationnel et en simulant à l'entrée un ensemble de paramètres cohérents.
- Les équipements programmés numériques pour lesquels le fonctionnement maintenance est particulier, le programme opérationnel est arrêté au profit d'un programme permettant :
 - . de positionner directement à une valeur donnée par le Digibus les différents paramètres de sortie analogiques (hors Digibus).
 - . de coder sur le Digibus la valeur de toutes les entrées analogiques
 - . d'accueillir dans leur mémoire RAM et d'exécuter des logiciels chargés à partir du Digibus
 - . de dérouler, à partir d'un ordre donné par le Digibus, des tests internes complémentaires.

Les valeurs de positionnement des paramètres de sortie analogiques sont délivrées à l'aide de Codes de Données Maintenance de Positionnement (CDMP).

Les valeurs des paramètres analogiques d'entrée sont données dans les Codes de Données Maintenance de Lecture (CDML).

Le déclenchement des logiciels de test chargés et des tests complémentaires s'effectue à l'aide des Codes de Données Maintenance de Déclenchement (CDMD).

La trame sol est donc le support de la transmission des informations nécessaires à la réalisation des fonctions décrites ci-dessus.

La trame sol comprend :

- Tous les messages émis sur Digibus en trame opérationnelle mais dont la cadence est réduite à une fois par cycle long et dont le contenu n'est pas significatif ; ceci d'une part pour ne pas avoir une trame d'échange trop complexe et trop chargée à gérer, et d'autre part, pour ne pas avoir une charge de calcul trop importante au niveau du gérant du digibus. Cependant, un certain nombre d'équipements demandent que quelques échanges s'effectuent au "rythme opérationnel" afin de pouvoir conserver un fonctionnement maintenance correct.
- Les messages permettant de valider les chaînes analogiques, c'est-à-dire les CDMP et CDML
- Les messages permettant de charger dans les équipements des logiciels de test
- Les messages permettant de déclencher des tests complémentaires chargés ou résidents (CDMD)
- Les messages permettant de lire et/ou d'écrire dans la mémoire du gérant ou des équipements
- Les messages de visualisation maintenance sur le PCN et la Tête Basse.

3.2.1.2 - Logiciel de Dialogue Système

La gestion des CDMP, CDML, CDMD est faite par le mécanicien. Lors du passage de la trame opérationnelle à la trame maintenance sol, les valeurs des CDMP sont telles qu'elles définissent un état initial maintenance pour le système ; a priori, toutes les valeurs sont positionnées à zéro. Les CDMD sont toutes positionnées à zéro. Le logiciel de dialogue permet au mécanicien :

- de positionner les paramètres analogiques à une valeur choisie par lui,
- de lire les valeurs des paramètres analogiques choisis par lui
- de déclencher les tests complémentaires à son initiative.

L'organe de dialogue utilisé est le Poste de Commande de Navigation (PCN) qui possède toutes les commandes nécessaires, ainsi que les éléments de visualisation indispensables. La tête basse est utilisée comme bloc-notes, c'est-à-dire qu'elle permet de conserver l'affichage des huit dernières opérations effectuées au PCN.

De plus, ce logiciel permet de donner au mécanicien toutes les informations nécessaires pour surveiller le fonctionnement maintenance du système d'armes.

3.2.1.3 - Gestion des logiciels complémentaires

Les logiciels complémentaires sont chargés soit dans le gérant soit dans les équipements à partir d'une procédure définie par ce logiciel de gestion. Les logiciels à exécuter sont contenus dans un support informatique interne ou externe à l'avion et connecté au digibus. Ce logiciel de gestion ne concerne que le chargement des logiciels à exécuter. L'exploitation des résultats des traitements effectués par les logiciels chargés est assurée par le logiciel de dialogue système.

3.2.1.4 - Gestion de l'outillage sol

Ce programme de gestion permet le dialogue avec un équipement au sol connecté au digibus.

Il rend possible :

- La lecture de tout ou partie de la mémoire d'un équipement par un outillage sol
- L'écriture en mémoire dans les équipements.

3.2.2 - Logiciels Complémentaires

Les logiciels complémentaires sont contenus dans un équipement interne ou externe au système et chargés dans les équipements à l'aide du programme de gestion des logiciels complémentaires.

3.2.2.1 - Logiciels complémentaires de test système

Ces logiciels sont chargés et exécutés dans le CP et ont pour but d'exécuter un test particulier s'adressant à tout ou partie des équipements du Système d'Armes. Ainsi, le logiciel de Test Conversationnel Complet (TCC) permet de qualifier à cent pour cent la qualité du dialogue digibus de tous les équipements du système d'armes.

3.2.2.2 - Logiciels complémentaires de test équipement

Ces logiciels sont chargés et exécutés dans les équipements y compris le gérant. Ils sont exécutés en interne dans les équipements concernés et ne s'adressent en aucune façon à un de leurs périphériques.

Ils peuvent être considérés comme des autotests internes complémentaires.

Nota : Il faut bien remarquer que la maintenance intégrée ne consiste pas à rechercher une panne par un processus automatique mais à donner au mécanicien tous les outils nécessaires à la conduite d'un dépannage ou à la validation d'une chaîne fonctionnelle.

3.2.3 - Mise en oeuvre des Logiciels de base d'Aide à la Maintenance Sol

Le système d'armes ne peut passer en configuration maintenance qu'à partir du fonctionnement opérationnel.

L'avion étant sous tension, le passage de la configuration opérationnelle à la configuration maintenance ne peut s'effectuer que si :

- Les conditions de sécurité armement sont réunies
- Le train est verrouillé bas
- Le commutateur secondaire du PSM est sur la position "MAINT"
- Le mécanicien a frappé au clavier du PCN le code de demande "Trame Sol".

A la réception de ce code par le gérant, celui-ci émet, en trame opérationnelle et une seule fois, un code d'ordre de commutation en fonctionnement maintenance des équipements. Ensuite, il y a un silence Digibus de 2 cycles longs pour permettre l'initialisation des équipements en fonctionnement maintenance puis émission de la trame maintenance proprement dite.

Le passage de la configuration maintenance à la configuration opérationnelle du système d'armes s'effectue par :

- la mise sur une autre position que "MAINT" du rotacteur secondaire du "PSM"
- la mise hors tension puis sous tension du système d'armes.

4 - CONCLUSION

Le principe de maintenance intégrée décrit ici, a été étudié de telle sorte que les dispositifs de maintenabilité représentent un pourcentage faible des matériels et logiciel des équipements. Au niveau système, le logiciel représente environ 25kmoats de programmes résidents pour un avion comme le MIRAGE 2000.

Les avantages de cette maintenance intégrée sont multiples :

- Grande indépendance par rapport aux évolutions des logiciels équipements
- Recherche de panne sans simulation des conditions de vol
- Mise en oeuvre sans utilisation de moyens de test extérieurs à l'avion sauf en ce qui concerne les capteurs et certains circuits d'armement
- Exploitation sur avion des résultats avec possibilité d'avoir une bonne connaissance de la nature et de la durée des pannes qui se sont produites en vol et qui ne sont pas toujours décelables au sol.
- Mise en oeuvre quasiment instantanée des logiciels permettant les recherches de pannes, ou les validations de chaînes fonctionnelles du Système d'Armes.
- Grande souplesse dans les procédures à utiliser pour effectuer un dépannage, le mécanicien utilisant au mieux et à sa seule initiative, les dispositifs développés dans le Système.

AN INTEGRATED AFCS FOR THE "PROFILE"-MODE

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1. Summary:

Existing Automatic Flight Control Systems for civil and military aircraft mostly use separate subsystems, such as the Flight Control Computer (FCC) and the Thrust Control Computer (TCC) with a very limited amount of communication between the "black boxes".

In most cases integrity reasons have dictated such a separation but in order to obtain optimal flight guidance results one should use an integrated design which for instance should reflect the coupled nature of such variables as airspeed and vertical speed.

Especially with the advent of the Flight Management Computer (FMC) with its flight profile management capability and accompanying new control modes such as the "PROFILE"-mode the integrated AFCS becomes a must. Nevertheless the appropriate control system design method should be able to incorporate proven structures.

In 1979 flight trials of such an integrated (digital) flight control system in a two-engined jet aircraft (HFB 320) demonstrated the advantages of such a concept /1/.

The very promising results were obtained by a joint effort of the DFVLR and German companies working in the aerospace field, supported by the Ministry of Research and Technology (BMFT).

For transferring these achievements to other aircraft that often have extended flight-envelopes a fast and transparent parameter design method (no control design *ex machina*) is required which puts the control engineer in the middle of the design process /2/.

This paper will describe the structure of the integrated control system and the appropriate parameter design method using the AIRBUS A300 as an example. The structure of the system is characterized by: (1)

- o a mixture of feedforward and feedback control thus allowing to separate the design of the command and disturbance characteristics, (2)
- o decoupling speed- and height loop.

The control system design relies on the knowledge of aircraft and engine characteristics, both known to the FMC or similar devices. To ease the analytic parameter design, the relevant (longitudinal) aircraft dynamic is simplified by the incorporation of well proven basic loops such as pitch attitude feedback and $N1/EPR$ -control.

Within the scope of this simplification, the design engineer prescribes eigenvalues of the entire system in a straight forward manner. Energy-saving aspects (in a higher frequency range) by the reduction of throttle lever excursions may also be considered using this approach.

List of symbols

EPR	:	engine pressure ratio
F	:	thrust
G	:	aircraft weight
H	:	vertical speed
m	:	aircraft mass
M	:	dimensional derivative, pitch moment equation
$N1$:	fan-speed
u	:	control input vector
V	:	airspeed
Vk	:	flight path velocity
Vw	:	wind speed
W	:	drag
Z	:	state space vector
Z	:	dimensional derivative, Z-force equation
z	:	elevator
z _{TR}	:	trim-tail-plane
α	:	pitch attitude
γ	:	flight path angle

1. Introduction

What is new with the PROFILE-MODE? This question is justified because existing autopilots and autothrottles do already compute pitch-attitude and thrust commands which let an aircraft fly along a certain trajectory. But as in CLIMB-MODE or DESCENT-MODE airspeed (Mach) and vertical speed (H) are not independent of each other, because these trajectories are flown with fixed thrust:

- maximum thrust during CLIMB
- minimum thrust during DESCENT.

In case were both pitch attitude and thrust are available as control variables, as in CRUISE-MODE or LAND-MODE, autopilot and autothrottle have to keep airspeed or Mach constant and the deviation from the reference altitude close to zero. These are typical hold-modes. Furthermore airspeed/Mach and vertical speed are not decoupled because this desirable characteristic can only be achieved, if the automatic flight control system (AFCS) reflects in some way or other the flight mechanical coupling of these variables. Conventional AFCSs however do not show cross-links between autopilot and autothrottle.

In our opinion the performance of the PROFILE-MODE will profit from an integrated AFCS and therefore this paper will concentrate on the description of an appropriate control system design-method.

2. The control system design method

2.1 Objectives

The objectives of the control system design method were as follows:

- o It should allow to design command- and disturbance characteristics separately
- o Flight path velocity and vertical speed should be decoupled or if any coupling were desirable it should be deliberately introduced.
- o The design method should rely on well proven basic loops such as pitch attitude feedback and N1-EPR control. (see Fig.1).

2.2 Simplification of the flight mechanical equations

The introduction of pitch attitude (and rate) feedback

$$\Delta \eta = K_{\theta} \cdot \{1 + T_Q \cdot s\} \cdot \Delta \theta \quad (1)$$

allows to neglect the acceleration term in the pitch-moment equation. Both the truncated pitch-moment equation and the pitch attitude basic loop can be incorporated easily in the remaining force equations.

The result is a completely controllable and observable second order system with vertical speed H and flight path velocity V_k as state variables, pitch attitude command and thrust as control variables.

$$\dot{\underline{x}} = A \cdot \underline{x} + B \cdot \underline{u} \quad (2)$$

$$\underline{x}^T = [\dot{H}, V_k] \quad (3)$$

$$\underline{u}^T = [\Delta \theta_c, \Delta F] \quad (4)$$

Fig. 2 compares the step responses of the complete 4-th order aircraft/basic-loop-system with that of the equivalent second order system using the AIRBUS A300 in a cruise condition as an example. The results need no further discussion. The only thing, which is not yet realistic is the fact, that thrust as an input variable is not available. However, this can be easily achieved using the well proven N1/EPR basic loop and the correlation between thrust and N1/EPR. This latter characteristic is well known to the FMC, which uses it to predict optimal guidance parameters.

A typical N1-control loop is given on Fig. 1.

Comparing the time constants of the elevator actuation system and the N1-basic loop with the eigenvalues of the equivalent system, they obviously can be neglected in the control parameter design process.

2.3 Design of the feedforward (FF) control

Having shown the good approximation by the second order system the structure of an ideal feedforward system can be sketched in a straightforward manner. Given the trajectories or model functions (see Fig. 3) for a change in altitude.

$$(\ddot{h}_c, \dot{h}_c, \ddot{h}_c) = F_{MH}(s) \quad (5)$$

and/or a change in airspeed

$$(V_c, \dot{V}_c) = F_{MV}(s) \quad (6)$$

one only needs a control law which gives

$$\begin{aligned} \ddot{h} &= \ddot{h}_c \\ \dot{V}_k &= \dot{V}_c \end{aligned} \quad (7)$$

The identities

$$\begin{aligned} \dot{h} &= \dot{h}_c \\ V_k &= V_c \end{aligned} \quad (8)$$

automatically hold.

The appropriate control law is derived from the equivalent state space representation:

$$\begin{aligned} \underline{u}_c &= B^{-1} \cdot \{ \dot{\underline{x}}_c - A \cdot \underline{x} \} \\ &= B^{-1} \cdot \{ \dot{\underline{x}}_c - A \cdot \underline{x}_c \} \end{aligned} \quad (9)$$

The block diagram of the ideal feedforward system is shown on Fig. 4.

An essential part of the design objectives has been achieved at this point:

In case of perfect modelling and absence of disturbances one gets the trajectories as they are described in the so called model functions through feedforward control alone. The eigenvalues of the system have not been touched.

But to cope with imperfect modelling and with ever existing disturbances (wind etc.) one additionally has to introduce feedback control.

2.4 Design of the feedback (FB) control

For the derivation of the appropriate feedback control one again uses the state space representation of the equivalent system which as a first step leads to the following equation:

$$\underline{u}_d = B^{-1} \cdot \{ \dot{\underline{x}}_d - A \cdot (\underline{x} - \underline{x}_c) \} \quad (10)$$

This control law, which decouples vertical speed and airspeed gives the ideal basis for the then following closures of altitude and airspeed loops (see Fig. 5).

$$\ddot{h}_d = k_1 \cdot (\ddot{h}_c - \ddot{h}) + k_2 \cdot \int (\ddot{h}_c - \ddot{h}) dt + k_3 \cdot (\dot{h}_c - \dot{h}) \quad (11)$$

$$\dot{V}_d = \int (k_4 \cdot (V_c - V) + k_5 \cdot (\dot{V}_c - \dot{V}_k)) dt \quad (12)$$

Due to the fact that vertical speed and airspeed are decoupled the eigenvalues of the individual loops can be chosen separately. Similar eigenvalues are recommended considering energy aspects.

2.5 Superposition of feedforward and feedback control

Both control schemes can be superposed

$$\underline{U} = \underline{U}_c + \underline{U}_d = \underline{U}(\text{FF}) + \underline{U}(\text{FB}) \quad (13)$$

Fig. 6 shows the complete integrated control system and Fig. 7 show the simulated time responses characterizing command- and disturbance behaviour of the aircraft (cruise condition) with its integrated AFCS. The simulation has been performed using the complete models. The reduced order state space representation has been used only for design purposes.

The resulting integrated AFCS shows no exotic features: it uses the same sensor and control inputs as existing conventional systems. For instance control inputs and basic loops can be made identical to the airbus systems if the trim-tail-plane is incorporated into the integrated AFCS by using simple block-diagram algebra (see Fig. 8).

3. Reduction of throttle activity

The crux of all flight modes which require a full-time autothrottle action to acquire and hold a computer optimized or pilot selected airspeed is the unavoidable measurement of unwanted higher frequency wind disturbances.

$$\Delta V = \Delta V_R - \Delta V_W \quad (14)$$

It is a dilemma: a FMC computes optimal trajectories in order to save fuel. To keep the aircraft flying according to the optimized Mach number one needs a fulltime autothrottle and that again may lead to increased throttle activity due to (see above) with all its unwanted side effects.

In order to alleviate the situation one can deliberately introduce again a defined coupling between vertical speed and flight path velocity. This is not a new idea but the previously shown design method gives nearly automatically the necessary structure as it shown in the following.

During cruise the X-force equation

$$m \cdot \dot{V}_R = F - W - G \cdot \delta \quad (15)$$

reduces to

$$\dot{V}_R = \frac{\Delta F}{m} - \frac{g}{V_R} \cdot \dot{h} \quad (16)$$

In case of strict altitude hold ($\dot{h}=0$) thrust activity is proportional to the horizontal acceleration.

$$\dot{V}_R = \frac{\Delta F}{m} \quad (17)$$

In case of a constant increase in horizontal wind speed ΔV_W in order to keep airspeed constant the following equation holds

$$\Delta V_R = \int \dot{V}_R dt = \int \frac{\Delta F}{m} dt = \Delta V_W \quad (18)$$

with $\dot{V}_R = \dot{V}_{R \text{ requ.}}$

In case of a ΔV -command input (no wind case) the following equation is derived in an analog manner

$$\Delta V_R = \int \dot{V}_R dt = \int \frac{\Delta F}{m} dt = \Delta V \quad (19)$$

with $\dot{V}_R = \dot{V}_{R \text{ requ.}}$

Throttle activity in terms of $\int \Delta F dt$ therefore cannot be influenced.

The situation changes if the altitude loop participates in speed acquire and hold actions.

$$\dot{V}_{k\text{requ.}} = \frac{\Delta F}{m} - \frac{g}{V_k} \cdot \dot{H}_{\text{requ.}} \quad (20)$$

$$\dot{H}_{\text{requ.}} = -\alpha \cdot \frac{V_k}{g} \cdot \dot{V}_{k\text{requ.}} \quad (21)$$

$$0 < \alpha < 1$$

In order to keep the altitude constant in a stationary sense the above control law has to be modified using a washout

$$\dot{H}_{\text{requ.}} = -\frac{T \cdot s}{1 + T \cdot s} \cdot \alpha \cdot \frac{V_k}{g} \cdot \dot{V}_{k\text{requ.}} \quad (22)$$

Introducing this control law which represents a defined coupling between the two loops the throttle activity reduces to

$$\begin{aligned} \frac{\Delta F}{m} &= \left(1 - \alpha \cdot \frac{T \cdot s}{1 + T \cdot s} \right) \cdot \dot{V}_{k\text{requ.}} \\ &= \frac{(1 + (1 - \alpha) \cdot T \cdot s)}{(1 + T \cdot s)} \cdot \dot{V}_{k\text{requ.}} \end{aligned} \quad (23)$$

3.1 Reduction of throttle activity for command inputs

Having an integrated system where vertical speed and flight path velocity are decoupled it is very easy to introduce the above coupling for command inputs.

$$\Delta \dot{H}_{\text{requ.}} = -\frac{T \cdot s}{1 + T \cdot s} \cdot \alpha \cdot \frac{V_k}{g} \cdot \dot{V}_{k\text{requ.}}$$

$$\Delta \dot{H}_c = \Delta \dot{H}_{\text{requ.}} = -\frac{T \cdot s}{1 + T \cdot s} \cdot \alpha \cdot \frac{\bar{V}_k}{g} \cdot \Delta V_c$$

$$\Delta \dot{H}_c = F_{mh}(s) \cdot \Delta \dot{H}_{sel}$$

$$\Delta V_c = F_{mv}(s) \cdot \Delta V_{sel}$$

$$\Delta \dot{H}_{sel} = -\frac{T \cdot s}{1 + T \cdot s} \cdot \alpha \cdot \frac{\bar{V}_k}{g} \cdot \frac{F_{mv}(s)}{F_{mh}(s)} \cdot \Delta V_{sel} \quad (24)$$

with $F_{mv}(s)$ and $F_{mh}(s)$ the command input model functions. Fig. 9 shows a mechanization of that command input coupling with $F_{mv}(s) = F_{mh}(s)$ whereas Fig. 10 compares the time responses with and without command input coupling.

3.2 Reduction of throttle activity for disturbance inputs

The introduction of the above coupling for the feedback portion of the integrated AFCS follows the same guidelines.

$$\Delta \dot{H}_{\text{requ.}}(\text{FB}) = -\frac{T \cdot s}{1 + T \cdot s} \cdot \alpha \cdot \frac{V_k}{g} \cdot \dot{V}_{k\text{requ.}}(\text{FB}) \quad (25)$$

$$\dot{V}_{k\text{requ.}}(\text{FB}) = \frac{\Delta F_c(\text{FB})}{m} \quad (26)$$

where $\Delta F_c(\text{FB})$ is that part of the thrust command which comprises all the relevant feedback signals. (see Fig. 6).

This signal is added as an additional command signal to the ($\dot{h}_c - \dot{h}$) feedback signal.

$$\Delta \dot{h}_{\text{requ.}}(\text{FB}) = -\frac{T \cdot s}{1 + T \cdot s} \cdot \kappa \cdot \frac{V_k}{g \cdot m} \Delta F_c(\text{FB}) \quad (27)$$

The additional command signal, which is necessary for (Hc-H) feedback is derived by pseudo integration.

$$\Delta h_{\text{requ.}}(\text{FB}) = \frac{T_I}{1 + T_I \cdot s} \cdot \Delta \dot{h}_{\text{requ.}}(\text{FB}) \quad (28)$$

A mechanization of this additional coupling is shown on Fig. 11.

The effect of that additional coupling is shown in Table 1 for flight through vertical gusts and horizontal gusts comparing standard deviations

	without (FB)-coupling	with (FB)-coupling
$\sigma_{\Delta v} [\text{ms}^{-1}]$	1.02	1.03
$\sigma_{\Delta h} [m]$	0.49	0.50
$\sigma_{\Delta F} [\text{Ns}^{-1}]$	2400 $\hat{=}$ 100%	1300 $\hat{=}$ 54%
$\sigma_{\ddot{h}} [\text{ms}^{-2}]$	0.62	0.62

Table 1

$$\sigma_{uw} = \sigma_{ww} = 1 \text{ ms}^{-1}$$

5. Conclusion

An aircraft which shall fly along a given trajectory or profile in order to fulfil certain requirements (to save fuel for instance) should have an AFCS with defined characteristics. This paper shows a design method for the vertical modes of an integrated AFCS which already has been flight tested successfully. The step by step design is simple and transparent through the use of a reduced order model of A/C and basic loop dynamics. It allows to separate the design of command and disturbance characteristics by a formal method. The (linear) design method gives the structural frame and control system parameters but it leaves ample space for the tricks control system designers usually have at hand to meet further real world demands.

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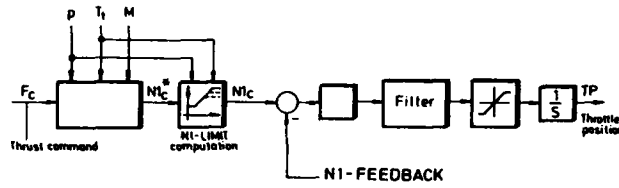
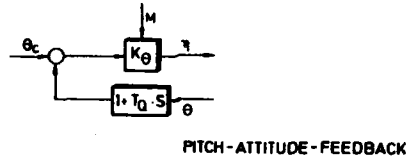


FIG. 1 BASIC LOOPS

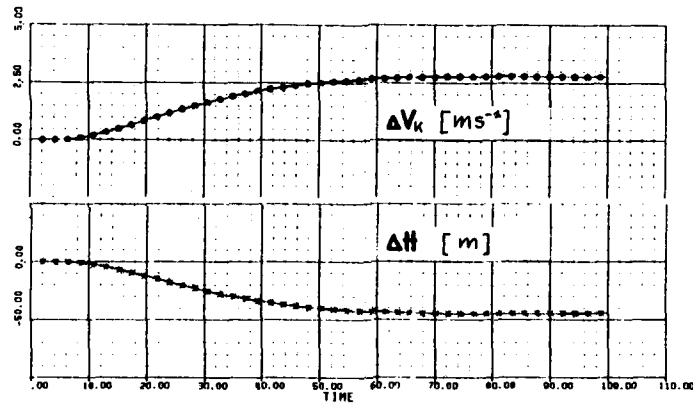


FIG. 2-1 COMPARISON OF MODELS $\Delta\theta_c = -0.5^\circ$

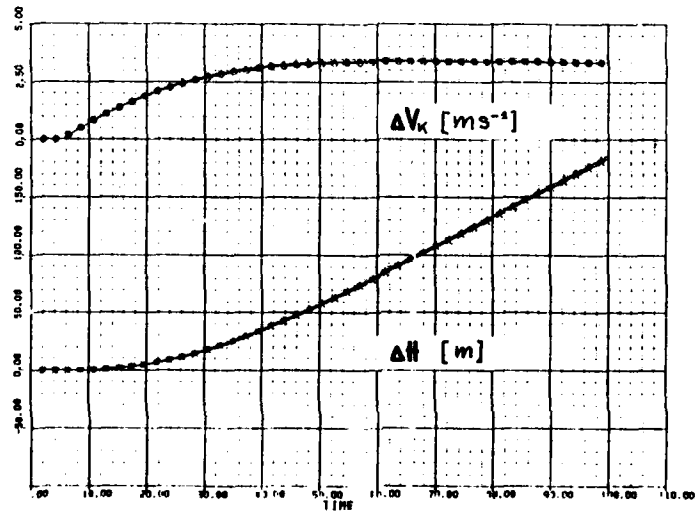


FIG. 2-2 COMPARISON OF MODELS $\Delta F = 20000 \text{ N}$

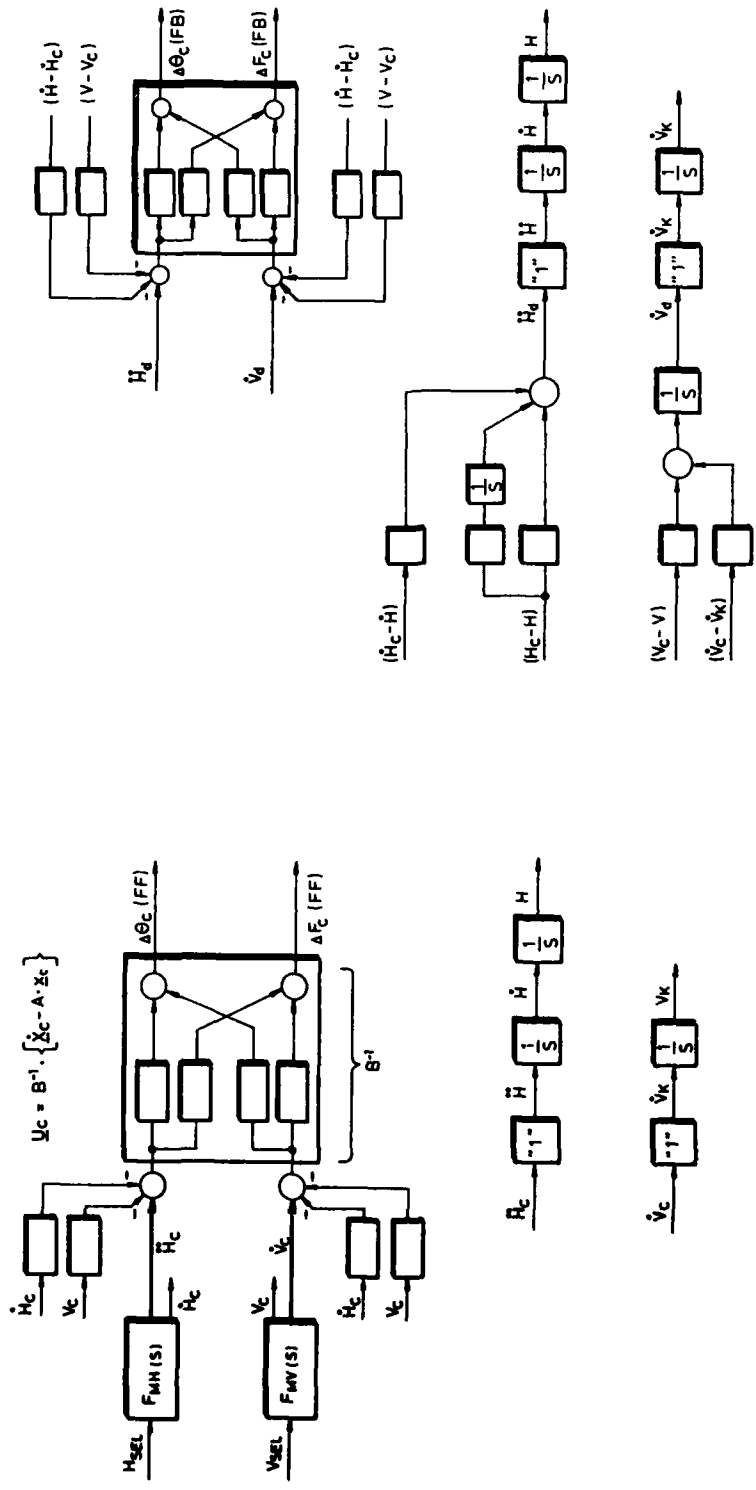


FIG. 4 FEEDFORWARD CONTROL

FIG. 5 FEEDBACK CONTROL

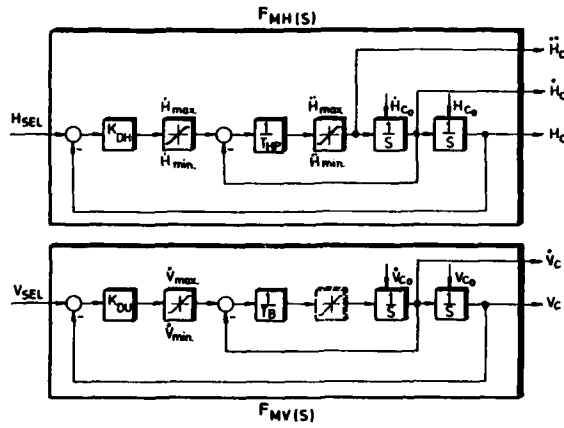


FIG. 3 COMMAND MODELS

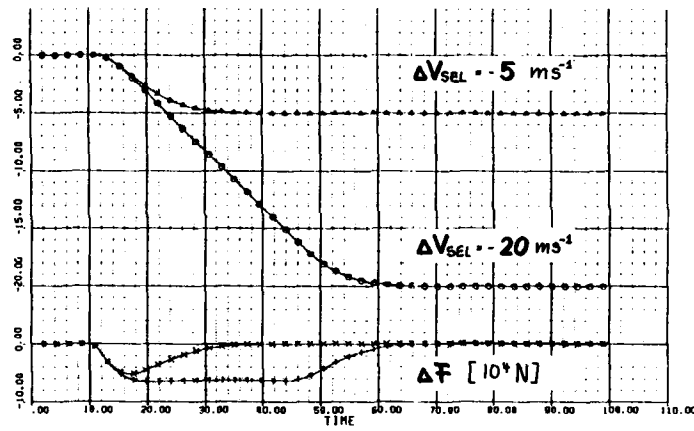


FIG. 7-1 TRANS. RESPONSE BASIC SYSTEM ΔV_{SEL}

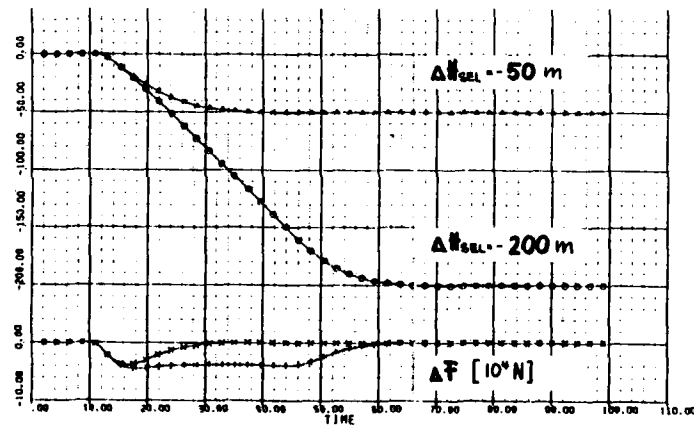


FIG. 7-2 TRANS. RESPONSE BASIC SYSTEM ΔH_{SEL}

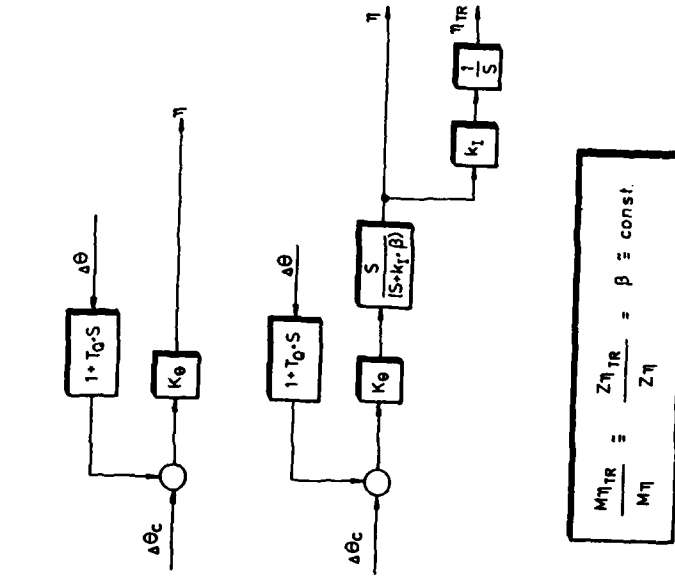


FIG. 8 INCORPORATION OF A TRIM TAIL PLANE

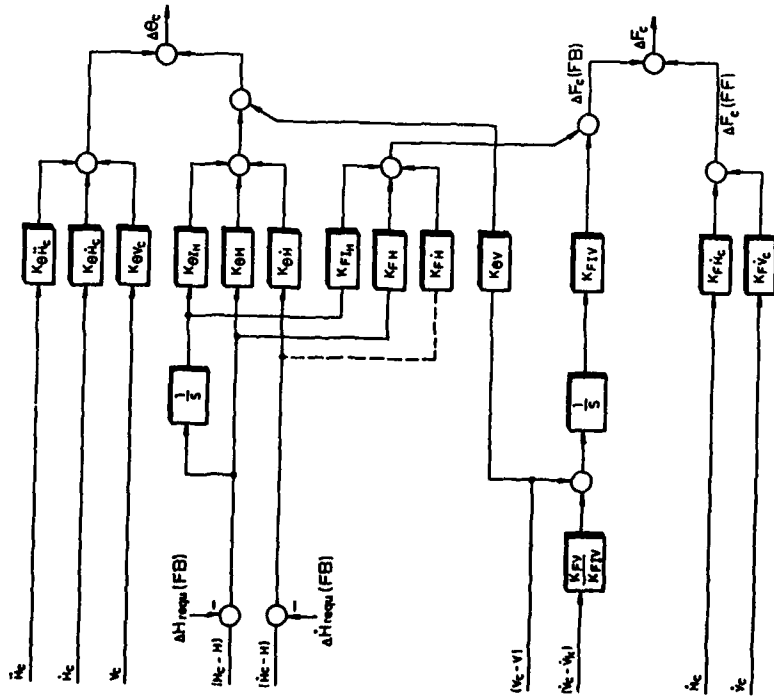


FIG. 6 BASIC INTEGRATED SYSTEM

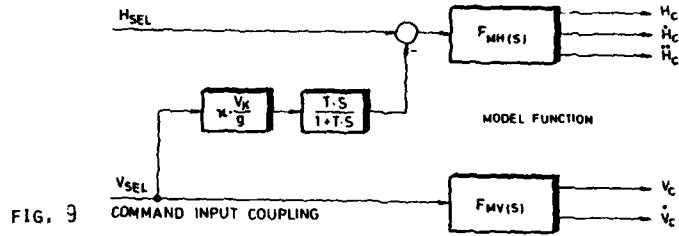


FIG. 9

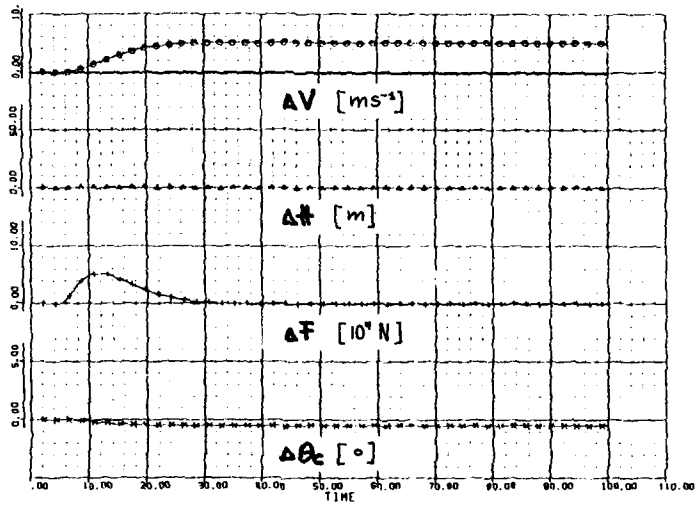


FIG. 10-1 ΔV_{SEL} -INPUT WITHOUT COMMAND-COUPLING

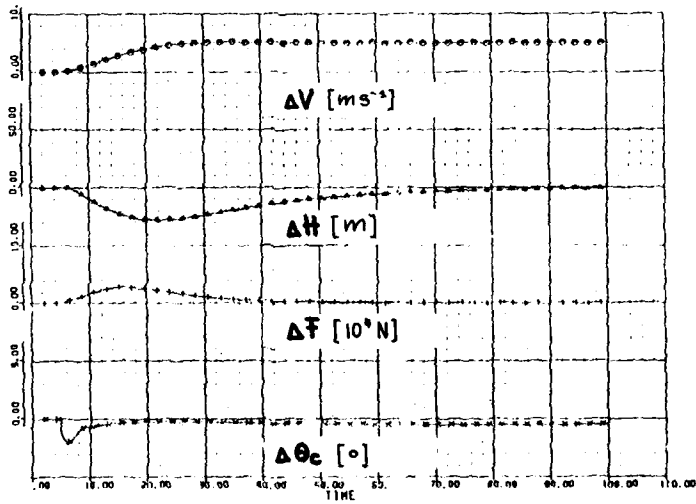


FIG. 10-2 ΔV_{SEL} -INPUT WITH COMMAND-COUPLING

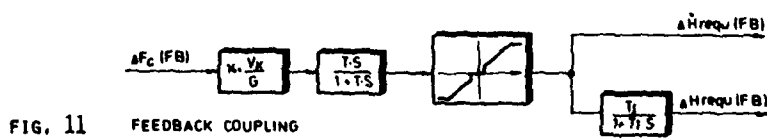


FIG. 11

FEEDBACK COUPLING

AD P002284

19-1

THE INTEGRATION OF FLIGHT AND ENGINE CONTROL FOR VSTOL AIRCRAFT

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Summary

A control philosophy appropriate to future jet VSTOL aircraft, ~~and which~~ leads naturally to the integration of flight and engine control systems ~~is discussed~~. The potential benefits of this class of system are stated and integration aspects, both in terms of control laws and at the hardware level, are considered. Some of the areas surrounding this approach to VSTOL control ~~which require deeper study~~ are outlined.

Introduction

In recent years, the capabilities of individual flight and mission critical control systems have advanced and the computing power available to them has increased rapidly. As the various systems have become more advanced it is becoming apparent that only through integration can their full potential be realised.

Much of the effort to date has been devoted to integrating mission critical systems with one another. With the advent of full authority digital electronic engine control, the combination of this system with others becomes a more practical proposition.

While most variations on the integration theme offer similar benefits both to "conventional" and to VSTOL aircraft, the integration of flight and engine control systems is especially appropriate for VSTOL configurations.

This paper will show how engine and flight control integration may be seen as particularly desirable for VSTOL aircraft if the possibilities of this type of vehicle are fully to be exploited. It will present the particular benefits anticipated and a means of realising them, recognising that there are various levels of integration.

1. Handling Aspects

The current Harrier family of aircraft is controlled in a manner that may be described as conventional with the exception of the additional nozzle angle lever. With his right hand, the pilot controls aircraft pitch and roll, with his feet he exercises directional control, and with his left hand he controls the two degrees of freedom of the engine, namely thrust and additionally thrust direction. Thus it may be said that with his right hand the pilot controls airframe parameters and with his left, engine parameters. Any integration of these two sides of the control coin that is necessary in order to perform a given manoeuvre must be done by the pilot. He must choose between a choice of three longitudinal controls (pitch stick, throttle control and nozzle angle control) for which he has only two hands.

Although in theory the possibility of longitudinal control ambiguity exists, this does not appear to be a problem with the Harrier and its derivatives. It is indeed a credit to the design of the aircraft that this method of control works so well. Pilots regard the aircraft as offering outstanding versatility in both airfield (or ship) operations and in combat flying.

It could be argued that this versatility is only obtained at the expense of higher pilot workload, particularly in the transition between jet-borne and wing-borne flight. While some studies would indicate that this is the case (ref 1), recent combat experience involving shipborne operations in poor weather show that the workload involved is by no means beyond the capabilities of well-trained service pilots.

In spite of this success, it would be unwise to assume that a future jet VSTOL aircraft will inevitably be blessed with such good handling characteristics. Perhaps more importantly, it is attractive to free the airframe designer from some of the constraints imposed by the need to provide inherent good handling, allowing him to explore more fully the potential of this class of aircraft. Furthermore, electronic control systems themselves offer new freedoms which the designer may wish to exploit.

One avenue of research that is being explored is the development of an integrated manoeuvre demand system. As mentioned above, the pilot currently integrates airframe and engine control himself in order to attain the desired manoeuvre. It would seem more efficient for the pilot simply to be able to demand a manoeuvre, and for the flight and engine control systems to determine the optimum way of attaining the manoeuvre. A manoeuvre demand system which in addition requires inputs through only two cockpit inceptors would obviate the possibility of control ambiguity.

There are a variety of handling benefits that add to the attractiveness of an integrated manoeuvre demand system. The transition between jet-borne and wing-borne flight is the regime most likely to raise handling problems in a VSTOL aircraft. Manoeuvre demand, with appropriate limits (e.g. α limits), can facilitate transition flying and reduce pilot workload. Furthermore, a manoeuvre demand system would lend itself readily to integration with landing aids such as microwave guidance systems to produce a fully automated approach and, if desired, landing system.

It is essential, in fully and partially jet-borne flight, to ensure that large longitudinal demands do not result in vertical transients or a serious loss of jet lift. Thus, height demands must take precedence over longitudinal demands in terminal flight phases, but limited longitudinal transients will be permitted if the pilot inputs a large normal demand.

In combat flying, the ability to vector thrust has been shown (ref 2) to give tactical advantages. An integrated manoeuvre demand system can produce thrust vectoring in a combat situation in a well-behaved manner. If the pilot demands more normal acceleration than is available from wing lift (as defined by reaching an α limit), then a certain amount of extra acceleration can be generated from thrust vectoring. In order to avoid excessive energy loss associated with large thrust vector angles, it is suggested that longitudinal demands take precedence over normal demands in up-and-away flight. Thus rapid deceleration due to thrust vectoring would only result from a large deceleration demand, and not from a large normal demand alone.

2. Implementation

2.1 Conceptual.

There are many combinations of control possible, of which the following description is just one example. Roll rate demand is proposed for the lateral axis and sideslip demand (perhaps with a lateral acceleration feedback) blending to a pure yawing demand in the hover for the directional axis.

The solution to the challenge posed by the symmetric axes is less conventional. The approach adopted here is that the right hand inceptor should demand normal (z-axis) manoeuvring and that the left hand inceptor should demand longitudinal (x-axis) manoeuvring. "Normal manoeuvring" would consist of normal acceleration, perhaps blending to normal velocity at the hover, and "longitudinal manoeuvring" would be longitudinal acceleration, again blending to a longitudinal velocity at the hover. There is a requirement for independent pitching of the airframe in the hover (to allow the pilot to inspect a landing site before a vertical landing). If the control power is available, independent pitching in the form of fuselage pointing is a potentially beneficial feature in combat flying. This mode, in the form of incremental pitch attitude demand about an appropriate datum (e.g. landing attitude in the hover) can be demanded through a rotary input on the left hand controller. An appropriate inceptor design will be described later.

2.2 Control Scheme

The key feature of the concept presented above is that the pilot no longer has direct control over any one control surface, or over the engine. The flight control system (FCS) produces motivator demands in order to generate the required manoeuvre. Engine thrust and thrust vector angle are, as far as the control system is concerned, an extra pair of motivators. Fig 1 illustrates this; note that there is no "throttle", but two inceptors, engine demands coming from the FCS. Also, there is no reason why the engine nozzles should not be controlled independently of each other to give, say, enhanced pitch control or a greater c.g. range in the hover.

A variety of control schemes of varying degrees of complexity may be imagined to satisfy the conceptual requirements of integrated manoeuvre demand. The symmetric axes of one particular system of control laws are illustrated in Figs 2 and 3. They fall into two separate parts; the manoeuvre demand laws themselves, that resolve pilot inputs into motivator (tailplane) and pseudo-motivator (normal and forward thrust) demands, and the pseudo motivator control law that generates thrust magnitude and nozzle angle (or mean nozzle angle) demands.

The longitudinal axis of the manoeuvre demand system basically uses longitudinal acceleration feedback to generate a forward thrust demand. The normal axis is built around normal acceleration demand with incidence limiting and a term for pitch damping. In addition, the normal acceleration error signal is fed forwards through a non-linear function of speed and incidence to provide a normal thrust demand when aerodynamic lift is exhausted.

The two thrust demands are fed into the pseudo motivator control law. Since in a vectored thrust engine a change in nozzle angle can be accomplished more rapidly than a change in thrust, simply to resolve the two cartesian demands into vector and magnitude demands would result in the sort of transient response of Fig 4. The control law of Fig 3, by using an estimate of the current thrust provided by the Engine Control System in the calculation of nozzle angle demand, synchronizes the movement of the nozzles with the actual engine response. Thus the transients of Fig 4 will be substantially reduced.

This pseudo motivator scheme is also designed to give priority to normal thrust if the demanded thrust magnitude is greater than that available (e.g. when hovering) and to forward thrust when the minimum thrust boundary is encountered (e.g. for ground handling reasons).

2.3 Physical Integration

For the purpose of this paper, it is intended now to consider some of the ramifications of applying control laws such as that illustrated to a VSTOL aircraft.

The close functional integration implicit in this system obviously dictates that there be some degree of physical integration of the FCS and the engine control system (ECS). It is assumed that the ECS will be some form of digital electronic engine control (DEEC), both for reasons of ease of communication between systems, and for the consistency of engine response that DEEC is expected to confer.

The first question to be addressed is that of physical location. While it may at first appear desirable to co-locate ECS and FCS computing, this raises problems. If the combined computing is airframe mounted, it may not be possible for it to be in the close proximity of the engine, leading either to a multiplicity of wires between the control system and engine (for both control of the fuel system and signalling to the ECS of engine parameters) or to the adoption of digital signalling to and from engine mounted multiplexer and A-D/D-A converters. Collocation of the combined computing on the engine, while tidier, is again likely to run into space problems, not to mention the unattractiveness of placing all the flight critical computing in such a hostile environment.

As engine mounted DEEC has already been demonstrated on both R-R Pegasus (ref 3) and P&W F-100 engines (ref 4), it is suggested that engine demands (either of thrust or of a combination of engine parameters that will approximate to thrust) be generated by an airframe-mounted flight control system, while "inner loop" engine control is performed by an engine mounted DEEC unit.

Communications between FCS and ECS must be of high integrity. A digital databus of the 1553B type is designed neither to provide the necessary level of integrity, nor the speed of communication for safe and satisfactory integrated control. Consequently, a dedicated high integrity data link between FCS and ECS is needed.

Any air data required by the ECS for optimisation of engine performance would be provided from the FCS's dedicated air data sensors and computation, via the FCS. This will not introduce unsatisfactory delays in getting the data to the ECS if an asynchronous multiprocessor architecture (ref 5) is used, rather than the current monolithic architecture; the asynchronous multiprocessor will have a cycle time of the order of 1-2 ms, while current processors typically operate with a cycle time of the order of 20 ms.

Integration is not limited to the FCS and ECS. As the control philosophy described was borne of a desire to rethink the way in which the pilot may control his aircraft, attention must be given to the cockpit, and in particular the two hand inceptors.

It is envisaged that a miniature side stick will allow right hand inputs, namely normal and lateral manoeuvre demand. As mentioned earlier, the left hand will be responsible for both fore-and-aft and incremental pitch attitude inputs. An inceptor designed around this requirement is illustrated in Fig 5. Quite simply, fore-and-aft movement of the inceptor will generate longitudinal demands, while twisting of the end segment of the controller will generate the pitching demand. Inputs are in exactly the same sense as the demanded response. Both axes of inceptor movement will be self-centring. The tactile feedback provided by inceptor movement is valuable, and a pure force/moment controller is considered inappropriate.

3. The Technical Challenges

No matter how attractive the benefits of a novel system, there are invariably a variety of potential problem areas; the extent to which the full benefits of a manoeuvre demand system may be realised will depend at least in part on the overcoming of these challenges in a manner that is cost-effective. It is appropriate, therefore, to identify areas that may require particular attention and to consider how best to address them.

The exact nature of the manoeuvres to be demanded by the two inceptors will have to be considered most carefully, and in particular any blends between modes. The need for harmony between the various axes will need to be borne in mind, particularly in the hover. It is possible that the lateral control will need to be attitude demand rather than roll rate demand in the hover; this will give approximately sideways velocity demand, to be consistent with the fore-and-aft longitudinal velocity demand.

The question of pilot acceptability of the unconventional hover control, where right hand inputs will produce an up-and-down (height) response (rather than left hand inputs as in more traditional VSTOL control), must be addressed. It would be desirable to obtain the opinions (from simulation) of both VSTOL and "conventional" pilots, since ultimately such a system as this would be flown by pilots without previous VSTOL experience.

There are ground handling implications of any fully integrated control system where the pilot does not have direct control over the engine; the ability to move the nozzles so as to reduce the risk of foreign object damage or hot gas reingestion while sitting on the ground; the engagement of the reaction control system to aid manoeuvring on a slippery surface; checking engine acceleration before take-off; coping with high residual thrust while taxiing. These are examples of the less obvious problems that may be anticipated, yet which must be resolved for an operational system.

Operational techniques and careful design of the control laws may answer some of these particular questions, for example giving priority to longitudinal thrust demands when on the ground. Such specifics as engine checks can be enabled by means of cockpit switches, but in general switches associated with flight critical systems are to be avoided. Not only are they a possible source of unreliability, but one has to guard against the risk of inadvertent operation in the wrong part of the mission or the flight envelope.

Achieving a satisfactory level of integrity from the combined engine and flight control system will require collaboration between the airframe, engine and systems manufacturers. Different levels of redundancy and different failure management philosophies between the FCS and the ECS will not ease the problem of communications between these systems. Bi-directional fibre optic links could be applied here to good effect. A possible overall system architecture is illustrated in Fig 6. A triplex flight control system receives inputs from the pilot and from air data and motion sensors (plus any other information required by the system, like control surface positions). The FCS communicates via bi-directional links with dual redundant engine controllers and receiving from them a thrust estimate. Consolidation is performed at the downstream end of data flows. The flight control computers (FCC's) also transmit demand signals to control surface and nozzle actuators. FCS inter-lane communications are shown. Each FCC and engine controller will also be attached via an optically isolated interface to remote terminals on a digital databus, probably dual redundant and to MIL STD-1553B. (For clarity, this has been omitted from Fig 6). The databus interface will permit communication between the flight critical systems (engine and flight control) and non-flight critical systems such as navigation, weapon aiming and maintenance/diagnostic systems. This would be the channel for integration of flight and mission critical systems.

It is intended to study the integration of flight critical systems (with themselves and with mission critical systems) on an Avionic Systems Demonstration Rig and an ACT rig at B.Ae. Brough in the near future. Along with pilot-in-the-loop simulation, solutions to the above problems will be investigated and demonstrated.

The "bottom line" of these deliberations is that any integrated manoeuvre demand system must at least equal a more conventional three inceptor control system in terms of versatility and pilot workload, yet this must not be at such a price in terms of cost and weight that the customer finds the solution unacceptable.

Concluding Remarks

Greater freedom to exploit the unconventional aspects of jet VSTOL configurations can be conferred upon the airframe designer by the adoption of a fully integrated manoeuvre demand system. This type of control system also offers piloting benefits, especially in the transition.

The satisfactory implementation of a manoeuvre demand system on a VSTOL aircraft, dictating as it does close functional and physical integration of flight and engine control systems, requires coordinated study in a variety of fields in order to achieve a cost-effective design.

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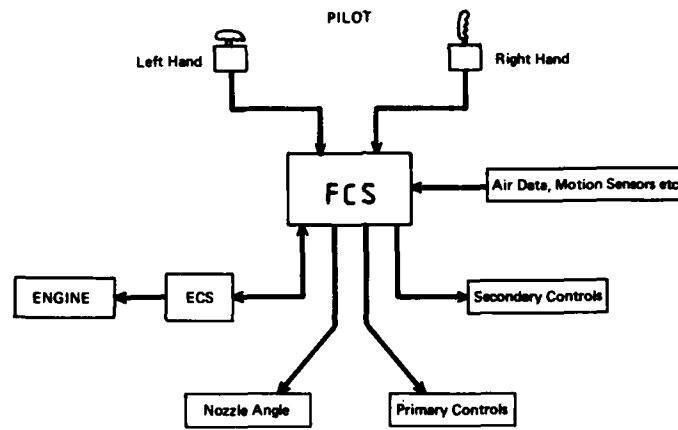


Fig 1. VSTOL Manoeuvre Demand System : Basic Architecture.

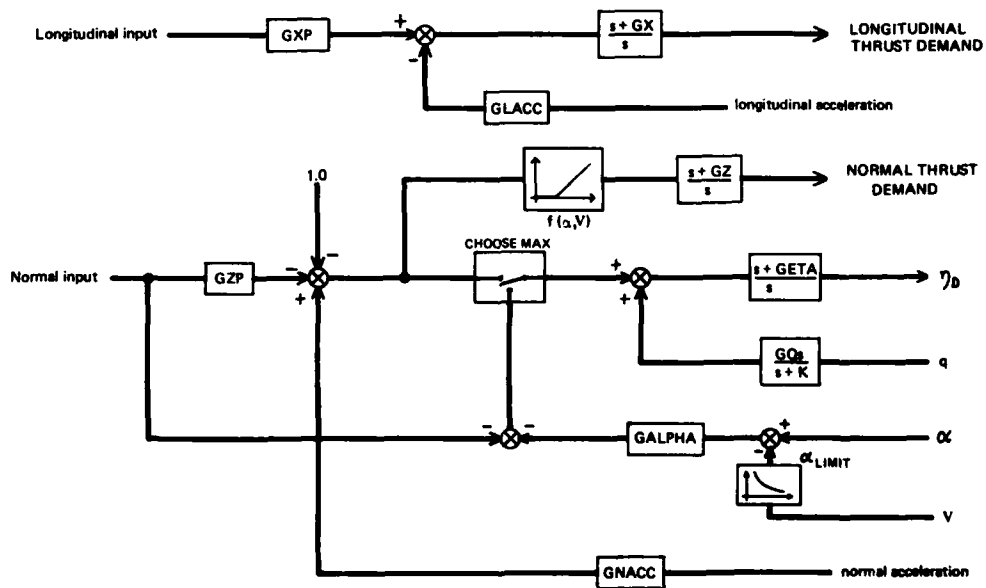


Fig 2. Simplified Manoeuvre Demand Control Laws.

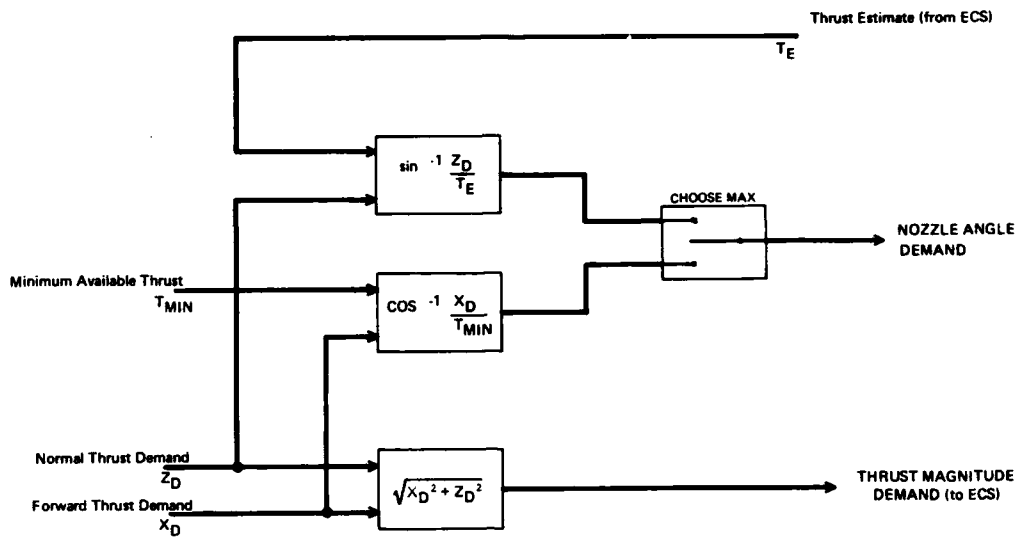


Fig 3. Simplified Pseudo - Motivator Control Law.

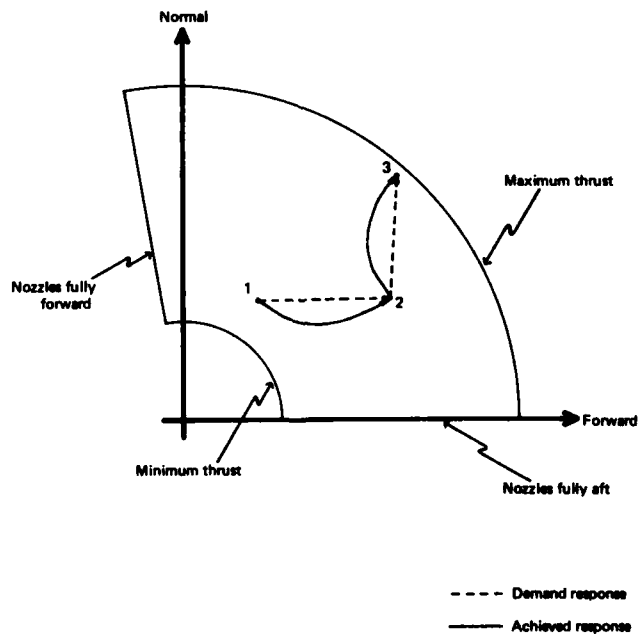


Fig 4. Available Thrust Polar Envelope. Simple Resolution of Cartesian Thrust Demands.

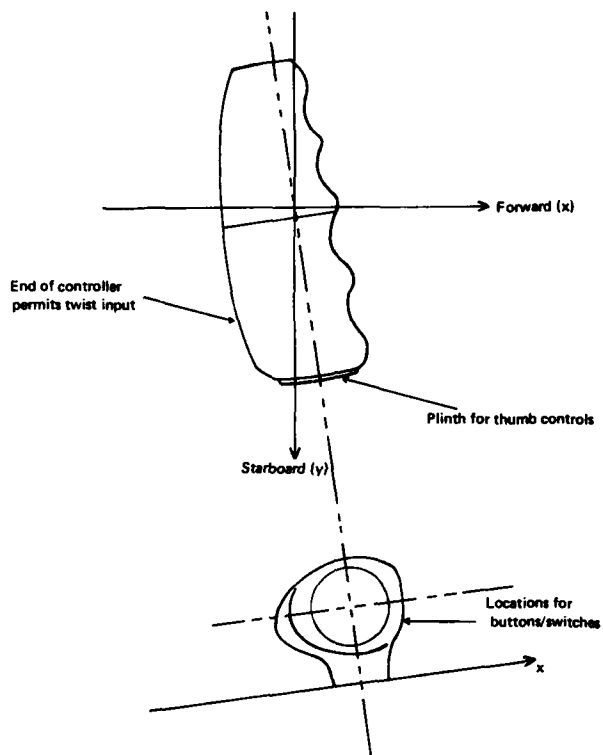


Fig 5. Two-view Sketch of Proposed Left Hand Controller.

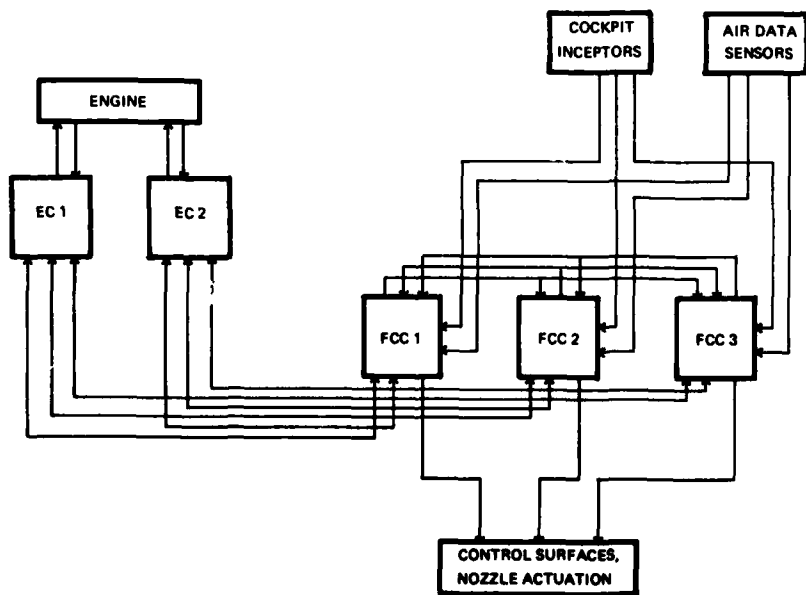
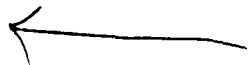


Fig 6. Outline FCS and ECS Integration Architecture.



**SUR UNE LOI DE COMMANDE NON-LINÉAIRE
POUR LE PILOTAGE DES AVIONS EN PHASE D'ATTAQUE AIR-SOL**

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RESUME

Dans le concept de la Commande Automatique Généralisée des avions, cette communication présente une loi de pilotage manuelle, non-conventionnelle, destinée à améliorer l'efficacité de la visée en tir au canon air-sol. La mise en oeuvre suivant une architecture de système intégrant une conduite de tir air-sol, un régulateur de tir et un régulateur de pilotage est détaillée sous forme analytique. Par rapport aux systèmes de commandes classiques, l'innovation porte principalement sur l'aspect élaboré du traitement multivariable des ordres du pilote. L'application de méthodes modernes de calcul de commande non-linéaire rend par ailleurs la loi bien adaptée pour les manoeuvres d'alignement de grande amplitude de l'avion. La pilotabilité d'une version linéaire de la loi a été démontrée au cours d'essais sur simulateur.

ABSTRACT

In the Control Configured Vehicle concept, this paper presents a manual and non-conventional aircraft control law which provides an improved target tracking in air-to-ground gunnery. The system architecture which integrates a weapon aiming system, a fire control system and a flight control system is analytically detailed. In comparison to classical flight control systems, improvements come mainly from the elaborate processing of the pilot steering commands. The application of modern non-linear control techniques to the design of the control law makes it otherwise well-suited for aircraft alignment manoeuvres of large magnitude. The controllability of a linear version of the control law has been demonstrated on a manned simulator.

NOTATIONS

u, v, w	composantes du vecteur vitesse aérodynamique dans le trièdre avion
p, q, r	composantes du vecteur rotation instantanée dans le trièdre avion (roulis, tangage, lacet)
ϕ, θ, ψ	angles d'Euler (angle de gîte, assiette longitudinale, azimut)
$\delta_1, \delta_m, \delta_n$	braquages des gouvernes de gauchissement, profondeur, direction
e_x, e_y, e_z	angles de présentation de la cible
h_x, h_y, h_z	corrections de tir canon (hausse)
$\epsilon_x, \epsilon_y, \epsilon_z$	erreurs angulaires de visée
$p_a = (\dot{p}u + qv + rw)/V$	vitesse de roulis aérodynamique
\dot{p}	vecteur vitesse aérodynamique de l'avion
\dot{v}	vecteur vitesse de la cible
u_T, v_T, w_T	composantes du vecteur vitesse de la cible dans le trièdre avion
Y, Y', \dots	coefficients aérodynamiques dimensionnels
L, L', p_L, p_L', \dots	termes de couplage gyroscopique

1. INTRODUCTION

Pour assurer le pilotage général d'un avion, il est tout à fait envisageable de commander des variables directement associées à son mouvement. Ce mode de pilotage "par objectifs" (PO) simplifie l'exécution de certaines manoeuvres exigeant une action coordonnée sur les gouvernes, tâche autrement difficile si elle est confiée au pilote. Des études de PO faites à l'ONERA ont permis ainsi d'établir, suivant diverses méthodes, des lois de braquage de gouvernes appropriées à la commande des variables d'état du mouvement (dérapage, vitesse de roulis autour de l'axe avion ou autour du vecteur vitesse, vitesse de tangage, vitesse de lacet [1-4]). Les essais sur simulateur avec pilote humain dans la boucle ont mis en évidence la pilotabilité de telles lois ainsi que les qualités de vol améliorées qu'elles confèrent à l'avion. La loi citée en référence [2] a notamment été testée avec succès dans le tir au canon air-sol.

Le tir au canon air-sol est une phase de vol très critique où un alignement rapide de l'avion sur la cible et une visée précise, utilisant une hausse mobile pour étendre le domaine de tir, requièrent d'habitude une habileté et une finesse de pilotage particulièrement importantes de la part du pilote. La mise à la disposition du pilote de variables de commande appropriées peut alors contribuer à alléger la charge de travail et améliorer l'efficacité de la visée. La loi de pilotage proposée dans cette communication est basée sur ce concept de PO. Elle est spécifique au tir air-sol car des objectifs encore plus élaborés que ceux cités plus haut sont mis en oeuvre.

La mise en oeuvre de la loi de commande suivant une architecture de système intégrant une conduite de tir canon air-sol, un régulateur de tir et un régulateur de pilotage est détaillée sous

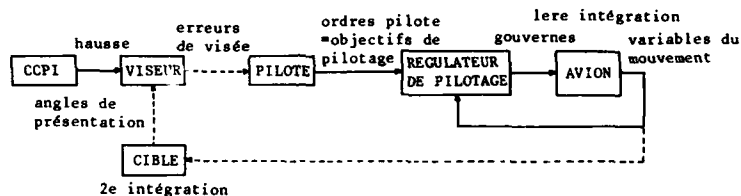
forme analytique. La formulation mathématique de la loi de commande fait appel à une méthode de commande non-linéaire non-interactive par laquelle les sorties d'un système non-linéaire sont contrôlées de manière indépendante.

Une simulation numérique de passe de tir air-sol illustre le fonctionnement du système de commande. Quelques résultats obtenus sur un simulateur d'avion de combat et portant sur une version linéaire de la loi de commande sont également présentés.

2. COORDINATION DES BOUCLES DE PILOTAGE

Dans une passe de tir air-sol, le fonctionnement dans le temps du système composé par la cible au sol, l'avion et le pilote est caractérisé par deux niveaux d'intégration. A partir d'un ordre de braquage des gouvernes du pilote, auquel s'ajoutent éventuellement des ordres de stabilisation, le mouvement de l'avion s'obtient par une première intégration des équations dynamiques. L'intégration des équations cinématiques donne ensuite la trajectoire et il est évident que, mis à part l'effet du rapprochement, une modification de la trajectoire entraîne une modification des positions relatives de la cible et de l'avion. Il est donc vraisemblable que l'alignement de l'avion sur la cible sera d'autant plus facile que la correction désirée concerne les variables se trouvant plus en aval dans la chaîne d'intégration.

1) Ainsi, le contrôle direct des variables dynamiques (cf. [1-4]), en donnant à l'avion des réponses pures et découplées, contribue à réduire la charge de travail du pilote dans la tâche de visée. La boucle de pilotage-visée obtenue avec un tel système est schématisée sur la Fig. 1, dans le cas où le PO est appliqué aux vitesses angulaires de roulis aérodynamique, de tangage et de lacet. Les trois commandes classiques du pilote (manche en latéral, manche en longitudinal, palonnier) affichent des consignes de vitesses angulaires à réaliser ; le régulateur de pilotage élabore les ordres de braquage des gouvernes qui stabilisent le mouvement de l'avion et assurent la réalisation des objectifs affichés. La vitesse longitudinale de l'avion est supposée être commandée indépendamment au moyen de la poussée-moteur, non représentée sur la figure.



PO = consignes de vitesse de roulis aérodynamique, de vitesse de tangage et de vitesse de lacet à réaliser.

CCPI = calcul continu du point d'impact (conduite de tir air-sol)

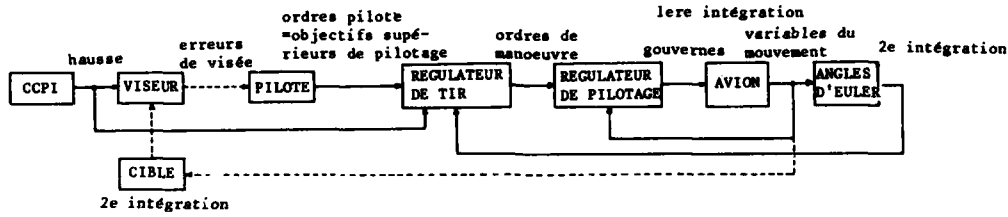
Fig. 1 Boucle de pilotage-visée avec un niveau d'intégration.

2) Le contrôle direct des variables cinématiques se heurte au fait que le pilotage des avions s'effectue normalement en vitesse et non en position. Par ailleurs, l'automatisation totale rendrait nécessaire la mesure des variables cinématiques (ici la distance avion-cible, les erreurs de poursuite, etc...) et entraînerait par conséquent une complication de la chaîne des capteurs.

3) La solution adoptée ci-après se place entre les deux cas décrits plus haut, en ce sens qu'elle réalise une commande manuelle, en vitesse, préservant donc des conditions classiques de pilotage, facilitant cependant la visée grâce à l'existence d'une relation directe entre la réponse des erreurs de visée et l'ordre du pilote. Elle consiste à établir une loi de variation des vitesses angulaires de l'avion (objectifs de pilotage du cas(1)) qui annule les erreurs de visée suivant une dynamique spécifiée. Dans ce but, la boucle est "fermée" par le pilote qui estime les erreurs de visée, ces-ci n'étant pas mesurées à bord. Pour calculer les ordres de braquage des gouvernes qui réalisent la loi de variation des vitesses angulaires précédentes, il suffit d'utiliser le régulateur de pilotage de la Fig. 1. La boucle de pilotage-visée obtenue avec le système proposé est représentée sur la Fig. 2.

La boucle interne de stabilisation et de pilotage est identique à celle représentée sur la Fig. 1, l'entrée du régulateur de pilotage étant toujours des consignes de vitesse de roulis aérodynamique, de vitesse de tangage et de vitesse de lacet à réaliser. Ces consignes issues du régulateur de tir sont calculées de sorte que si l'ordre du pilote (affichage de deux quantités homogènes à des angles) est identique à chaque instant à l'erreur de visée, cette dernière tend vers zéro. Le pilote joue donc en quelque sorte le rôle d'un estimateur d'état. La pilotabilité d'un tel système (problème de pompage piloté, par exemple) n'est pas assurée a priori et devra être vérifiée sur un simulateur avec pilote humain dans la boucle. Quelques considérations d'ordre analytique sur la pilotabilité sont exposées au chapitre suivant, montrant en particulier que la ligne de visée est contrôlée en vitesse au travers des ordres du pilote.

La commande de roulis mise à la disposition du pilote est présentée à l'entrée du régulateur de tir sous forme d'un angle de gîte à rallier, ceci pour rester homogène avec les erreurs de visée



PO = affichage de deux quantités homogènes à des angles
consigne d'angle de gîte à réaliser

Ordres de manoeuvre = consignes de vitesse de roulis aérodynamique, de vitesse de tangage et de vitesse de lacet à réaliser (cf. Fig. 1)

Fig. 2 Boucle de pilotage-visée avec deux niveaux d'intégration.

qui sont des variables cinématiques. Afin de respecter la contrainte de pilotage en vitesse, l'affichage de l'angle de gîte désiré s'obtiendra par intégration des déplacements latéraux du manche ou efforts latéraux sur le manche.

Un avantage de l'architecture de système présentée sur la Fig. 2 est que la stabilisation du mouvement de l'avion est assurée par une boucle indépendante. Le pilotage général de l'avion semble à celui défini sur la Fig. 1 s'obtient par simple commutation des ordres du pilote sur l'entrée du régulateur de pilotage ; l'avion ne se trouve donc jamais en boucle ouverte, même au moment de la commutation.

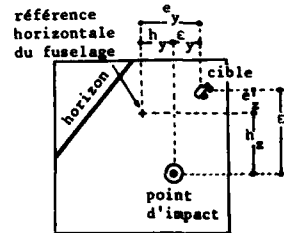
Dans le chapitre suivant, les lois de régulation du régulateur de tir et du régulateur de pilotage sont formulées suivant une méthode de calcul de commande non-linéaire, prenant explicitement en compte les non-linéarités des équations cinématiques d'une part, des équations dynamiques d'autre part. Grâce aux lois non-linéaires, les manoeuvres de grande amplitude de l'avion, notamment en roulis pour rallier une cible latéralement très écartée, peuvent être ainsi réalisées avec une grande précision.

3. FORMULATION MATHÉMATIQUE DES LOIS DE RÉGULATION

Le problème consiste à établir une loi de braquage des gouvernes $u = (\delta_1, \delta_m, \delta_n)^T$ qui permet de ramener, sous un angle de gîte quelconque ϕ_d affiché par le pilote et suivant une dynamique spécifiée, un point du viseur vers un autre point du viseur. Le premier point est la cible liée au sol et dont les angles de présentation dans le viseur sont $z = (e_y, e_z)^T$; le deuxième est le point d'impact des obus au sol calculé par la conduite de tir r_a à bord de l'avion et dont la position dans le viseur est donnée par les corrections de tir ou hausse $z_{rd} = (h_y, h_z)^T$ (cf. Fig.3).

e_y, e_z = erreurs angulaires de visée

Fig. 3 Angles de présentation de la cible dans le viseur et corrections de tir canon air-sol (hausse).



Comme il est indiqué plus haut, une loi de variation des vitesses angulaires $\dot{z} = (p_a, q, r)^T$ de l'avion est d'abord calculée pour annuler les erreurs de visée $\dot{z} = \dot{z} - \dot{z}_{rd} = (\dot{e}_y, \dot{e}_z)^T$ (loi de régulation du régulateur de tir). Une loi de braquage des gouvernes est ensuite calculée pour réaliser les ordres de manoeuvre précédents ainsi que pour stabiliser le mouvement de l'avion (loi de régulation du régulateur de pilotage).

Equations cinématiques et dynamiques

Le vecteur contrôlé d'ordre supérieur du régulateur de tir $z_s = (z_r^T, \phi)^T$ vérifie l'équation cinématique non-linéaire suivante :

$$\dot{z}_s = C_s(z_s, \bar{V}_T - \bar{V}, X) + B_s(z_s, z_a)\Omega \quad (1)$$

avec $z_a = (\phi, \theta)^T$ et $\Omega = (p, q, r)^T$. X est la distance avion-cible projetée sur l'axe longitudinal de l'avion. Pour le tir air-sol, il est commode de négliger la vitesse de la cible devant celle de l'avion, c'est-à-dire $\bar{V} - \bar{V}_T \approx \bar{V}$.

Les équations du mouvement de l'avion s'expriment par les relations différentielles non-linéaires suivantes :

$$\dot{x} = A(x) + Bu \quad (2)$$

avec $x = (x_v^T, \Omega^T, x_a^T)^T$ et $x_v = (v, w)^T$. La vitesse longitudinale u est considérée comme un paramètre du système.

Le vecteur contrôlé du régulateur de pilotage $z = (p, q, r)^T = D(\Omega, x_v)$ vérifie en particulier l'équation dynamique non-linéaire suivante :

$$\dot{z} = C(z) + B^*(z)u \quad (3)$$

L'expression des différentes matrices est donnée en annexe.

Loi de régulation du régulateur de tir

La valeur vers laquelle doit tendre le vecteur contrôlé z_s est égale à : $z_{sd} = \begin{bmatrix} z_{rd} \\ \phi_d \end{bmatrix}$

La commande non-interactive suivante (cf. [4,5]) :

$$\Omega = -B_s^{-1} [C_s + \rho (z_s - z_{sd})] \quad (4)$$

où ρ est une matrice diagonale 3×3 , et où B_s et C_s sont les expressions non-linéaires de l'état figurant dans l'Eq.(1), donne par substitution de Ω dans l'Eq. (1) l'équation différentielle linéaire et découplée suivante :

$$\dot{z}_s = -\rho (z_s - z_{sd})$$

Pour des valeurs de z_{sd} constantes ou constantes par morceaux, l'erreur de poursuite $(z_s - z_{sd})$ est nulle en régime permanent moyennant un choix approprié de la matrice ρ . La loi de commande (4) a ainsi pour effet d'éliminer algébriquement les non-linéarités de l'Eq. (1).

Dans les conditions normales de fonctionnement du régulateur de tir, plus précisément lorsque la distance de tir n'est pas nulle, la matrice B_s est toujours inversible et le système (1) est donc contrôlable.

La loi (4) présente l'inconvénient de ne pas pouvoir contrer asymptotiquement des perturbations ou des erreurs de modélisation ; l'erreur sur les matrices B_s et C_s provient éventuellement des erreurs de mesure des variables d'état de l'avion, ou encore du fait que la cible n'est pas fixe au sol. Une solution consisterait à utiliser une loi non-interactive, proportionnelle et intégrale de la forme :

$$\Omega = -B_s^{-1} [C_s + \rho z_s + \lambda \int_0^t (z_s - z_{sd}) d\tau]$$

où ρ et λ sont des matrices diagonales 3×3 . Par substitution de Ω dans l'équation non-linéaire (1), z_s satisfait l'équation différentielle linéaire suivante :

$$\dot{z}_s = -\rho z_s - \lambda (z_s - z_{sd})$$

On montre que pour des perturbations additives constantes, l'erreur de régime permanent $(z_s - z_{sd})$ serait nulle. Le vecteur z_s n'étant pas mesuré, sauf pour la 3e composante ϕ , il a été jugé plus simple de mettre en oeuvre la loi proportionnelle (4) et de laisser au pilote la charge d'estimer les écarts $z_c = z_s - z_{sd}$ et de compenser les perturbations au moyen de son affichage z_{cp} . La loi (4) est alors évaluée au point courant de fonctionnement $z_c^* = z_{cp} + z_{rd}$; de même, la distance projetée X avion-cible dans la matrice C_s (cf. Eq.(1)) est remplacée par la distance D avion-point d'impact, puisqu'aucune mesure sur la cible n'est supposée faite.

La relation ordre du pilote-réponse de l'erreur de visée possède les propriétés suivantes :

- si $z_{cp} = z_c$, alors $\lim_{t \rightarrow \infty} z_c(t) = 0$; si l'ordre du pilote est identique à chaque instant à l'erreur de visée, l'erreur de visée tend donc vers zéro.

- si $z_{cp} = 0$, alors $z_c(0) = 0 \Rightarrow z_c(t) = 0 \forall t$; si l'ordre du pilote est nul, le point d'impact des obus reste fixe sur le sol.

- si $\phi = \phi_d = \text{constante}$, ϕ_d pouvant prendre des valeurs de 0 à 2π , une relation analytique approchée entre z_c et z_{cp} est donnée par :

$$z_c = -\rho_1 z_{cp} + \frac{u-u_T}{D} (z_c - z_{cp})$$

où $\rho_1 = \text{diag}(\rho_1, \rho_2)$, ρ_1 et ρ_2 étant les deux premiers éléments de la diagonale de la matrice ρ (cf. Eq.(4)). Pour une distance avion-cible fixe, $u-u_T = 0$ et la vitesse angulaire de la ligne de visée est donc proportionnelle à l'ordre du pilote. Le point d'impact instantané des obus devient immobile au sol quand le pilote annule son affichage.

Loi de régulation du régulateur de pilotage

La loi de commande (4) se réalise en fait suivant une certaine dynamique qui est celle de l'avion donnée par Eq. (2). La valeur vers laquelle doit tendre le vecteur contrôlé z est ainsi égale à :

$$z_d = D(\Omega_d, x_v)$$

Ω_d se substituant à Ω dans Eq. (4).

La poursuite de z_d par z se fait au travers d'un régulateur de pilotage structuré en deux niveaux :

- le premier niveau reproduit une réponse z_d convergeant vers z_d de manière idéale mais restant représentative du mouvement de l'avion, grâce à un modèle d'avion défini par des équations dynamiques où n'intervient aucune perturbation extérieure ; il conditionne la manière dont l'avion doit répondre aux ordres de manoeuvre de grande amplitude ;

- le deuxième niveau, qui est l'organe véritable de régulation, asservit avec une dynamique relativement plus rapide le mouvement réel de l'avion à la réponse du modèle, ce qui implique une compensation d'une part des erreurs de modélisation, d'autre part des effets des perturbations extérieures.

Ce partage en deux niveaux de régulation assure une bonne manoeuvrabilité de l'avion (1er niveau), associée un bon comportement en présence de turbulence (2ème niveau).

1) Afin de mieux rendre compte au 1er niveau du comportement de l'avion réel, et d'avoir ainsi des gains importants au 2ème niveau pour contrer les perturbations extérieures, un modèle d'avion non-linéaire est utilisé, régi par les équations non-linéaires (2). L'application d'une méthode de régulation non-linéaire semblable à celle utilisée pour le régulateur de tir donne la loi de commande suivante (cf. Eq.(3)):

$$u_m = -B_m^{-1} [C_m + qz_m + \Lambda \int_0^t (z_m - z_d) dt] \quad (5)$$

L'indice m indique que les variables appartiennent au modèle. On vérifie que la matrice B_m est inversible, c'est-à-dire que le système (3) est contrôlable. L'erreur de poursuite $(z_m - z_d)$ est nulle en régime permanent moyennant un choix approprié des matrices diagonales q et Λ . L'utilisation d'une loi proportionnelle et intégrale au lieu d'une loi proportionnelle permet en outre d'accroître le nombre des paramètres de réglage et par conséquent, une plus grande liberté dans le choix de la forme de la réponse (amortissement, fréquence).

2) Le 2ème niveau utilise une méthode classique d'optimisation linéaire avec un indice de coût quadratique pour établir une loi permettant d'une part la poursuite du vecteur z_m du modèle par le vecteur z de l'avion, garantissant d'autre part la stabilité de l'avion en boucle fermée. La loi est améliorée du point de vue de la robustesse (annulation des effets des erreurs de modélisation et des perturbations extérieures) grâce à la technique exposée dans [3,6]. La stabilisation du mouvement de l'avion est étudiée avec un vecteur d'état réduit $x_r = (x_v^T, \Omega^T)^T = C_r x$ relatif à la dynamique seule de l'avion.

L'écart $\delta x_r = x_r - x_{rm}$ entre l'avion et le modèle, ainsi que l'erreur de régulation $e = z - z_m$ sont donnés par le système linéarisé suivant, en remarquant que $z = C(x_r)$:

$$\begin{aligned} \delta \dot{x}_r &= F \delta x_r + G \delta u + \delta_1 \\ e &= H \delta x_r + \delta_2 \end{aligned}$$

où $\delta u = u - u_m$, $F = C_r \frac{dA}{dx}(x_0) C_r^T$, $G = C_r B$ et $H = \frac{dC}{dx_r}(x_{r0})$ (cf. Eq.(2)).

Le point de linéarisation x_0 correspond au vol rectiligne uniforme. Les quantités δ_1 et δ_2 rendent compte des résidus de linéarisation de $A(x)$ et $C(x_r)$ respectivement. Pour des valeurs de δ_1 et δ_2 constantes par morceaux, le système précédent est équivalent au système :

$$\begin{bmatrix} \delta \dot{x}_r \\ e \end{bmatrix} = \begin{bmatrix} F & 0 \\ H & 0 \end{bmatrix} \begin{bmatrix} \delta x_r \\ e \end{bmatrix} + \begin{bmatrix} G \\ 0 \end{bmatrix} \delta u$$

On vérifie que le système est contrôlable. La minimisation de l'indice de coût quadratique :

$$J = \int_0^\infty \{ (\delta x_r^T e^T) Q \begin{bmatrix} \delta x_r \\ e \end{bmatrix} + \delta u^T R \delta u \} dt$$

où Q et R sont des matrices de pondération, donne la loi optimale :

$$\delta u = K \begin{bmatrix} \delta x_r \\ e \end{bmatrix}$$

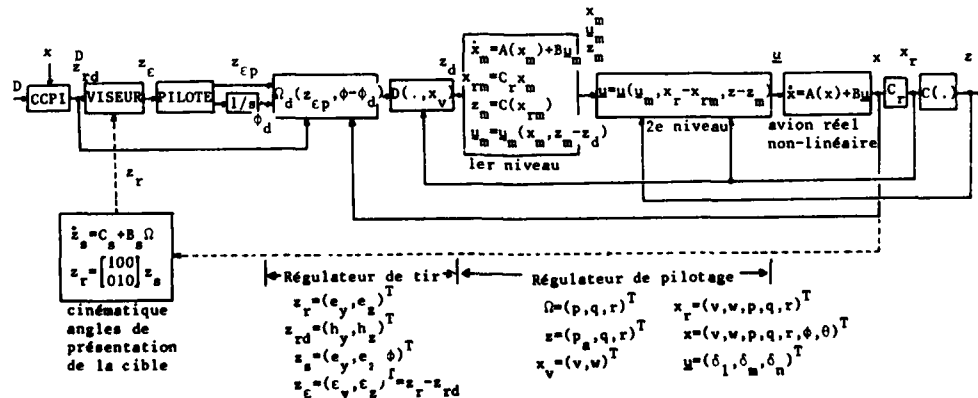


Fig. 4 Représentation du système de commande sous forme mathématique.

c'est-à-dire :

$$u = \delta u_0 + u_m + K_1(x_r - x_{rm}) + K_2 \int (z - z_m) d\tau \quad (6)$$

La réponse du système commandé par la loi (6) est donc telle que :

$$\delta x_r \rightarrow 0 \text{ et } e \rightarrow 0 \quad \text{lorsque } t \rightarrow \infty$$

Dans l'Eq. (6), u_m est donné par Eq. (5), $K = [K_1, K_2]$ est la matrice des gains optimaux et δu_0 est une constante d'intégration qui peut être elle-même optimisée [3]. La valeur de δu_0 est prise ici égale à 0 pour simplifier.

Les différentes étapes conduisant à l'élaboration des braquages des gouvernes à partir des ordres du pilote, sont résumées sur la Fig. 4.

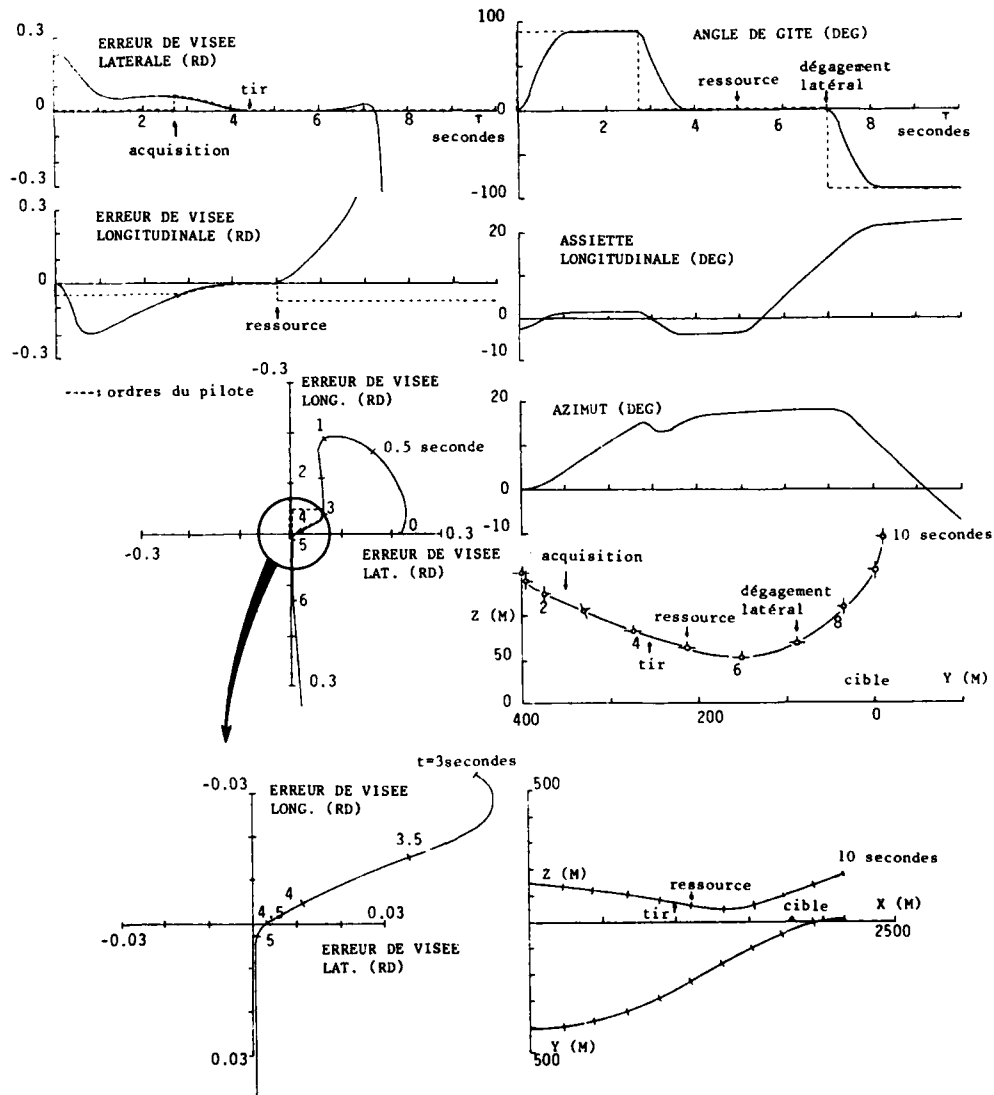


Fig.5. Simulation numérique d'une passe de tir air-sol

4. SIMULATION DE PASSES DE TIR AIR-SOL

Les résultats d'une simulation numérique d'attaque au sol sont montrés sur la Fig. 5. Les conditions de la simulation sont définies par une vitesse initiale de l'avion égale à 230 m/s, une assiette longitudinale initiale égale à -2° et un angle de gîte initial nul. La loi de PO a été utilisée pour toute la durée de la passe de tir, y compris la phase de ressource évasive ; aucune commutation de lois de pilotage n'est donc faite. Les coordonnées initiales de la cible dans le repère avion ($X = 1800\text{m}$, $Y = 400\text{m}$, $Z = 150\text{m}$) traduisent des conditions assez sévères concernant la présentation de l'avion face à la cible (altitude relativement basse, écart latéral initial important). Ces conditions ont été choisies pour vérifier les performances du système de commande pour des manoeuvres de roulis de grande amplitude.

La Fig. 5 montre que, la cible étant latéralement très écartée à l'instant initial ($0,22\text{rd}$), une commande simultanée $\phi_d = 90^\circ$, $\epsilon_{yp} = 0$ et $\epsilon_{zp} = 0,05\text{rd}$ est affichée par le modèle de pilote (supposé parfait et sans retard, ni dynamique). L'effet de la commande est de produire dans le repère lié à l'avion une rotation de la cible autour de la ligne de visée, l'erreur de visée initialement latérale devenant progressivement longitudinale. A partir de l'instant $t = 1\text{s}$ où l'angle de gîte commandé est atteint, la commande ϵ_{zp} entraîne une réduction rapide de l'erreur de visée longitudinale ; la variation de l'erreur de visée latérale n'est pas affectée, conformément au découplage recherché. L'azimut évolue rapidement et au moment où l'erreur de visée devient inférieure à $0,08\text{rd}$ ($t = 2,7\text{s}$) le modèle de pilote annule la commande ϕ_d et effectue les corrections fines ($\epsilon_{yp} = \epsilon_y$, $\epsilon_{zp} = \epsilon_z$). L'erreur de visée est pratiquement annulée à partir de l'instant $t = 4\text{s}$.

Le modèle de pilote affiche $\phi_{yp} = 0$ et $\phi_{zp} = -0,07\text{rd}$ à l'instant $t = 5\text{s}$ pour commander la ressource d'évasive, puis $\phi_d = -90^\circ$ à l'instant $t = 7\text{s}$ pour commander le dégagement latéral.

Bien que sur le plan analytique il soit montré que le système de PO permette une amélioration de la visée tout en préservant les conditions classiques de pilotage, la pilotabilité d'un tel système doit être vérifiée sur un simulateur avec un pilote humain dans la boucle. Une version linéaire de la loi de pilotage [7,8], limitant les manoeuvres de l'avion à de faibles angles de roulis (inférieurs à 20°), a été testée par 4 pilotes d'essais sur un simulateur d'avion de combat. La faisabilité du concept du pilotage du point d'impact ou contrôle en vitesse de la ligne de visée a été vérifiée par les pilotes, des appréciations favorables sur ce mode de PO ayant même été formulées : facilité de pilotage, visée très stable.

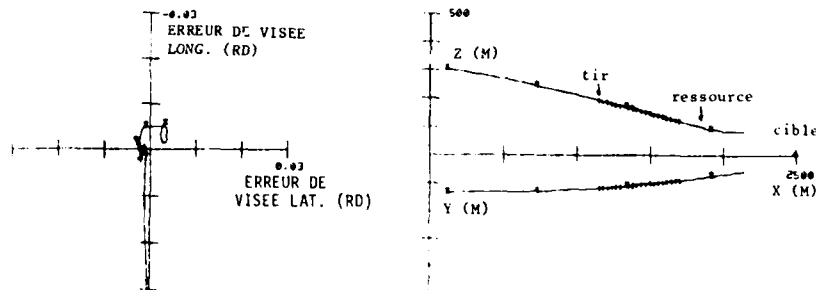


Fig. 6. Simulation pilotée d'une passe de tir air-sol

La Fig. 6 montre, avec la même version de régulateur linéaire, l'enregistrement d'une passe de tir typique où sont représentées de gauche à droite :

- l'évolution des erreurs de visée ϵ_y et ϵ_z ; c'est aussi la trace des obus au sol, la cible étant située au centre de la figure ;
- la trajectoire de l'avion (évasive non représentée).

Les courbes sont graduées tous les 600m par un astérisque (*) à partir d'une distance fixe égale à 2400m de la cible. Les croix (+) matérialisent les instants du tir. Sur cet exemple, la force du vent latéral est égale à 20 noeuds ; sur la cible de dimension $4\text{m} \times 4\text{m}$, le pilote a obtenu un score simulé de 20 obus ; la vitesse à l'instant du tir est égale à 479 noeuds.

5. TRAJECTOIRE D'APPROCHE NON-RECTILIGNE

Une approche non-rectiligne sur une cible défendue augmente les chances de survie de l'avion par rapport à une approche rectiligne. Le système de PO de la Fig. 4 permet un déplacement de l'avion tout en maintenant la visée stabilisée grâce à la commande de roulis.

Le pilote dispose en effet de 3 commandes pour piloter l'avion :

- commandes de vitesse de déplacement de la ligne de visée \dot{z}_p ;
- commande de roulis $\dot{\phi}$.

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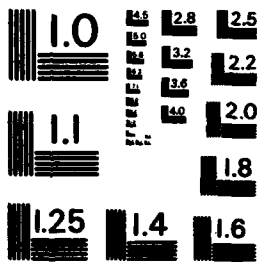
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Lorsque z_{ep} est nul, le point d'impact demeure superposé à la cible ; il est facile de voir que si l'angle de gîte de l'avion est nul, la trajectoire sera contenue dans un plan vertical. Une mise en roulis en visée stabilisée a donc pour effet d'incurver la trajectoire d'approche. Le déplacement de l'avion peut être mis en évidence en examinant l'évolution de l'axe cible-avion repéré par les angles de gisement et de site G et S (cf. Fig.7).

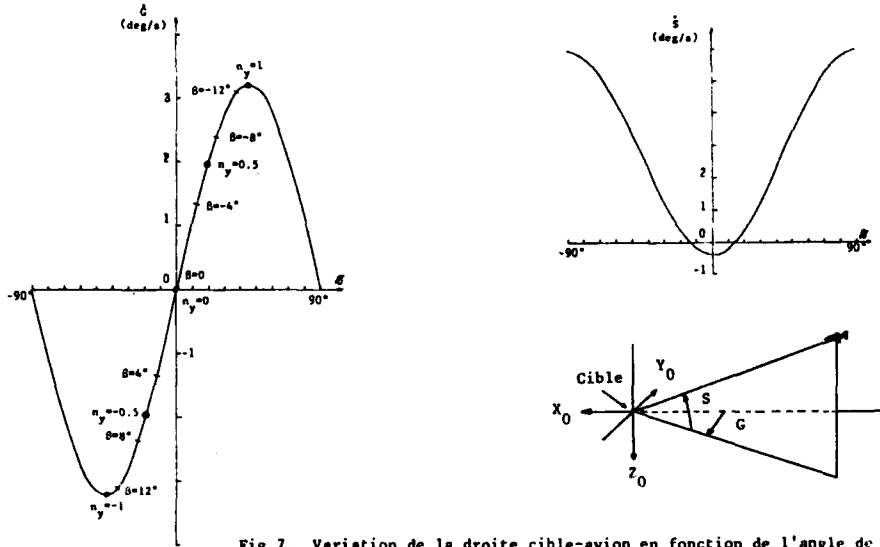


Fig.7. Variation de la droite cible-avion en fonction de l'angle de gîte (distance figée 750m, vitesse 230m/s)

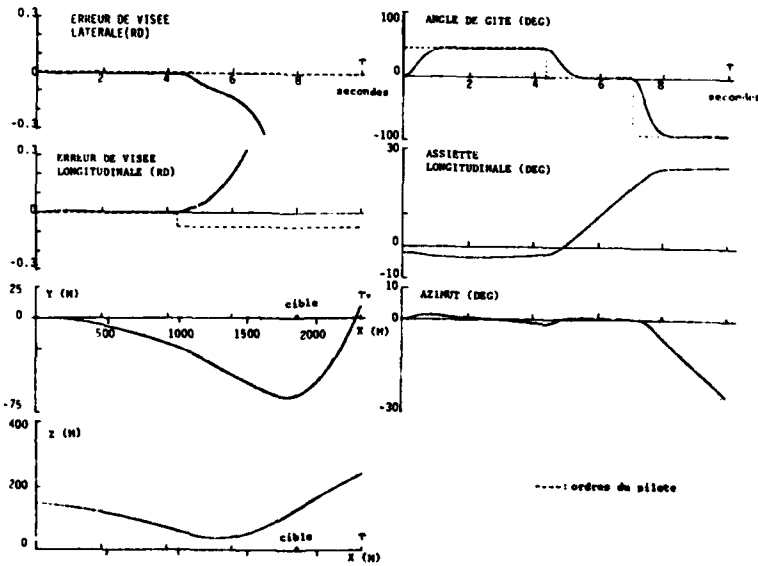


Fig.8. Simulation numérique d'une trajectoire de visée stabilisée

Moyennant une hypothèse simplificatrice consistant à figer la distance avion-cible, les valeurs de régime permanent des vitesses angulaires \dot{C} et \dot{S} en visée stabilisée peuvent être exprimées analytiquement en fonction de l'angle de gîte :

$$\begin{aligned}\dot{C} &= \dot{\psi} = (q \sin \phi + r \cos \phi) / \cos \theta \\ \dot{S} &= -\dot{\theta} = -q \cos \phi + r \sin \phi\end{aligned}$$

avec

$$\begin{aligned}q &= a \cos \phi \\ r &= b \sin \phi\end{aligned}$$

les quantités a et b dépendant de la distance et des coefficients aérodynamiques de l'avion. La Fig. 7 représente la variation de la droite cible-avion pour une distance fixe égale à 750m. La variation maximale en gisement est obtenue avec un angle de gîte égal à $\pm 45^\circ$ quelle que soit la distance. La variation en site est maximale pour $\phi = 0$ ou $\phi = \pm 90^\circ$ suivant les valeurs respectives de a et b . Il convient cependant de vérifier que ces configurations correspondent à un cas de vol réaliste ; à titre d'exemple, la courbe $\dot{C}(\phi)$ a été graduée en dérapage (jusqu'à 12°) et en facteur de charge latéral (jusqu'à $1g$), la vitesse de l'avion étant égale à 230m/s. On constate sur cet exemple que si le dérapage était limité à 5° et le facteur de charge latéral à 0,5g, l'angle de gîte maximum autorisé pour incurver la trajectoire serait inférieur à 20° .

L'étude paramétrique précédente ne donne qu'une valeur de régime permanent et approchée de la variation de la droite cible-avion, l'hypothèse simplificatrice d'une distance constante ayant été faite. La Fig. 8 montre le déplacement de l'avion sur une simulation numérique de passe de tir où le modèle de pilote affiche $\zeta_{CP} = 0$ et $\phi_d = 45^\circ$. Le déplacement latéral est égal à 24m au bout de 4,4 secondes (instant de la ressource) ; par rapport à la trajectoire rectiligne de même pente initiale, le déplacement vertical est de 33m. La visée est stabilisée avec une erreur inférieure à $0,5^\circ$ en régime transitoire.

6. CONCLUSION

Pour augmenter l'efficacité de la visée en tir au canon air-sol, une loi de pilotage non-conventionnelle a été proposée. Par cette loi, la ligne de visée est commandée en vitesse au travers des ordres de pilotage du pilote. La mise en oeuvre de la loi suivant une architecture de système comportant un régulateur de tir et un régulateur de pilotage placés en cascade permet la stabilisation du mouvement de l'avion au travers d'une boucle indépendante. L'utilisation des méthodes de calcul de commandes non-linéaires étend le domaine de vol de l'avion à des manoeuvres d'alignement de grande amplitude sans nuire à la précision de la visée. Les résultats de simulation montrent que le système de pilotage facilite les corrections de visée en tir air-sol tout en sauvegardant les conditions du pilotage classique.

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ANNEXE

$$C_s = \begin{bmatrix} \frac{v_T - v + (u - u_T) \operatorname{tge}_y}{X(1 + \operatorname{tg}^2 e_y)} \\ \frac{v_T - w + (u - u_T) \operatorname{tge}_z}{X(1 + \operatorname{tg}^2 e_z)} \\ 0 \end{bmatrix} \quad B_s = \begin{bmatrix} \frac{\operatorname{tge}_z}{1 + \operatorname{tg}^2 e_y} & \frac{\operatorname{tge}_y \operatorname{tge}_z}{1 + \operatorname{tg}^2 e_y} & -1 \\ \frac{\operatorname{tge}_y}{1 + \operatorname{tg}^2 e_z} & 1 & -\frac{\operatorname{tge}_y \operatorname{tge}_z}{1 + \operatorname{tg}^2 e_z} \\ 1 & \sin \phi \operatorname{tg} \theta & \cos \phi \operatorname{tg} \theta \end{bmatrix}$$

$$A = \begin{bmatrix} A_1 = pw - ru + g \cos \theta \sin \phi + Y_v v + Y_p p + Y_r r \\ A_2 = qu - pv + g \cos \theta \cos \phi + Z_0 w + Z_w w + Z_q q \\ A_3 = L_v v + L_p p + L_r r + L_{qr} qr + L_{pq} pq \\ A_4 = M_0 + M_w w + M_q q + M_{pr} pr + M_{pr}' (p^2 - r^2) \\ A_5 = N_v v + N_p p + N_r r + N_{pq} pq + N_{qr} qr \\ A_6 = p + q \sin \phi \operatorname{tg} \theta + r \cos \phi \operatorname{tg} \theta \\ A_7 = q \cos \phi - r \sin \phi \end{bmatrix} \quad B = \begin{bmatrix} Y_{\delta 1} & 0 & Y_{\delta n} \\ 0 & Z_{\delta m} & 0 \\ L_{\delta 1} & 0 & L_{\delta n} \\ 0 & M_{\delta m} & 0 \\ N_{\delta 1} & 0 & N_{\delta n} \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix}$$

$$C^H = \begin{bmatrix} (qA_1 + rA_2 + uA_3 + vA_4 + wA_5) / V \\ A_6 \\ A_7 \end{bmatrix}$$

$$B^H = \begin{bmatrix} (qY_{\delta 1} + uL_{\delta 1} + wM_{\delta 1}) / V & (rZ_{\delta m} + vM_{\delta m}) / V & (qY_{\delta n} + uL_{\delta n} + wM_{\delta n}) / V \\ 0 & M_{\delta m} & 0 \\ N_{\delta 1} & 0 & N_{\delta n} \end{bmatrix}$$

$$D = [(pu + qv + rw) / V, q, r]^T$$

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