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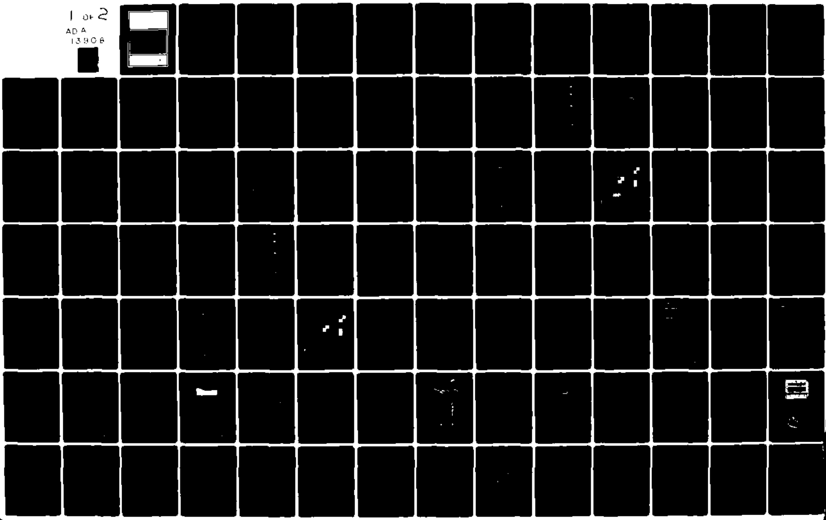

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AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARD CONFERENCE PROCEEDINGS No. 314

Guidance and Control Technology for Highly Integrated Systems

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY ON BACK COVER

AGARD-CP-314

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD *Conference Proceedings No.314*

GUIDANCE AND CONTROL TECHNOLOGY FOR HIGHLY INTEGRATED SYSTEMS

Papers presented at the Guidance and Control Panel 33rd Symposium held at the Agios-Andreas AFB, Marathon, Athens, Greece on 13–16 October 1981.

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Published February 1982

ISBN 92-835-0309-0

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

PREFACE

It is now rather more generally recognized that in order to achieve an adequate to high P_k, Probability of Kill, that NATO systems have to pursue a higher degree of integration than in the past *while yet being* adequately flexible and survivable. This was the essential theme of this Symposium which addressed the various relevant issues rather broadly in its various sessions, as suggested by the program listed at the end. Rather generally, such *systems, either on a specific platform* or else *in a battlefield scenario* involve the proper coupling of front end sensors to data processing systems which in turn either control the delivery of a weapon system accordingly or else display the requisite information for the system operator/controller who in turn controls the delivery of the weapon system. The guidance and control implications and realizations for systems functioning within such an integrated environment then follow accordingly, including the specifications on requisite mid-course guidance and control systems, terminal seeker capability requirements, and other fundamental requirements.

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Chairman: Mr U.K.Krogmann, Ge

SESSION V – AFFORDABILITY AND SURVIVABILITY CONSIDERATIONS

Chairman: Professor J.T.Shepherd, UK

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* Not available at time of printing.

TECHNICAL EVALUATION REPORT

by

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INTRODUCTION

The fact that NATO forces must react to WPF (Warsaw Pact Forces) of vast quantitative superiority in all areas through the continuing development of qualitatively superior weapons has been generally acknowledged for sometime. This was essentially the theme for the 33rd Guidance and Control Panel Symposium, namely, an examination of a number of significant diverse issues in developing qualitatively superior NATO weapons systems. This enormously important area will probably continue to receive treatment at various times in the future in subsequent NATO AGARD symposia.

Before examining the content of the 33rd and GCP Symposium it is perhaps worthwhile to briefly examine several examples of Air Force, Army, and Navy systems that achieve a most highly significant qualitative superiority through a high degree of effective systems integration. It should also be noted that at all times any approach to qualitative superiority through highly integrated systems must not be unduly exposed thereby to vulnerabilities as the result of enemy actions. The program content of the 33rd GCP Symposium systems was similar to those noted in the examples below.

First, exploring an Air Force example, it is highly desirable to blunt any WPF attempt to support a possible break-through (first echelon) of land forces through reinforcement (2nd echelon) by denying those 2nd echelon reinforcements. These reinforcements will, of course, include armor, APCs (armored personnel carriers), SPHs (Self-Propelled Howitzers), and other systems and support elements. In order to deny these reinforcements they must first be detected and targeted, that is, their composition and position must be determined to sufficient accuracy. At present in the NATO community this is being pursued through several means including airborne radar surveillance platforms. Once the 2nd echelon forces have been targeted then standoff weapons, such as Assault Breaker, may be delivered against them. These stand off weapons typically use mid-course guidance as well as terminally guided submunitions in order to achieve a high P_k (Probability of Kill). Of course, interdiction aircraft with target acquisition and laser designating pods may be utilized in addition to other weapons systems possibilities.

Now considering an Army system example, statistics from previous Army land battles have demonstrated that, for example, 2000 artillery projectile rounds are required before an armored kill is achieved. There are numerous serious faults with this. First, the highly mobile WPF would overrun NATO land forces before 2000 rounds per target kill could be expended. Secondly, the logistics requirements for 2000 rounds per kill are no longer tolerable for NATO forces. Other serious disadvantages may be readily cited. Obviously, what is desired is a highly mobile artillery piece, an SPH (Self Propelled Howitzer), which can achieve a very high P_k per projectile round. Such can, in fact, be achieved. The basic principles include an SPH with a precision (Self) position determination system such as GPS, JTIDS, or PLRS (Position Locating and Reporting System). In addition, an airborne surveillance system such as SOTAS (Standoff Target Acquisition System), 10 (Makrinos)*, serves the essential function of targeting and data linking targeting information to the SPH. In turn, the SPH then fires a PGM (Precision Guided Munition) which, as a result of its self-contained terminal seeker and guidance and control capability, can achieve a very high P_k. In other words, the capability of a target kill per round is well within the realm of feasibility.

Naval systems are also, of course, rich with possibilities in integrated systems. For instance, passive or active, how sonar arrays functioning in concert with towed arrays, either on surface ships or on submarines, can effectively target, that is, carry out the so-called localization process, for the delivery of stand-off weapons which can then relocalize when in the vicinity of the target in order to achieve a high P_k.

*Numbers and author(s) name(s) refer to papers presented at the 33rd GCP Symposium. See Contents, pages v and vi.

We turn now to an examination of the contents of the 33rd GCP Symposium. After noting briefly that the keynote address covered issues such as those noted above, among other issues.

OPERATIONAL REQUIREMENTS SESSION

The introduction demonstrated by three significant examples for Air Force, Navy, and Army systems how operational requirements can be met in a much more effective manner through systems integration, and this was elaborated on further in this session. The paper by Chislett[1] developed this by pursuing the specific but very broad area of air defense systems issues and the necessary systems in the specific case of the UKADGE, UK Air Defense Ground Environment, Improvement Program. The second paper by Krogull and Hutter[2] examined the broad and fundamental issues of command and control systems and their force multiplier implications. How can operational requirements be determined either for specific platforms or systems or else for a broad collection of systems? There is, of course, no single specific means for doing so, but rather a combination of means. In particular, analyses of operational situations can be essentially useful, and, indeed, in many instances is the only available avenue. In other cases, prior experiences with particular platforms can be essentially useful. Still in other cases field exercises such as Reforger, Strong Express, Red Flag, and others are essentially useful. In truth, the combination of all these, that is, analyses, prior experiences with specific platforms, large scale field exercises, and other such means are essentially valuable in defining operational requirements and, it might be added, operational doctrine, for instance, as well. Implied in all of the above, of course, is an adequately accurate definition of the threat, not only as it is, but as it might be. The fundamental principle here is the predictive intelligence and preemptive engineering as the context within which operational requirements are to be evolved.

SESSION ON COOPERATIVE INTERDEPENDENT SYSTEM CONSIDERATIONS

The papers by Stavropoulos[3], Brodie[5], and Milosevic[6] emphasized issues with respect to command and control systems. The paper by Stavropoulos presented a comprehensive rationale for the definition and development of command and control systems. The paper by Brodie on JTIDS and Milosevic on SINTAC III described specific systems realizations of systems for the ICNI, Integrated Communication, Navigation, and IFF function so essential to command and control systems. It is worth noting that such systems were first presented over 10 years ago at a GCP NATO AGARD meeting, suggesting once again the long development time all too characteristic of many importantly essential NATO systems. The paper by Brammer[4] was an interesting presentation of the integration of such systems for the achievement of an essential operational capability, namely, penetration survival by penetration at low altitude, that is, penetration at an altitude low enough to survive enemy air defense systems. One of the importantly essential messages in this session was that in the future integrated systems both at the platform level and for systems involving a large conglomeration of a broad array of platforms cooperative/interdependent system considerations will be indispensably essential.

SESSION ON THREAT AND TARGET DETECTION AND IDENTIFICATION

Considering the theme of this symposium, most certainly an absolutely essential issue is threat or target detection and identification. This session addressed a number of new systems, and technologies for this issue were addressed in this session. Target identification is one of the most challenging issues before the NATO community, and is being pursued by a variety of means. The paper by Vattrodt, et al.,[7] dealt with one of the most intriguing new approaches, acoustic means. It is difficult to tell yet where this technology will lead to in so far as operationally viable systems are concerned. However, for now, at least, it appears to definitely merit continuing consideration. The paper by Rayner, et al.,[8] explores the issue for target detection for air ocean surveillance systems. In the two decades after World War II these systems were in a high learning rate phase. The last decade and a half has seen some rather interesting and significant realizations of some of these systems, and the paper by Rayner dealt with a specific recent realization of advances in this area.

In yet another area, one of the most potentially significant recent technology developments has been in the development and testing of mm (millimeter) wave systems, both passive and active. This technology is being explored for Army, Air Force, and Navy systems. The paper by Heiden[9], in particular, dealt with one of the most interesting current development areas, namely, 94 GHY active mm systems for highly effective tank fire control systems.

Every Reforger field exercise from 1976 on has involved the utilization of SOTAS, (Standoff Target Acquisition System), one of the single most significant recent developments for battlefield surveillance over a very large area. The paper by Makrinos and Keneally[10] eminently demonstrated the potentially enormous significance of this system in a wide variety of field army applications.

SESSION ON AUTONOMOUS INTEGRATED WEAPONS SYSTEMS

It is well recognized that while many advantages accrue to integrated systems, the individual elements or platforms must have an adequate degree of autonomy in order to continue to be effective in the face of enemy threats which would attempt to disrupt or destroy the integrated functioning of NATO systems. The first two papers in this session by Lerche and Wicher[11] and by Camberlein, et al.,[12] both dealt with the extremely important issue of individual aircraft autonomously penetrating at a survivable low altitude through TF/TA (Terrain Following Terrain Avoidance) techniques and through terrain correlation techniques. The issues here now seem to be well in hand and are deployed in existing systems. Further significant advances may be expected to be continuously made. The next two papers by Fraedrich[13] and Arrowsmith[14] dealt with advances in Tornado integrated guidance and control systems providing a higher degree of effective autonomous mission capability. In both impressive systems accuracy performance improvements were demonstrated for these integrated systems. The paper by Leyland[15] continued the treatment of autonomous Tornado operations, applicable, of course, to other classes of aircraft as well, this time for the ADV, Air Defence Variant. The remarkable potential of the ADV both within the command and control environment and in the autonomous mode was developed and elaborated upon in this paper.

Another significant trend in improving the autonomous capability of aircraft was developed in the paper by Landy, et al.,[16]. Very significant improvements in air to air gunnery and air to ground weapon delivery performance through the utilization of IFFC, Integrated Fire Flight Control, concepts were presented in this paper. It is gratifying to see such significant techniques currently under active development. The paper by Shapiro[17] developed some significant operational scenarios which clearly demonstrated the highly significant potential for autonomous systems for rotary wing aircraft (helicopters) and described some detailed systems realizations. Although the paper by Bryant[18] was not presented the next GCP Symposium will be devoted to PGMs, Precision Guided Munitions, and so much will be presented in this area during the next Symposium.

Penetrating aircraft will be subject to an intense air defense environment. As these systems will have to carry a significant weapons payload capability anything in the way of a SPW, Self Protect Weapon, capability will have to represent a modest payload demand and yet be most highly effective. The paper by Johnson[19] presented a comprehensive analysis of the SPW design effectiveness tradeoffs, and presented some intriguing design points with apparently great potential in the SPW role.

SESSION ON AFFORDABILITY AND SURVIVABILITY CONSIDERATIONS

Callaghan's paper[20] led this session off with a comprehensive excellent articulation of the disciplined approach that will have to be followed. This was then followed by an excellent paper by Logus[21] that elaborated on a cycling approach to system testing that offers the potential of significantly greater reliability improvement. Nesline's paper[22] explored hardware and software trade-offs for one of the important new NATO air to air missile systems, the AMRAAM. This session closed with an excellent paper by Jones and Karmarkar[23] on a new modular software discipline for integrated systems which can accommodate growing requirements on integrated systems and verify the integrity of the system, indeed, most significantly, providing very impressive fault tolerance.

CONCLUSION

Integrated systems both for individuals platforms and for platforms which are within an overall system are now an established and growing phenomenon in the NATO community. This is, of course, an essential necessity in the face of the growing threat environment, both in numbers and capabilities. At the same time such integrated systems must have an adequate degree of autonomy because of the possibility of erosion of degrees of integration as a result of enemy action. Such integration must be achieved in an affordable and reliable manner. All these issues and many more were treated in this symposium. However, it is clear that, because of the continual growth and continuing developments in this very broad area throughout the NATO community that the subject area of this symposium will have to be scheduled again at appropriate times in the future.

USE OF JOINT TACTICAL INFORMATION DISTRIBUTION SYSTEM (JTIDS) FOR AUTONOMOUS AND COOPERATIVE WEAPON GUIDANCE

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SUMMARY

One especially useful application of force coordination is the use of a Joint Tactical Information Distribution System (JTIDS) tactical grid for both the autonomous and cooperative control of guided weapons. Although there are presently no plans or intentions to do so, JTIDS enables a weapon with moderate accuracy low cost inertial reference equipment to regularly update its grid location in a completely passive manner. A full exchange of tactical data (i.e., fire control information) can be made to the weapon over the JTIDS grid. Significantly greater guidance accuracy is also achieved due to the precise Time of Arrival (TOA) information available on the JTIDS grid.

1. INTRODUCTION

Successful tactical strategies against adversaries with greater resources of manpower and equipment require the complete coordination of available resources. A highly coordinated and flexible force can have its effectiveness greatly amplified as its assets can be brought together in the most effective manner. JTIDS is one way to achieve this unified command and control structure.

JTIDS is a jam resistant, secure CNI system presently in its Full Scale Development program phase. As a local tactical grid system, JTIDS possesses the ability to distribute fire control and weapon guidance data as well as precise TOA ranging information. Present (and planned) radio guidance systems require the establishment of special weapon support stations for both ground and airborne use. By contrast, a JTIDS based weapon system would operate with standard unmodified JTIDS tactical terminals which usually will have been deployed in the tactical area for other purposes.

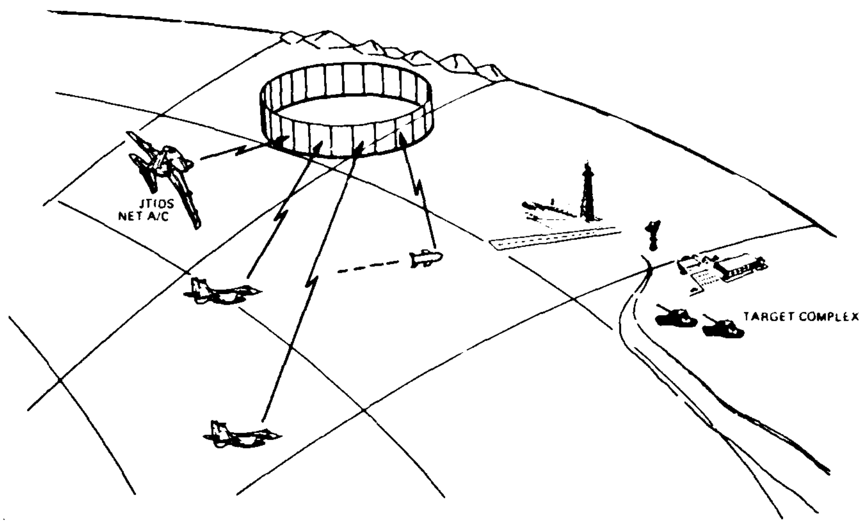
2. BENEFITS OF JTIDS BASED WEAPON GUIDANCE

Several distinct guidance benefits accrue with the use of JTIDS. Although many of these benefits can be achieved with existing systems, none except JTIDS has the ability to supply them all with such little system impact. Some of the more important of these features necessary for cooperative or C^2 guidance are listed below:

- Direct and Over-The-Horizon Communication with the weapon via a JTIDS net.
- Direct and Over-The-Horizon Guidance with JTIDS.
- Real Time knowledge of the location of targets and missiles in flight.
- Ability to retarget in flight, as each is located in a JTIDS Relative Navigation Grid.
- Improved Seeker Probability of Acquisition.
- Reduced Probability of Enemy Detection

Despite the fact that reception of JTIDS signals are on a line of sight basis, use of a JTIDS grid with typically dispersed members permits communication and control to be exercised between members that are not within direct sight of each other. This can be accomplished by means of the normally available JTIDS relay function. Thus fire control, targeting, force location, weapon status and distribution information can be made available to the command and control member of the grid on a near real time basis. In a similar manner, guidance information can be uplinked to a missile either in a direct or over the horizon application. JTIDS then supplies the information necessary for either cooperative or autonomous missile guidance.

In the cooperative case, two-way communication is maintained between JTIDS grid participants. In a JTIDS TDMA (Time Division Multiple Access) grid, communication is often represented by a communication "ring" or net, as shown in Figure 1. Each user has preassigned slots in which he can transmit information in a broadcast fashion. A TOA measurement is usually made upon the receipt of transmitted data. As each net member regularly transmits his grid location as well, it is possible for the weapon to compute its own grid location by the reception of data from two or more grid participants.



JTIDS AUGMENTED WEAPON GUIDANCE
FIGURE 1

For autonomous guidance using JTIDS, the weapon terminal need only receive; i.e., remain completely passive or radio silent in flight. Once synchronized into the JTIDS net, the missile terminal can listen to the normal transmissions of active net members to regularly locate itself in a navigational grid. While remaining passive, fire control information supplying data on target location or changes, arming and recall commands can be easily transmitted in multiple weapon scenarios. A form of JTIDS Relative Navigation is the most appropriate way of implementing control for both autonomous and cooperative cases. The only difference is that in the fully passive case position is correlated into system clock time. In practice, Relative Navigation is achieved by the optimal combination or statistical filtering of both radio ranging and inertial sensor data. In this manner, performance is achieved which is superior to that obtainable in either system alone (i.e., a radio navigation system or an inertial navigation system). The ranging portion of JTIDS is certainly more accurate than the positional accuracy of an inertial system. However, the inertial system supplies continuous navigation thereby permitting a navigation grid to be developed based on sequential range measurements from several net members non-synchronously. With this technique, navigation can in fact be achieved between as few as two net members. However, the most important feature of Relative Navigation is that it creates a network of users locked into a precise grid system. Each grid member acts as a distributed sensor in this network system which can be easily structured (via the TDMA link) to perform weapon guidance. DME target data are brought into this precise positioned grid, and coordinated weapon delivery can be routinely performed as positional data are exchanged between all participants in this common tactical grid. Thus, data sensed by one grid member (either external data such as targeting or inertial data such as member position) can be effectively utilized by another grid member as if sensed directly. This form of data augmentation is one of the most useful features of JTIDS Relative Navigation.

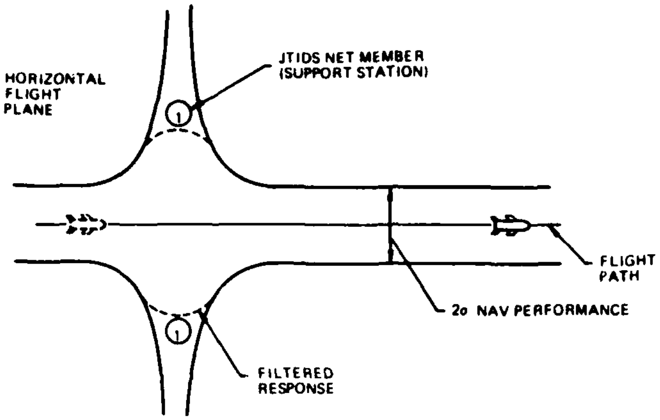
3. TERMINALLY AUGMENTED GUIDANCE

Although JTIDS navigation exhibits accuracies consistent with most high performance radio ranging systems, there are many weapon applications where greater performance is desired. To achieve such performance levels, terminal seeking systems (radio, radar, optical, etc.) are usually employed. There are additional reasons for using terminal guidance systems as well in conjunction with JTIDS. For example:

- To attack moving targets -
  Normally supplied fire control targeting updates may become stale very quickly. Even rapidly updated cooperative targeting via JTIDS is usually less desirable than autonomous target sensing.

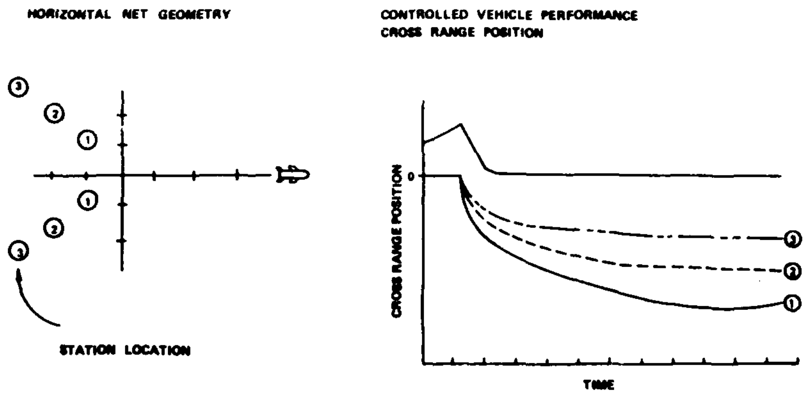
- Targets with EW jamming capability -
  By design, JTIDS terminals exhibit a strong jam-resistant capability. However, as a jammer is approached, there will be some range at which communication with the weapon is no longer possible. Autonomous terminal sensing is usually essential in these applications. The objective of JTIDS guidance is then to provide sufficient flight accuracy and positional knowledge to permit terminal sensor acquisition of the target.

In these terminal flight segments, JTIDS performance is no longer a factor with ultimate guidance accuracy then dependent upon terminal seeker accuracy and vehicle aerodynamic maneuverability constraints.



GDOP SINGULARITIES AND POSITIONAL PERFORMANCE
FIGURE 3

Figure 4 indicates the second or more common form of GDOP. Here weapon guidance is obtained in reference to alternate pairs of JTIDS stations indicated by the numerals 1 to 3. Resulting cross-range guidance performance for an aerodynamically controlled vehicle is shown on the right. The topmost curve represents actual vehicle performance while the lower curves represent where the JTIDS terminal believes the vehicle to be. Numbers 1, 2 and 3 correspond to the location of JTIDS support terminals shown on the left. Stations located at 3 represent good grid geometry which results in a cross range position error (as shown on the second curve from the top) which is essentially equal to the JTIDS 2 ranging error. Lower curves 2 and 1, represent progressively poorer performance, corresponding directly to the poor station locations shown on the left. Actually, these position sensitivities are not as bad as they may first appear due to the fact that most weapon trajectories are not over extremely short ranges. Thus the station location requirement is an important but not especially sensitive requirement for most applications.



GEOMETRIC DEPENDENCE OF PERFORMANCE
FIGURE 4

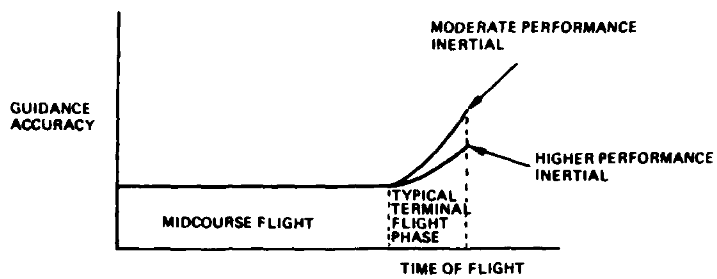
5. CONCLUSIONS

A standard JTIDS tactical net normally supplying secure communication and navigation functions can also be configured to support hybrid missile guidance. Due to the inherent relay feature of JTIDS, weapons can be commanded and guidance implemented even in over-the-horizon target cases.

With sufficient care in designing a missile guidance subnet, performance can be obtained consistent with normal JTIDS airborne operations. A particularly promising use of JTIDS guidance is in the midcourse flight phase, where it aids the establishment of required terminal flight conditions.

Nearly all terminal guidance systems need an inertial reference system to supply a measure of vehicle attitude. If the terminal flight phase is short enough, reverting to a pure inertial mode can often incur negligible guidance penalty. Additionally, inertial accuracy need only be moderate as terminal flight times are usually in minutes and can be supplied by either a strapdown or gimballed inertial reference. Accelerometers will of course be required with such a system. The two largest inertial errors to be considered are Gyro Drift and Platform Misalignment. Errors due to accelerometer bias will appear as a platform misalignment (after alignment) and are treated as such. Although the weapon's inertial reference system is to be leveled before flight, hybrid inflight alignment will be performed with JTIDS Relative Navigation. A JTIDS based alignment can be expected to update position and day to day gyro drift, but will not offer much improvement for accelerometer calibration. This is because the basic measurement supplied by JTIDS is position; i.e., velocity is a derived variable. In practice this accelerometer bias error will essentially not exist (due to pre-launch leveling).

Figure 2 indicates the general qualitative effect of shifting from a JTIDS inertial mechanization to a pure inertial terminal mechanization. Considering constant station geometry, JTIDS performance remains fairly flat until terminal phase where it grows steeply with flight time.



JTIDS SUPPORTED MIDCOURSE GUIDANCE
FIGURE 2

Naturally, in a missile with terminal seeking capability, impact errors are dependent upon the performance of this device. Effectively, inertial errors do not accrue in flight but are "reset" by the terminal seeker in a hybrid filtered implementation.

4. GEOMETRIC DILUTION OF POSITION (GDOP)

Aside from normally encountered ranging errors, GDOP is one of the most important factors in achieving adequate guidance accuracy. These positional errors (and resulting guidance errors) occur due to unfavorable location of JTIDS grid participants. GDOP errors are not unique to JTIDS, but routinely occur in most ranging systems utilized for navigation and guidance. Although related, two distinct types of GDOP errors are normally encountered in guidance applications. The first represents a true mathematical singularity, while the second is due to less optimal JTIDS grid member location. Reference to Figure 3 pictorially indicates the guidance singularity case. This condition is often encountered in cases where a weapon is launched external to the JTIDS grid. Ideally, a weapon should be initialized and launched within the JTIDS grid. It is feasible however, to launch external to the grid and initialize upon entering. This is the case represented in Figure 3 where the weapon is shown flying from left to right. Typically, JTIDS stations will be located along the flight path as depicted by reference number 1 in Figure 3. Superimposed upon the flight path are heavy lines representing the standard deviation of the navigation performance (i.e., ranging plus geometry errors). Note that as the vehicle approaches a point adjacent to the JTIDS station, navigational errors increase assymptotically. This is to be expected as the JTIDS participants form a ranging set with the weapon. As the weapon approaches these terminals, virtually no along track data can be obtained. Fortunately this is usually a relatively short portion of flight and weapon guidance can be obtained through an appropriate guidance filter design as indicated by the dashed lines.

S I N T A C - 3

par

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1 - AMELIORATIONS RECHERCHEES PAR RAPPORT AU JTIDS/SINTAC-2

Après une analyse détaillée des performances du JTIDS/SINTAC-2 et des besoins opérationnels exprimés pour le NIS, nous sommes venus à la conclusion que les améliorations suivantes doivent être apportées aux terminaux JTIDS/SINTAC-2.

- Côté MIDS
  - . simplifier les circuits d'acquisition,
  - . augmenter la capacité d'un terminal,
  - . augmenter la souplesse d'utilisation du terminal avec des formes de messages différentes : compactes ou distribuées,
  - . maintenir la protection contre le brouillage pour les terminaux simplifiés,
  - . permettre la simplification des terminaux pour les petits porteurs.
- Côté NIS
  - . utilisation du canal asynchrone pour l'identification par Q et R,
  - . avoir une disponibilité proche de 100 % pour l'identification par Q et R,
  - . pouvoir simplifier le terminal multifonction à la seule fonction d'identification par Q et R, si nécessaire, sans le pénaliser, tout en gardant l'interopérabilité avec tous les terminaux multifonction.

Ces caractéristiques s'ajoutent à celles déjà obtenues dans le SINTAC-1 et le SINTAC-2 -de facilité d'installation dans les porteurs- en ne rendant pas obligatoire l'utilisation de l'antenne directive à l'interrogation, grâce à la Désignation Interne.

Mais ceci n'exclut pas l'utilisation de l'antenne directive, soit pour la désignation, soit seulement pour l'augmentation de la protection contre le brouillage si l'une ou l'autre sont jugés nécessaires.

C'est une possibilité de plus qu'offre le système qui :

- permet l'identification par les porteurs sur lesquels il est difficile, sinon impossible, d'installer l'antenne directive de performances valables,
- améliore, dans la majorité des cas, les performances de discrimination,
- augmente la fiabilité et la sécurité du système, et,
- simplifie les terminaux.

CONCEPT NOUVEAU DU SYSTEME : SINTAC-3

- modification du procédé d'acquisition basé sur la dispersion à très large bande du spectre de modulation conformément à la technique proposée dans le NIS,
- maintien sans changement du procédé de modulation du texte utilisé dans le JTIDS/SINTAC-2,
- interopérabilité avec le JTIDS/SINTAC-2.

COMPLEXITE DES TERMINAUX (figure 1)

Cette figure montre la complexité relative entre le terminal JTIDS/SINTAC-2 classe II et le terminal SINTAC-3 de capacité et de protection contre le brouillage similaires.

TERMINAL JTIDS/SINTAC-2 CLASSE II AVEC 1 CHAINE DE RECEPTION

Utilise 8 récepteurs pour l'acquisition -8 récepteurs sont nécessaires pour une protection suffisante du préambule contre le brouillage- mais il est limité à la réception d'un seul texte à la fois.

TERMINAL SINTAC-3 DE CAPACITE EQUIVALENTE

Utilise une chaine de réception constituée de deux parties :

- une pour l'acquisition du préambule,
- une pour la réception du texte,

avec la même protection contre le brouillage due à l'utilisation d'une largeur de bande de 80 MHz pour le préambule d'acquisition.

PROTECTION CONTRE LE BROUILLAGE DU CANAL ASYNCHRONE (figure 2)

La figure montre le diagramme de protection contre le brouillage du canal asynchrone et son amélioration par le SINTAC-3 par rapport à celle obtenue par le JTIDS/SINTAC-2.

Figure du haut : Modulation à étalement du spectre à très large bande et la distance de brouillage pour tous les types de brouilleurs y compris le brouilleur intelligent.

Figure du bas : Modulation à étalement du spectre faible et la distance de brouillage pour un brouilleur intelligent

Energie totale de brouillage identique dans les deux cas*

COMPLEXITE DES EQUIPEMENTS ET DIFFICULTES D'INSTALLATION CORRESPONDANTES (figures 3a et 3b)

La figure 3a représente les configurations opérationnelles dans lesquelles peut être utilisé un avion.

Sur la figure 3b du haut, sont représentés les équipements nécessaires pour répondre aux demandes du MIDS et du NIS dans un avion :

- équipements du JTIDS/SINTAC-2,
  avec deux antennes : haute et basse,
- dans la bande basse 1, le récepteur,
- dans la bande basse 2, l'émetteur récepteur,
  avec deux antennes, basse et haute, communes.
- dans la bande haute :
  - . le récepteur avec 3 antennes hautes et 3 antennes basses, pour avoir un bon diagramme de rayonnement,
  - . l'émetteur avec antenne directive placée dans le nez de l'avion.

Sur la figure du bas, est représenté le SINTAC-3 qui a les caractéristiques exigées par le MIDS et le NIS et qui utilise les informations fournies par le système de navigation autonome, installé de toute façon dans l'avion, mais non exploité par les systèmes particuliers (décrits ci-dessus).

Dans ce système, la configuration des équipements, matériel et logiciel, est telle que le terminal puisse satisfaire à la fiabilité demandée.

POUVOIR DE SEPARATION (figure 4)

La figure montre les avantages de la désignation interne sur la désignation externe pour l'identification par Q et R.

- DISTANCE : 150 KM
- DESIGNATION EXTERNE PAR ANTENNE DIRECTIVE :
  - ouverture du lobe de 4°
    réalise :
    - . séparation latérale : 10 km
    - . Séparation en distance: 30 m
- DESIGNATION INTERNE UTILISANT LE SYSTEME DE NAVIGATION AUTONOME
  - avec une précision moyenne de 1 km
    réalise :
    - . séparation latérale : 1 km
    - . séparation en distance: 30 m

La désignation interne réalise un pouvoir de séparation plus grand et offre des avantages évidents d'installation et de coût des équipements.

2 - CONCEPT DU SINTAC-3

FORME DU MESSAGE (figures 5 et 6)

Le message est constitué de deux parties :

- le préambule, et
- le texte.

Le préambule sert à trois fonctions :

- acquisition du message,
- différenciation des sous-canaux,
- prédésignation du ou des correspondants par zone ou individuellement.

Dans ce but il est divisé en deux parties :

- le préambule A, constitué de la trame d'acquisition
- le préambule B, constitué de deux mots :
  - . "MODE" définissant le sous-canal,
  - . "PREDESIGNATION" indiquant la zone intéressée ou fournissant le code individuel du correspondant.

---

* La bande large étant réduite de 100 MHz à 80 MHz, il y a une réduction de protection de 1 dB environ.

Le texte fournit :

- la synchronisation fine,
- lève les ambiguités que la prédésignation a pu laisser,
- l'information portée par le message.

- les impulsions du préambule sont modulées à très large bande,
- les impulsions du texte sont modulées à bande relativement étroite, égale à celle du JTIDS ou SINTAC-2.
- les impulsions du préambule et du texte sont transmises par "page" (fig. 6) avec :
  - un taux de remplissage faible,
  - la position des impulsions dans la "page" est pseudo aléatoire définie par le canal.

Les taux de remplissage du préambule et du texte sont définis pour :

- permettre émission et réception simultanées entrelacées sans perte sensible, ce qui entraine aussi la possibilité de réceptions des messages entrelacés dans le même canal,
- avoir des messages suffisamment courts pour répondre aux besoins d'identification par Q et R.

Cette forme de message :

- augmente la disponibilité des terminaux,
- simplifie sensiblement l'organisation des canaux,
- améliore la compatibilité entre les systèmes indépendants utilisant, sur le même porteur, la même bande de fréquence (par la réduction du brouillage mutuel dû au blocage par paquets des récepteurs pendant l'émission).

3 - CONSTITUTION D'UN TERMINAL (figure 7)

Un terminal est constitué :

- des récepteurs d'acquisition,
- des récepteurs de texte,
- de l'émetteur, et
- des dispositifs de traitement des signaux et des messages.

- RECEPTEURS D'ACQUISITION

Un terminal complet est constitué de :

- récepteur d'acquisition à bande large dans le canal asynchrone,
- récepteur d'acquisition à bande large dans les canaux synchrones organisés par division en temps,
- récepteur d'acquisition à bande étroite dans les canaux synchrones dont l'utilisation est faite essentiellement en temps de paix mais qui est utilisé également dans la période de transition pour assurer l'interopérabilité avec le système actuel (il peut dans ce but être doublé).

- RECEPTEURS DE TEXTE

- les récepteurs de texte sont à bande étroite à saut de fréquence identiques aux récepteurs du système actuel.
  A cette différence que le taux de remplissage des messages est plus faible et que les réceptions entrelacées sont possibles.
- le nombre de ces récepteurs dépend de la classe du terminal,
- ils ne sont programmés qu'à la suite de réception du top de trame d'acquisition dans un des canaux disponibles,
- ils sont utilisés suivant la priorité des messages.

- EMETTEUR

Il y a un émetteur commun aux deux types de modulation

- bande large pour le préambule,
- bande étroite pour le texte.

- TRAITEMENTS

Le traitement des signaux et le traitement des messages sont adaptés à la nouvelle modulation, nouvelle forme des messages et nouvelle organisation des canaux.

4 - CLASSE DES TERMINAUX

Le terminal du SINTAC-3 est adaptable dans une très large gamme, à partir des grandes stations C^3, jusqu'au système d'arme à très courte portée -portée par l'homme- en passant par tous les intermédiaires.

Il est utilisé d'une façon régulière comme le système multifonction avec une antenne omni-directionnelle, mais si nécessaire,

- il peut être utilisé avec une antenne directive et pour un nombre réduit de fonctions et à la limite,
- il peut n'être utilisé que pour une seule fonction, plus spécialement pour l'identification comme un système spécialisé, dont la configuration est simplifiée en conséquence.

Mais avec cette caractéristique fondamentale que l'interopérabilité est assurée entre toutes les classes des terminaux.

Il peut de ce fait être parfaitement adapté à la période transitoire, dans laquelle il faut assurer la coexistence et la compatibilité avec les systèmes actuels militaires et civils.

Les différentes classes de terminaux sont caractérisées par :

- la puissance d'émission et leur portée,
- la capacité de traitement des messages et,
- la technologie suivant le type de porteur : sol, air, mer.

Un certain nombre de terminaux est représenté sur les planches figures 8 à 12.

SECURITE D'IDENTIFICATION DANS LE SYSTEME INTEGRE MULTIFONCTION PAR RAPPORT A CELLE DU NIS

- Le NIS est un système spécialisé qui utilise le seul procédé d'identification par question et réponse, en cas de panne l'identification est perdue.
- Le SINTAC utilise plusieurs procédés d'identification et les équipements redondants :

1. identification par diffusion :

   - message P et Identification indirecte,

2. Identification par question et réponse

Les deux, dans les deux familles de canaux :

- canal asynchrone, et,
- canaux synchrones.

Le système utilise, d'une façon normale, la désignation interne avec la position fournie par le système de navigation.
Mais en cas de besoin, on peut utiliser l'antenne directive si elle est disponible.

- la capacité d'identification du SINTAC est très élevée :

  - l'identification est prioritaire,
  - tous les récepteurs d'un terminal, si nécessaire, peuvent instantanément être utilisés pour l'identification,
  - le terminal peut répondre aux interrogations simultanées entrelacées.

Donc, sur les deux points :

- la redondance et la sécurité,
- la capacité instantanée,

le système intégré multifonction présente des avantages sur le système spécialisé.

- la charge d'identification par question et réponse est relativement faible car elle n'est utilisée que pour les porteurs inconnus,
- il est suffisant de réserver dans le canal synchrone cinq groupes de trois récurrences pour l'identification par Q et R (5 x 3 = 15 récurrences au total) pour répondre aux besoins de sa charge. Cette organisation des récurrences permet une tolérance d'erreur de synchronisation de $\pm$ 4 ms et le temps d'accès est $\leqslant$200 ms.
- la figure 13 montre l'organisation en temps du canal synchrone pour l'identification par Q et R avec la tolérance de l'erreur de la synchronisation.
- la figure 14 représente l'organisation avec la proportion d'occupation en temps du canal asynchrone et des canaux synchrones, pour un terminal type classe II.

SYSTEME MULTIFONCTION - SECURITE DE FONCTIONNEMENT

- REDONDANCE DANS LE SYSTEME,
- REDONDANCE ENTRE LES SYSTEMES,

Dans le SINTAC-3, l'ensemble des fonctions du MIDS et du NIS est réalisé dans la bande L : 962 - 1215 MHz.

L'objection est faite :

- qu'en cas de panne toutes les fonctions sont perdues simultanément, et,
- qu'un brouilleur unique peut brouiller l'ensemble des fonctions.

Il s'agit en fait d'un problème économique de redondance du matériel et de redondance entre les systèmes :

- en ce qui concerne les pannes, la configuration du système intégré est telle qu'elle assure la redondance du matériel dans les meilleures conditions économiques que dans le système spécialisé car la même redondance peut servir à plusieurs fonctions.
- en ce qui concerne le brouillage du système multifonction par un seul brouilleur tandis que pour deux systèmes spécialisés, dans deux bandes différentes, il est nécessaire d'avoir deux brouilleurs et que de ce fait la protection est meilleure, il est nécessaire d'examiner en fait le rapport du coût/efficacité dans les deux cas : le résultat est représenté dans le tableau ci-joint.

  On peut constater que la protection du système multifonction est équivalente à celle de deux systèmes spécialisés dans deux bandes différentes car le coût de la fonction d'identification est marginal mais que l'alternative du NIS en bande L de ce point de vue est plus défavorable, car dans ce cas le coût du système est double et les deux peuvent être brouillés par le même brouilleur.

Dans le cadre du système intégré multifonction, il est possible par contre d'améliorer le rapport coût/efficacité du système en réalisant la redondance du matériel avec des émissions dans deux bandes différentes, par exemple : bande L et bande S ou bande UHF.

Dans ce cas, l'augmentation du coût du système, pour assurer la redondance par rapport au coût de la redondance dans la même bande (la seule bande L) n'est que très partielle, elle ne concerne essentiellement que les antennes, et la partie d'E/R à la fréquence d'émission, tandis que le coût du brouilleur double car il faut un brouilleur dans chaque bande.

Ainsi, contrairement à l'objection faite, c'est dans le cadre du système intégré multifonction que se trouve la solution conduisant au rapport coût/efficacité optimal.

La redondance partielle peut aussi être réalisée entre les systèmes de communication essentiellement pour la phonie, comme on peut le voir sur la figure 16.

TEMPS DE PAIX, DE CRISE ET DE GUERRE

TEMPS DE PAIX :

- seuls les récepteurs à bande étroite sont utilisés,
- les récepteurs du texte sont complétés d'un circuit d'acquisition simplifié,
- le processus d'acquisition utilise :
  - . une fréquence unique,
  - . un code,
  - . avec une distribution pseudo-aléatoire en temps, la même utilisée dans le SINTAC-3.

TEMPS DE CRISE ET TEMPS DE GUERRE :

- les procédés complets du SINTAC-3 sont utilisés :

  Le programme d'acquisition à large bande se déroule en permanence (le temps de paix compris) et il est disponible sans aucun préavis.

PHASE DE TRANSITION - CONFIGURATIONS DU TERMINAL (figure 17)

1 - Solution actuellement envisagée

Les systèmes indépendants : le JTIDS/SINTAC-2 et le NIS.

2 - Solutions basées sur le SINTAC-3

2.1. Phase transitoire :

- JTIDS/SINTAC-2 sans modification et en plus :
- SINTAC-3 : émetteur-récepteur à large bande du préambule pour acquisition et les fonctions du NIS.

2.2. Phase finale :

SINTAC-3 complet remplissant l'ensemble des exigences du MIDS et du NIS.

CONCLUSION

INTERET DE LA SOLUTION : SINTAC-3

- Simplification des terminaux,
- Augmentation de la capacité du système à coût constant,
- Protection contre le brouillage du canal asynchrone,
- Protection contre le brouillage des terminaux simplifiés,
- Disponibilité 100 % du canal asynchrone d'identification,
- Souplesse d'organisation des canaux : canal asynchrone et canaux synchrones, par l'utilisation des sous-canaux définis dans le préambule,
- Réduction de la charge des récepteurs "texte" et des traitements de message par la prédésignation dans le préambule,
- Forme du message : préambule et le texte d'un taux de remplissage suffisamment faible pour admettre émission et réception simultanées entrelacées et suffisamment dense pour avoir le message relativement court permettant le temps d'identification court nécessaire pour la configuration air-air et pour le champ de bataille,
- Discrétion des émissions est optimisée :
  - . le préambule à bande large est plus discret que le préambule à bande étroite,
  - . le texte à bande étroite est inchangé mais il est émis avec les sauts de fréquences à chaque impulsion.
- Forme du message avec le préambule B définissant par le mot "MODE" le type de message (ou le sous-canal) et par le mot "PREDESIGNATION" le correspondant ou la zone intéressée,
  - . assure une disponibilité élevée, sans une organisation rigide,
  - . réduit sensiblement la charge de traitement des textes par l'élimination préalable des textes non utiles.

- Entrée au réseau :

  . utilisation du canal asynchrone réduit considérablement le temps d'entrée au réseau et simplifie la procédure.

- Localisation par les messages P

  . la capacité plus grande du terminal SINTAC-3, de même que la possibilité de réception des messages entrelacés permet :

    - soit d'augmenter la périodicité de réception du message P et augmenter la précision de localisation, qui pourra être utile dans la fonction d'anticollision,
    - soit de réduire la périodicité des émissions ce qui réduit, pour la même période de renouvellement de réception, la charge du réseau.

- Facilité d'utilisation du système

  . organisation des canaux simplifiée grâce à la forme du message qui permet émission et réception simultanées entrelacées.
  C'est très important pour la transmission par relais où le canal unique à plusieurs sous-canaux peut être utilisé sans organisation mutuelle et avec une disponibilité de 100 %.

- Compatibilité avec IFF/SSR actuel et avec TACAN/DME

  satisfaisante grâce à l'émission à bande large du seul préambule qui reste suffisamment court pour ne pas brouiller ni l'IFF ni le TACAN.

- Interopérabilité avec le JTIDS (SINTAC-2) peut être assurée dans la période de transition,

  . soit que le JTIDS (SINTAC-2) soit complété par des circuits d'acquisition à bande large et le logiciel complété en conséquence,

  . soit que le SINTAC-3 soit muni des circuits d'acquisition à bande étroite (pour la classe 2, 2 récepteurs par antenne) et également le logiciel complété en conséquence.

- Fonctionnement temps de paix, temps de guerre

  En temps de paix, seule l'acquisition en bande étroite est utilisée

  . le programme des canaux à bande large se déroule d'une façon continue mais il n'est pas utilisé -avec seule exception pour l'identification par Q et R- dont la charge est très faible.

  . ceci rend disponible le système à passer en fonctionnement de temps de guerre instantanément sans aucun préavis, seule contrainte est celle de la responsabilité opérationnelle.

  . cette procédure assure le maximum de sécurité et de discrétion du système en temps de paix.

ADDENDUM : COMPATIBILITE DES SYSTEMES DANS LA BANDE L

Depuis la rédaction du texte de cette conférence, nous avons examiné le problème de compatibilité entre les différents systèmes utilisés en bande L :

- systèmes militaires existants : TACAN et IFF MK 10,
- systèmes civils en développement : DABS,
- systèmes militaires futurs en étude et de développement :

  . NIS bande L,
  . JTIDS - TDMA et DTDMA
  . SINTAC-3.

Nous présentons ici la matrice résumant les résultats obtenus dans les configurations opérationnelles examinées qui nous semblent significatifs et qui incitent à des examens plus approfondis.

On peut constater que :

1 - Les systèmes existants : le TACAN et l'IFF MK 10 (12) sont compatibles avec d'autres systèmes mais leurs performances ou leurs protections contre le brouillage ne sont pas suffisantes et c'est la raison pour laquelle il faut chercher de nouveaux systèmes.

2 - Le DABS est compatible aussi avec les autres systèmes. Il pourrait cependant avoir des problèmes avec le NIS bande L suivant la conception et l'utilisation de ce dernier. Il s'agit là d'un système civil ne répondant pas aux besoins militaires ni de sécurité ni de protection contre le brouillage pour lequel il ne se pose que le problème de compatibilité.

3 - Le NIS bande L est par contre envisagé pour assurer un haut niveau de secret et de protection contre le brouillage. Le produit Bande x Temps (BxT) des impulsions est élevé et en fonctionnement autonome, sa charge en communication apparait incompatible avec le JTIDS TDMA. Il s'agit de blocage (du temps mort) des récepteurs de tous les systèmes en bande L, pendant l'émission de n'importe lequel d'entre eux.
Avec les messages compacts relativement longs du NIS, le brouillage du JTIDS TDMA par paquets dépasse la capacité de correction d'erreurs du JTIDS et les messages brouillés sont perdus.
Dans le cas du JTIDS DTDMA les messages sont distribués et le brouillage par paquet est limité à un nombre faible de symboles que le code de correction d'erreur peut reconstituer, d'où la compatibilité indiquée entre le NIS bande L et le JTIDS DTDMA.

4 - Le JTIDS TDMA comme le JTIDS DTDMA peut être utilisé pour l'identification, qu'il peut remplir de différentes façons : messages P, identification indirecte et par question et réponse mais dans ce dernier cas, ils n'ont pas toutes les performances exigées par le NIS comme c'est indiqué dans le tableau. Il est indispensable de les compléter dans un certain nombre de configurations par un système spécialisé, répondant aux spécifications du NIS, l'ensemble restant compatible.

Un tel système côté NIS pourrait être le NIS bande S compatible aussi bien avec le JTIDS TDMA que DTDMA, mais il pose des problèmes très difficiles dans la période de transition à cause du besoin d'antennes supplémentaires en bande S. De ce point de vue, il y a tout intérêt à utiliser la bande L pour les deux systèmes : le MIDS et le NIS.

5 - La solution peut être recherchée dans la combinaison, JTIDS DTDMA + NIS bande L. Les deux systèmes sont compatibles mais certaines performances, hachurées dans le tableau, sont assez tangentes. Il y a intérêt à chercher à optimiser cet ensemble.

6 - Le SINTAC-3 utilise les messages distribués pour l'ensemble des fonctions aussi bien du NIS, l'identification par question et réponse, que le MIDS. Le temps de blocage pendant l'émission est court distribué aléatoirement de la sorte qu'on ne perde jamais plus d'un symbole à la fois et la perte totale dans un message reste faible compatible avec le code de correction d'erreur, le taux de perte du message est très faible, parfaitement compatible avec les besoins opérationnels aussi bien du NIS que du MIDS.
Les messages du SINTAC-3, bien que distribués sont suffisamment courts pour que l'entrelacement des messages soit rare et s'il arrive, il ne dépasse qu'exceptionnellement 2 à 3, ce qui simplifie sensiblement la gestion du système.

Le SINTAC-3 est le système de la catégorie du système global : NIS bande L + JTIDS DTDMA mais optimisé :

- le message est distribué aussi bien pour le NIS que pour le MIDS d'où la réduction du brouillage par paquet et de la perte de messages ce qui rend compatible le NIS et le MIDS dans la même bande (bande L),
- le préambule est court grâce à la modulation à très large bande ce qui réduit la longueur du message et le nombre de messages entrelacés et qui simplifie la gestion du système,
- le préambule B est utilisé pour la présélection du sous-canal ("MODE") et du porteur intéressé par le message ("PREDESIGNATION"), ce qui soulage considérablement le traitement du texte (une proportion très faible étant destinée au porteur).
- le décodage du préambule est du type passif (on détecte le code attendu) comme dans l'identification MK 10 (mode A du SSR) ce qui donne un pouvoir de séparation élevé (théoriquement 1 chip) aussi bien à l'interrogation qu'à la réponse.
  Il est possible de décoder un grand nombre de messages entrelacés ; la limite étant donnée par l'étouffement des messages et le nombre de circuits prévus pour le décodage.
- dans la période de transition cependant, il peut être demandé au SINTAC à l'interrogation d'envoyer le message de prédésignation en coordonnées directives relatives par rapport à l'interrogateur. Leur décodage ne peut être qu'actif (décoder l'information envoyée, comme dans le mode C du SSR). Dans ce cas, avec la modulation PPM (utilisée dans le SINTAC-3) le pouvoir de séparation diminue (passe à la longueur de l'impulsion) et le nombre d'entrelacements acceptables ne dépasse pas 2 à 3 :
  - ceci pourrait être tangent ou insuffisant, côté interrogateur,
  - côté réponse, il n'y a pas de problème, le décodage est passif et le pouvoir de séparation suffisant permet un grand nombre de messages entrelacés.

Pour éliminer cette objection au SINTAC-3, nous avons envisagé une nouvelle version maintenant le pouvoir de séparation élevé, égal à un chip, même en cas de décodage actif du préambule B.

C'est une possibilité qui augmente encore la souplesse d'utilisation du SINTAC-3 mais qui complique légèrement le récepteur et ne sera utilisée que si effectivement elle se justifie opérationnellement.

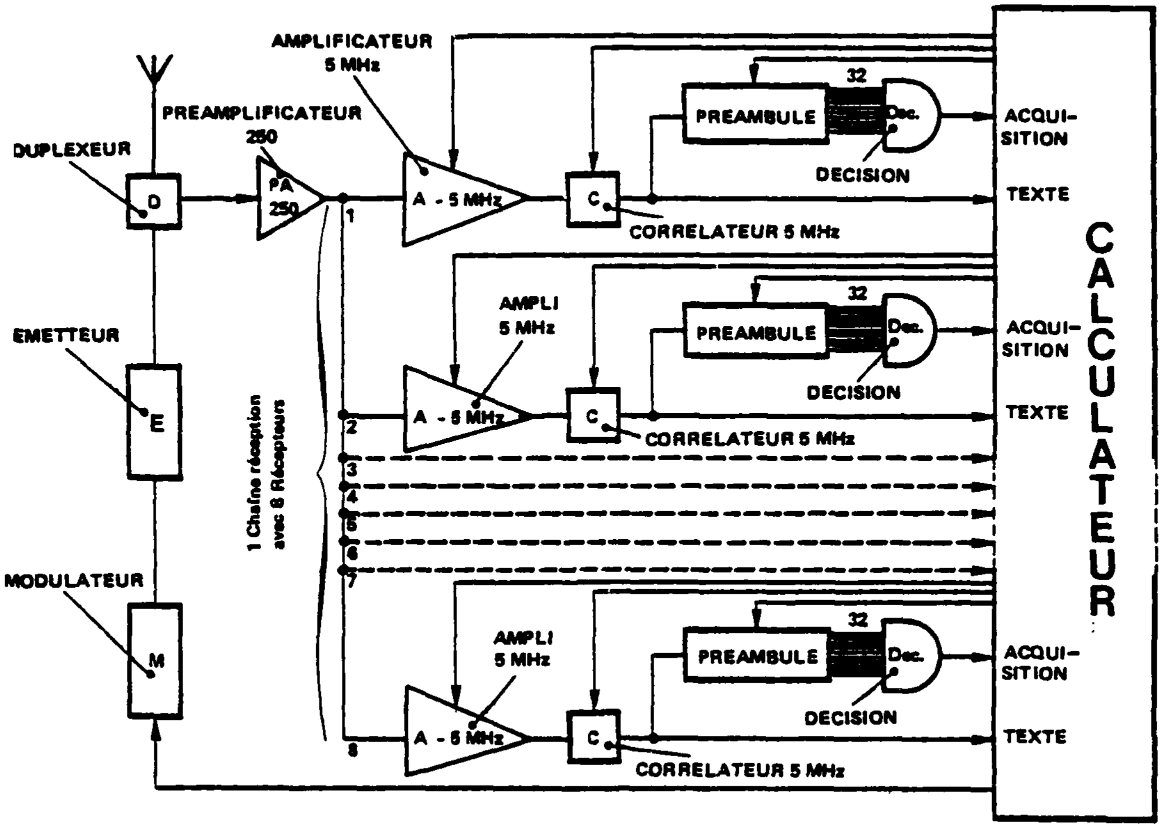
CONCLUSION

1 - Cette étude de compatibilité des systèmes en bande L indique que seuls deux systèmes sont susceptibles de répondre simultanément dans un porteur aux besoins du NIS et du MIDS :

1. NIS bande L + JTIDS DTDMA
2. SINTAC-3 V-2 ou si nécessaire V-3.

Le SINTAC-3 nous apparait comme la version optimisée du premier. Mais dans les deux cas, une étude plus détaillée du système est nécessaire.

2 - De cette étude apparait aussi qu'il est indispensable d'examiner la compatibilité entre tous les systèmes utilisés simultanément dans un porteur et de les définir compte tenu de cette compatibilité.
Ceci peut conduire aussi bien à l'adaptation des systèmes qu'à une modification de spécialisation des systèmes en tenant compte de leur coexistence.
Cette étude globale pourrait concerner non seulement la compatibilité des systèmes mais aussi leur complémentarité et leur redondance et répondre ainsi au thème de ce symposium : "LA TECHNOLOGIE DES SYSTEMES HAUTEMENT INTEGRES".
Mais cette intégration devrait s'étendre à l'étude des différents systèmes dans l'esprit de leur intégration ; elle doit donc concerner aussi le concept des senseurs et ne pas se limiter seulement à leur utilisation intégrée.
Il y a là une optimisation à faire aussi bien du point de vue de la sécurité que de la disponibilité et du meilleur rapport coût/efficacité du système durant la vie du porteur.



Terminal SINTAC 3 de capacité équivalente

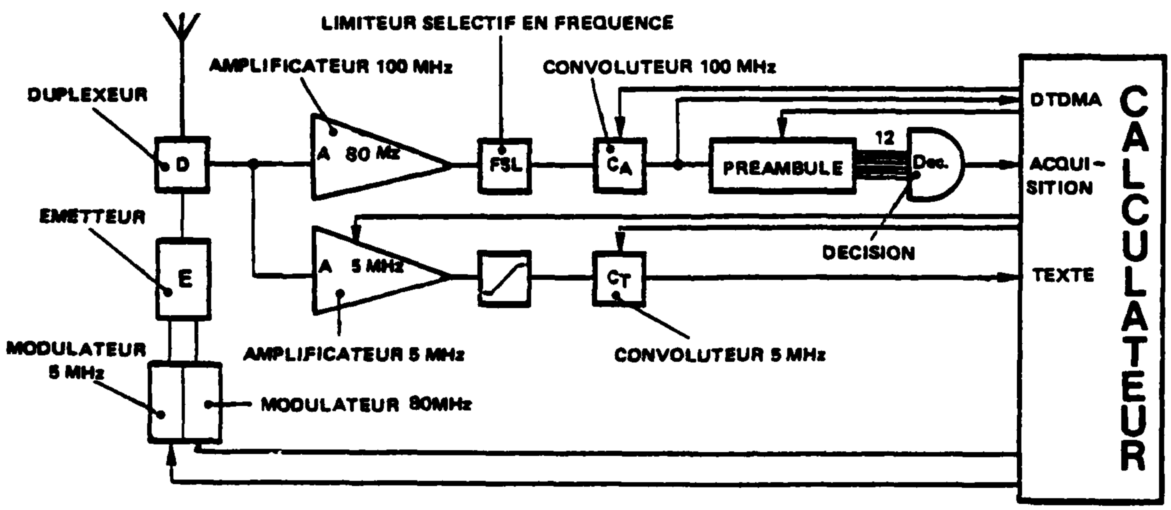


Fig.1 Complexité relative des terminaux JTIDS/SINTAC 2 et SINTAC 3

PROTECTION CONTRE LE BROUILLAGE EN dB		
BANDE MODULATION (MHz) \ BANDE DE BROUILLAGE (MHz)	5	100
5	6 dB	19 dB
100	19 dB	19 dB

Fig. 2b   Fig. 2a

Figure 2a

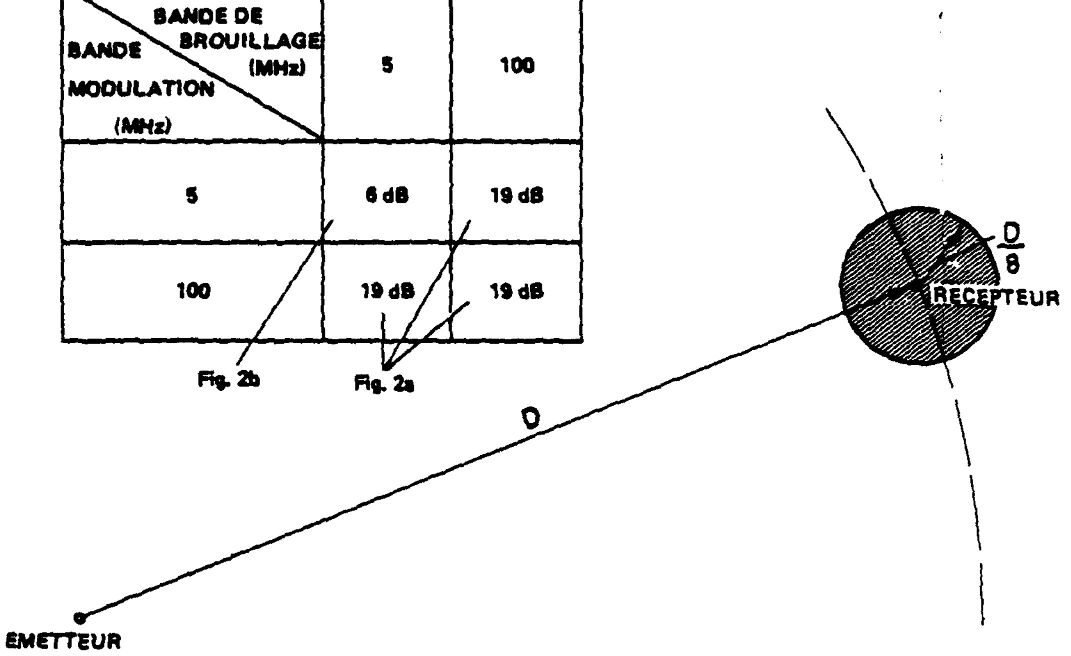


Figure 2b

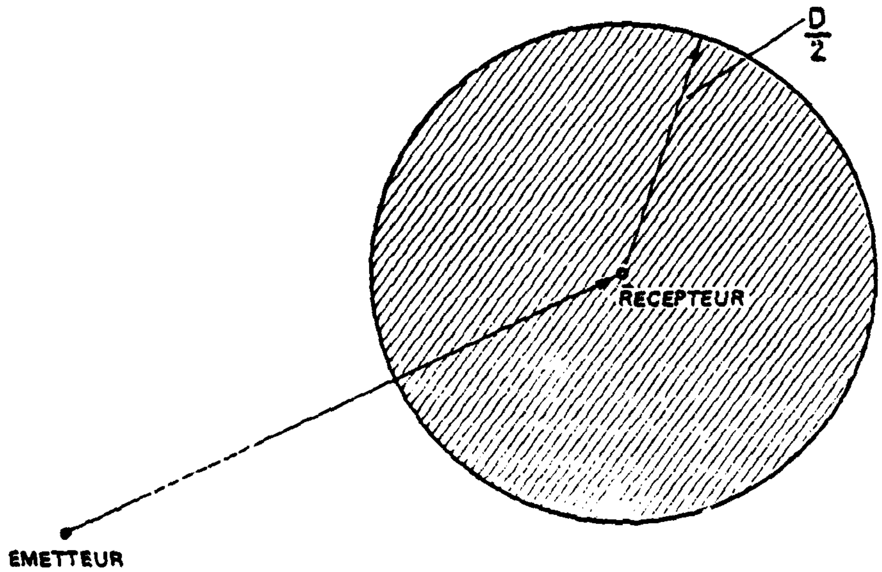


Fig.2 Distance de brouillage ennemi

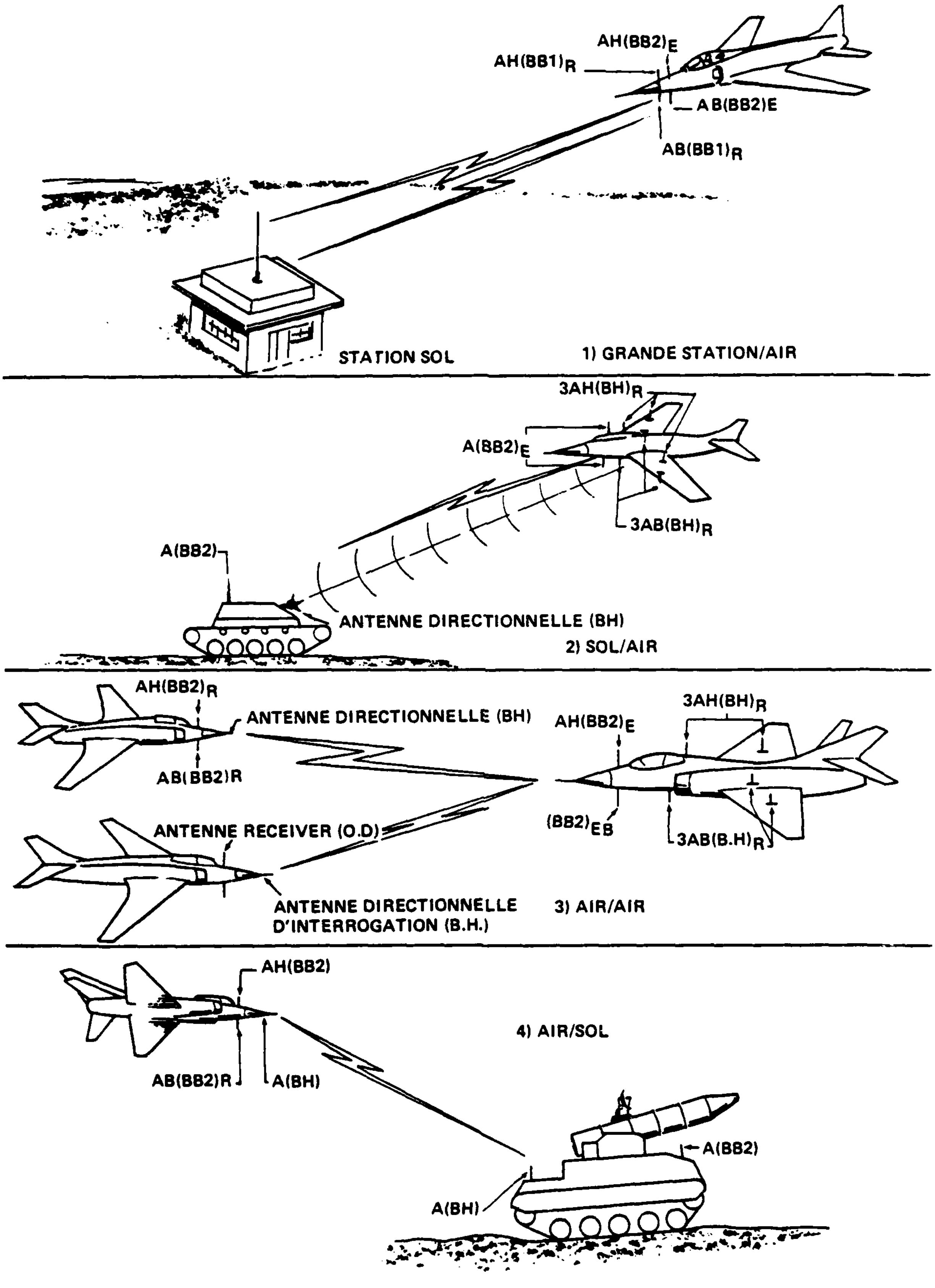
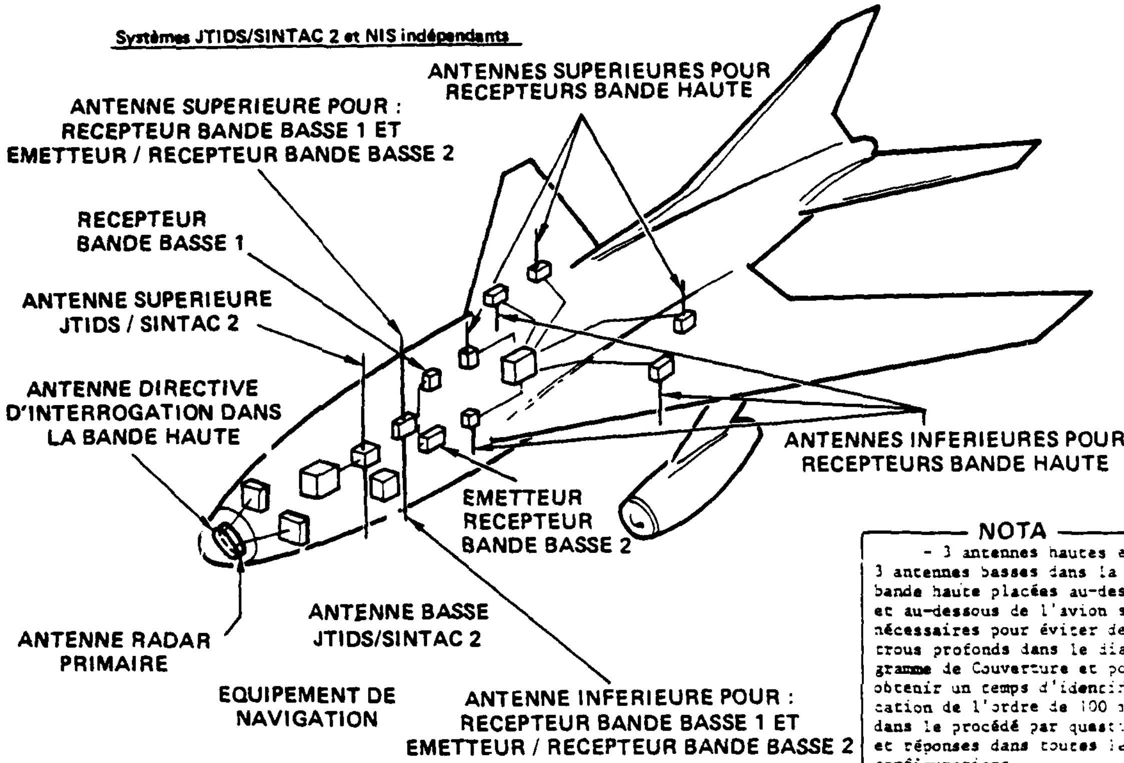


Fig.3(a) Configurations d'identification

Systèmes JTIDS/SINTAC 2 et NIS indépendants



NOTA

- 3 antennes hautes et 3 antennes basses dans la bande haute placées au-dessus et au-dessous de l'avion sont nécessaires pour éviter des trous profonds dans le diagramme de Couverture et pour obtenir un temps d'identification de l'ordre de 100 ms dans le procédé par questions et réponses dans toutes les configurations.

SINTAC 3 incluant les fonctions de NIS et le système de navigation autonome

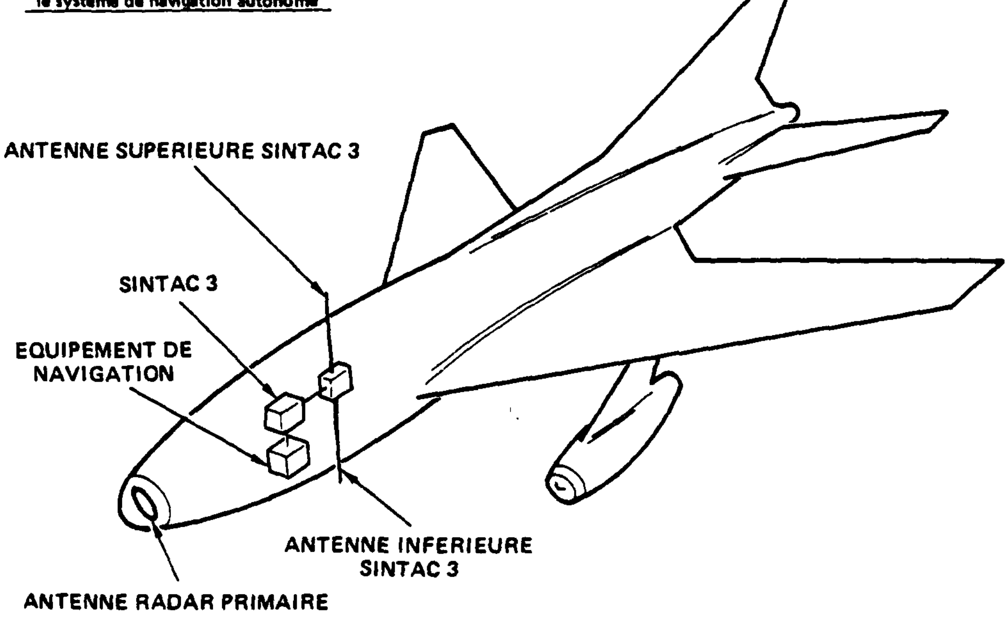


Fig.3(b) Equipements avions du MIDS et du NIS

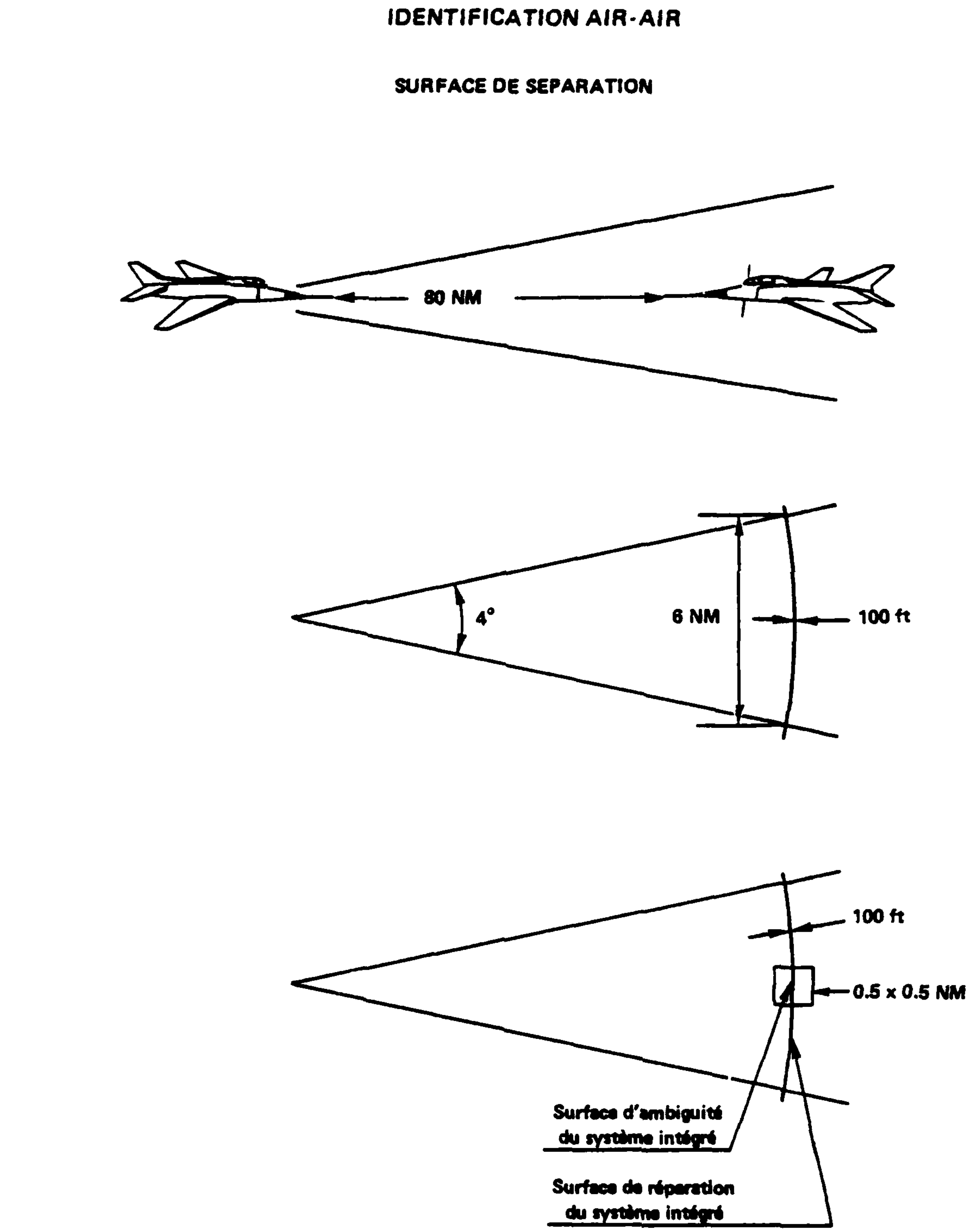
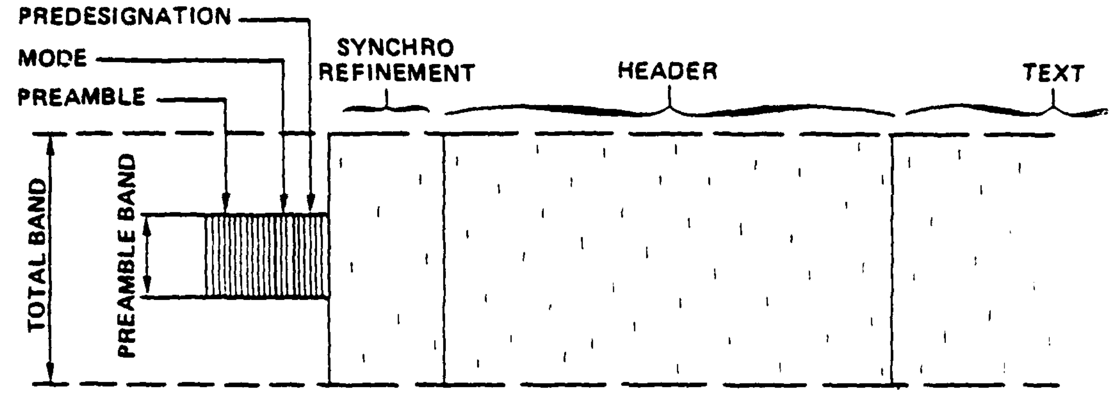


Fig.4 Surface de separation dans les deux modes de designation *directif* (θ, $\Delta\theta$) et par position absolue (X, Y; Δx, Δy)

VERSION 2 : INTERNAL DESIGNATION



VERSION 2 : EXTERNAL DESIGNATION

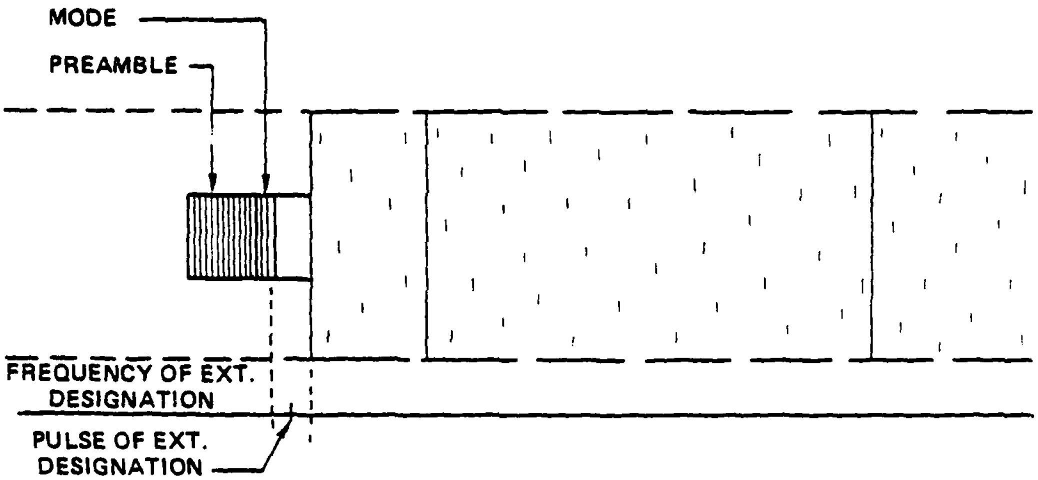


Fig.5 Message format

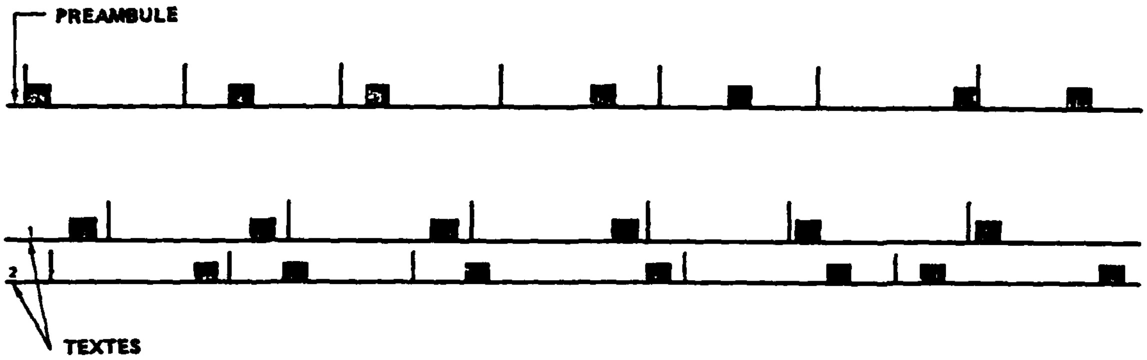


Fig.6 Brouillage du préambule par le texte

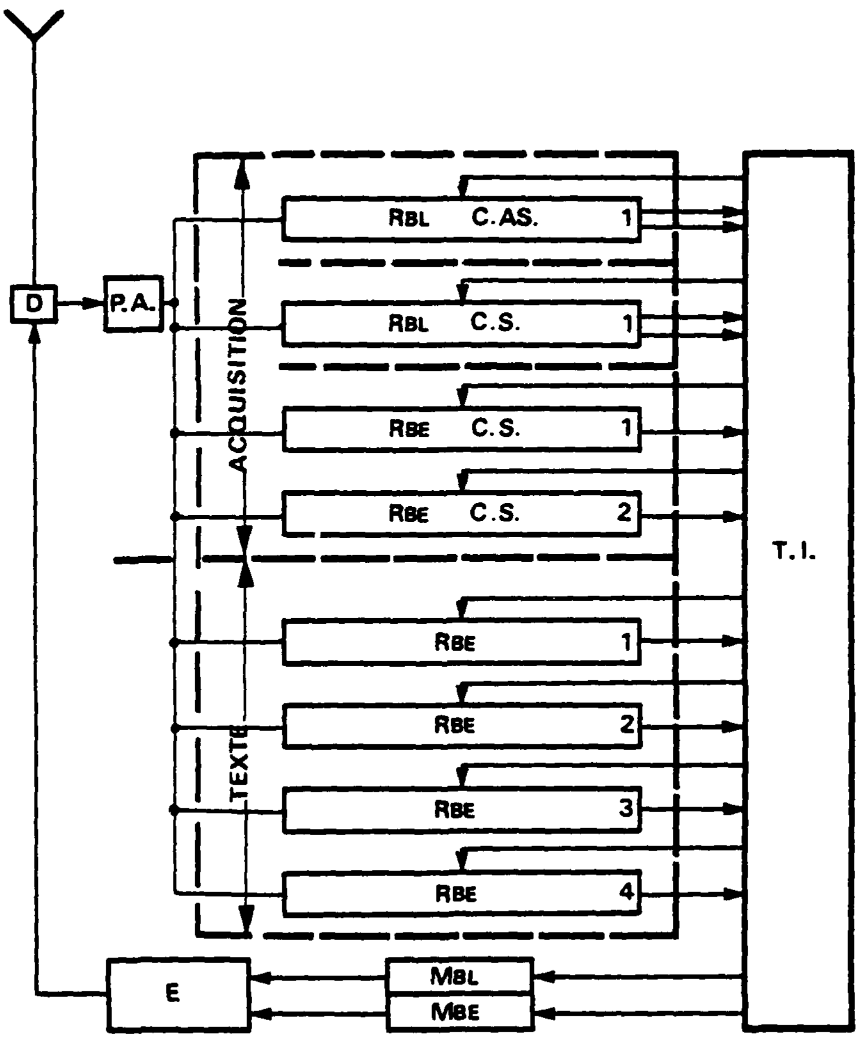


Fig.7 Terminal complet

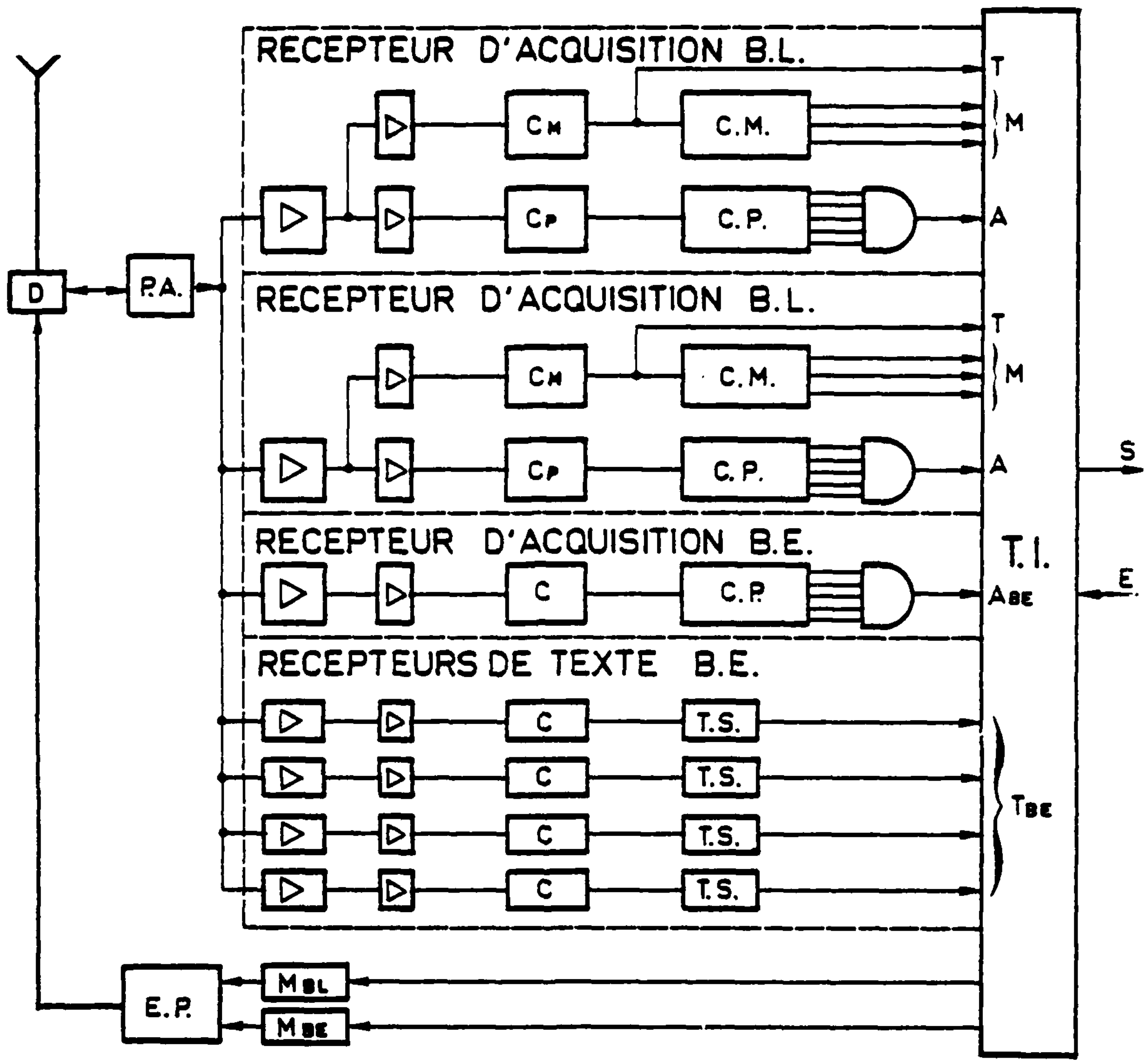


Fig.8 Terminal A2

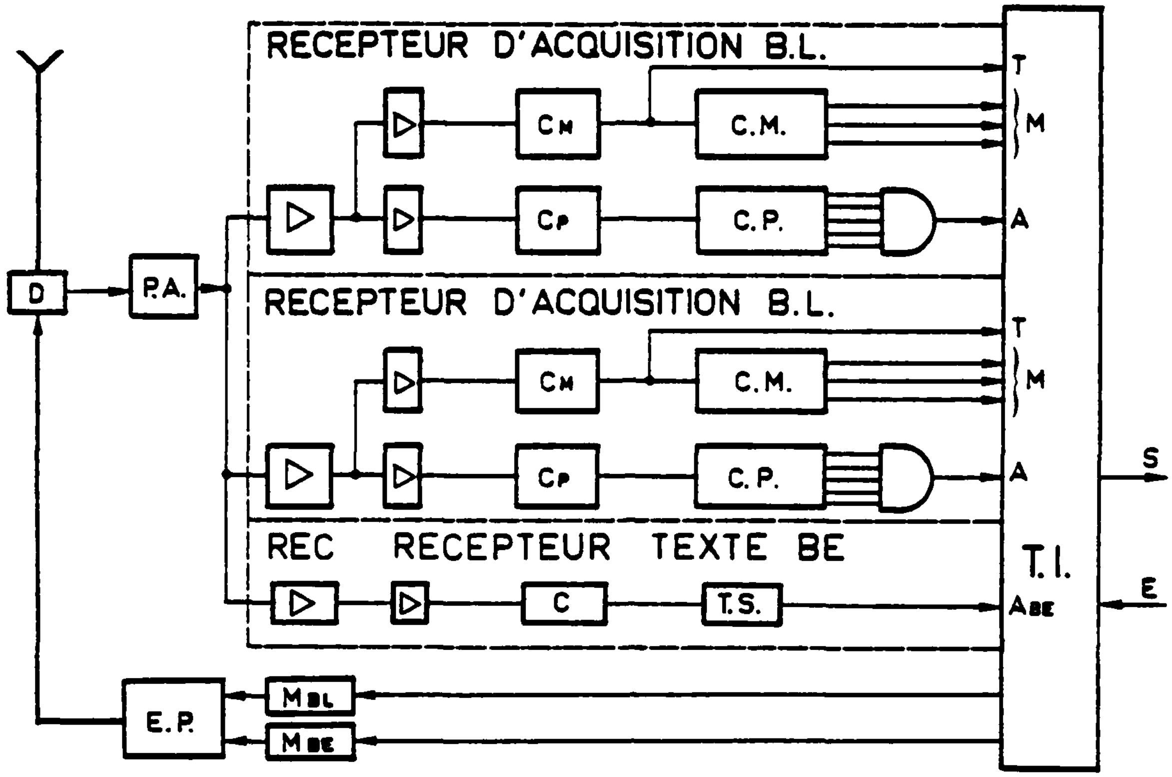


Fig.9 Terminal B4

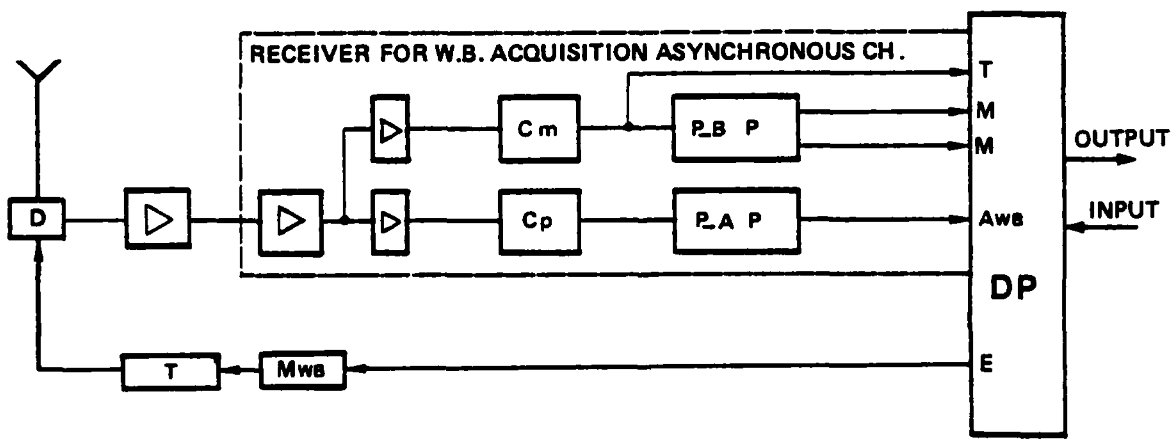


Fig.10 Terminal B6

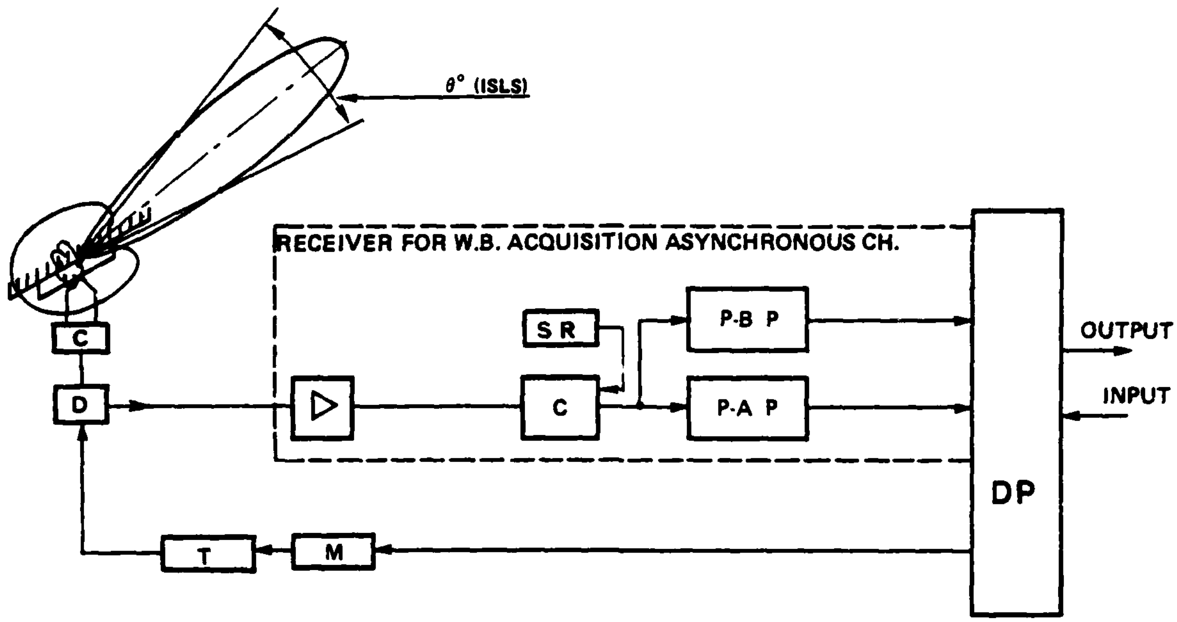
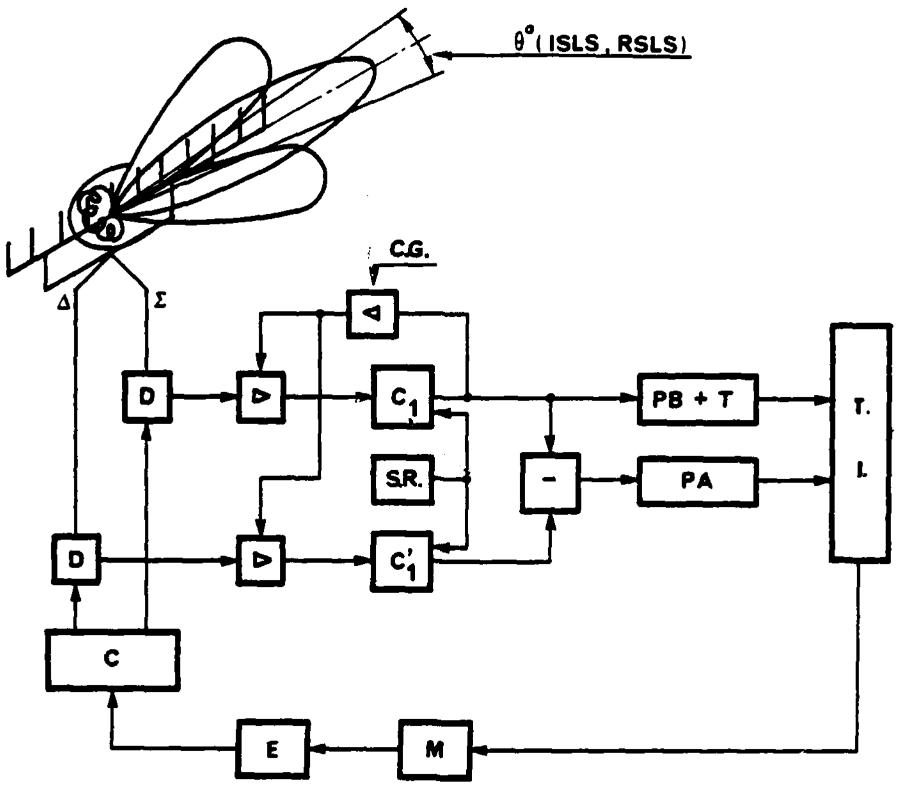
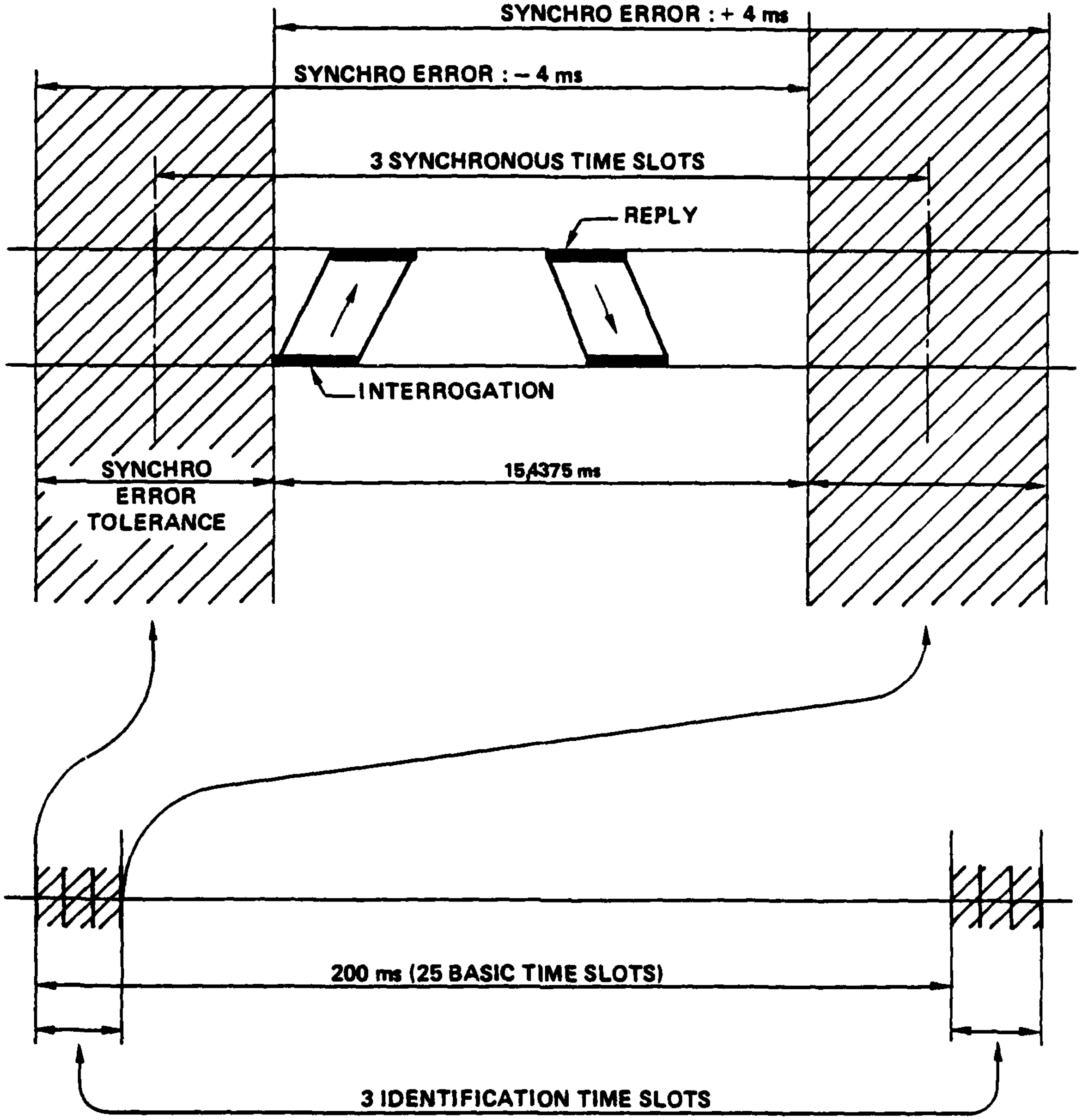


Fig.11 SINTAC 3 Terminal B7 For utilization with directional antenna and ISLS



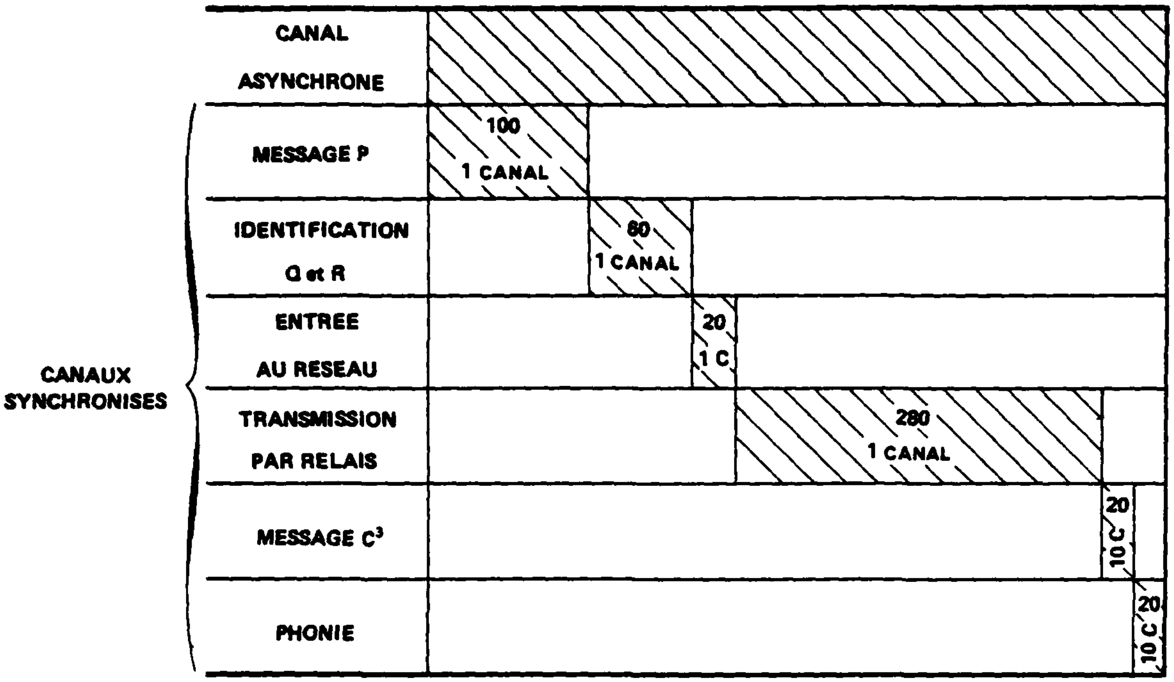
(C.G.): Commande de gain des 2 amplis par la voie Σ
(S.R.): Signal de référence commun aux 2 convoluteurs
(C_1, C'_1): Les 2 convoluteurs identiques montés sur le même substrat

Fig.12 SINTAC 3 – B7 (RSLS) – Pour utilisation avec antenne directive et ISLS + RSLS



SYNCHRONOUS IDENTIFICATION CHANNEL ORGANIZATION

Fig.13 Identification – Canal synchrone



Il s'agit du nombre de sous-recurrences de longueur de un quart de recurrence de base utilisées pour l'émission-reception du préambule A.

Fig.14 Organisation des canaux – Proportion de temps alloue à chaque canal

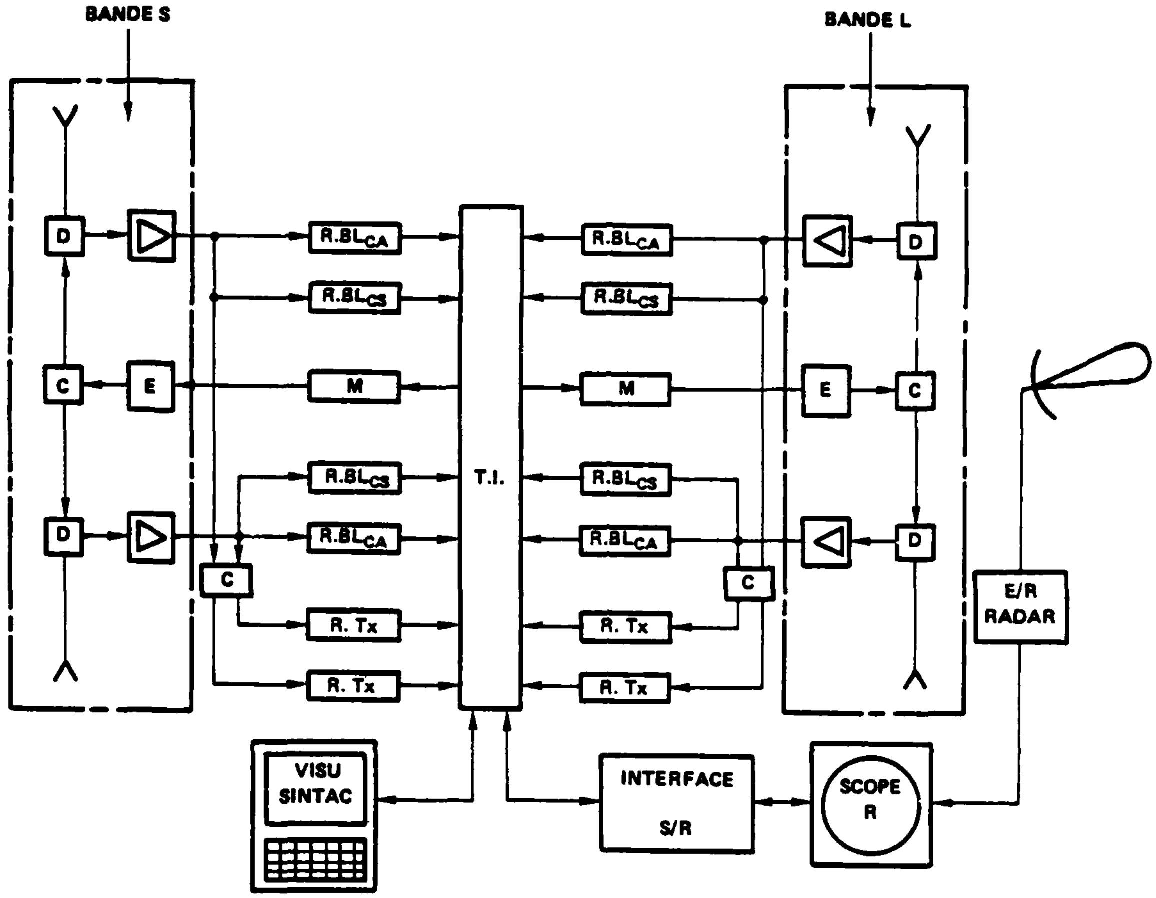
– SECURITE DE FONCTIONNEMENT

– BROUILLAGE PAR ENNEMI

	SYSTEMES	BROUILLEUR	RAPPORT SYSTEME/BROUILLEUR
1	JTIDS BL NIS BS	BROUILLEUR BL BROUILLEUR BL	2/2
2	JTIDS BL NIS BL	BROUILLEUR BL BROUILLEUR BL	2/1
3	SINTAC 3 { MIDS + NIS } BL	BROUILLEUR BL	1/1

BL = Bande L BS = Bande S

Fig.14(a) Système multifonctions a deux bandes: bande L et bande S



- LES RECEPTEURS BANDE LARGE CANAL SYNCHRONE ET CANAL ASYNCHRONE SONT INTERCHANGEABLES
- LES RECEPTEURS DE TEXTE SONT INTERCHANGEABLES
- SEULS LES ETAGES R.F. SONT PARTICULIERS EN BANDE S ET EN BANDE L
- CE TERMINAL ASSURE TOUTES LES FONCTIONS DE COMMUNICATION, DE NAVIGATION ET D'IDENTIFICATION PAR DIFFUSION ET PAR Q ET R EN BANDE L ET EN BANDE S SUIVANT L'ORGANISATION DES CANAUX ET LA SECURITE DEMANDEE.

Fig.15 Schema synoptique du terminal a deux bandes: L et S

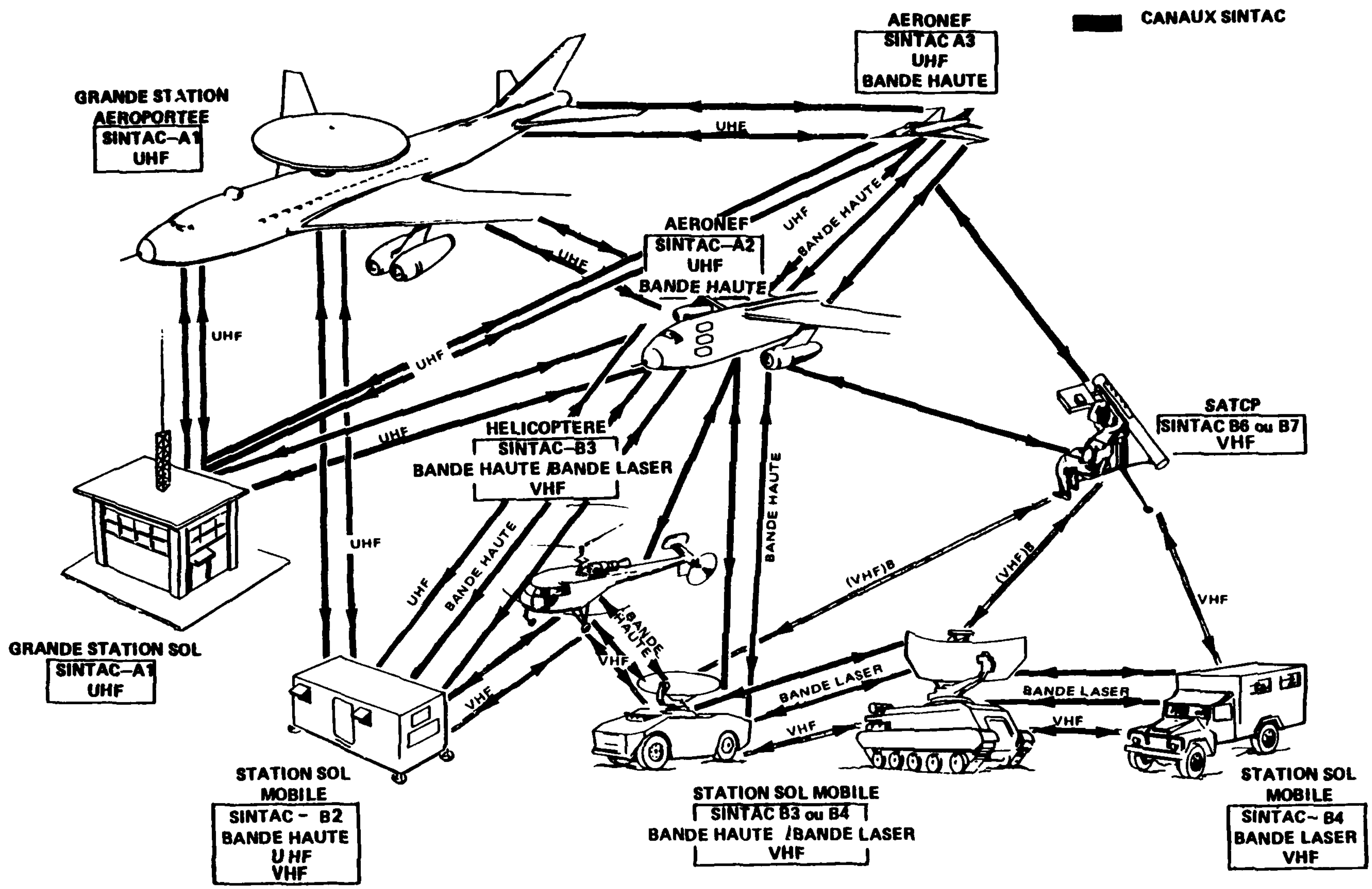


Fig.16 Types de liaisons suivant le porteur

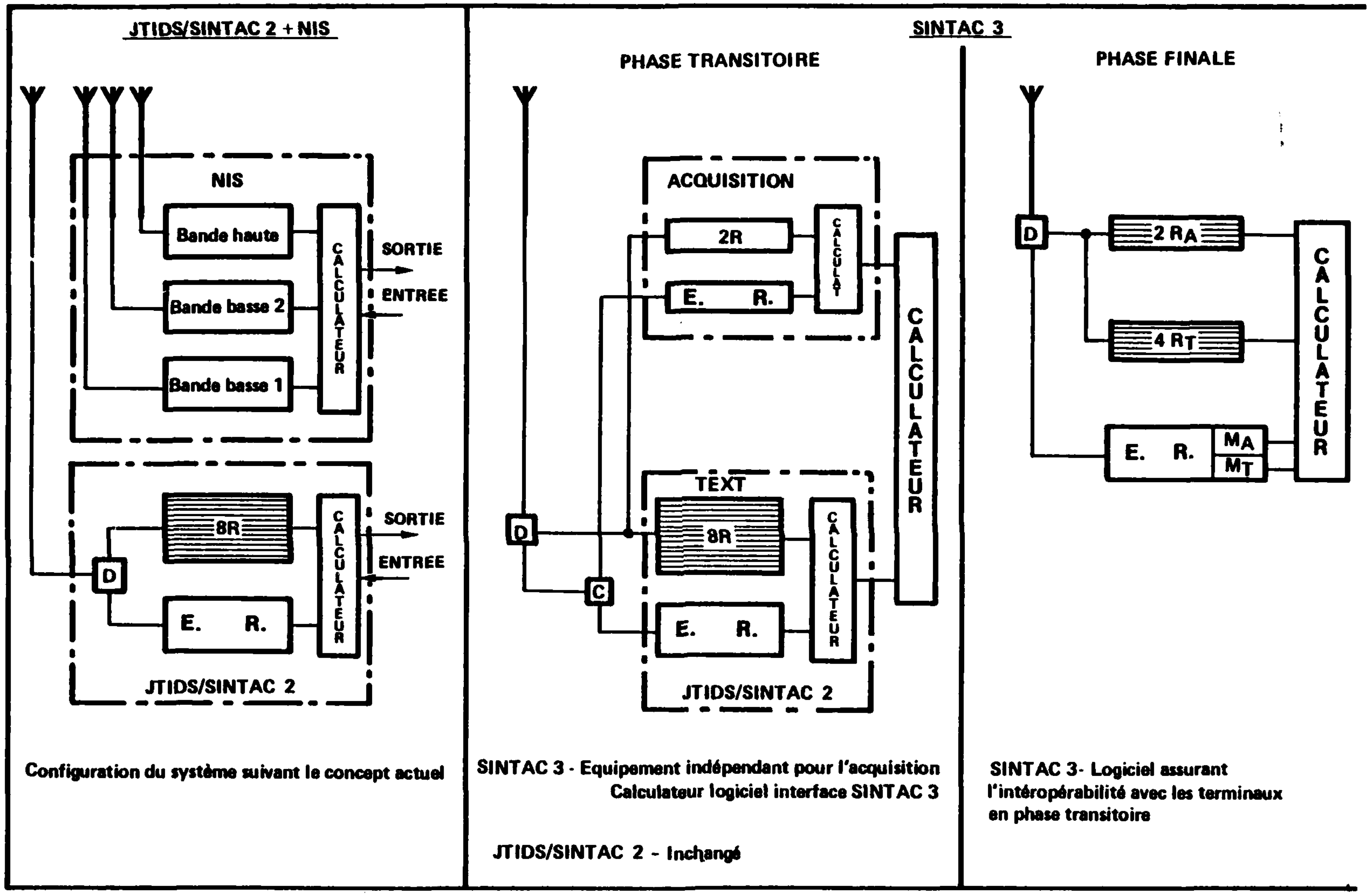


Fig.17 Configurations du terminal

SYSTEME	COMPATIBILITE								PERFORMANCES	
RECEPTION / EMISSION	TACAN	IFF MK10	DABS	NIS bande L	JTIDS TDMA	JTIDS DTDMA	SINTAC - 3 V - 2	SINTAC - 3 V - 3	NIS	MIDS
TACAN	1	1	1	1	1	1	1	1	☒	■
IFF MK10	1	1	1	1	1	1	1	1	■	☒
DABS	1	1	1	▒	1	1	1	1	☒	☒
NIS bande L	▒	▒	▒	▒	■	1	☒	☒	1	☒
JTIDS TDMA	1	1	1	■	▒	☒	1	1	■	1
JTIDS DTDMA	1	1	1	1	☒	1	1	1	■	1
SINTAC-3 V - 2	1	1	1	☒	1	1	▒	☒	▒	1
SINTAC-3 V - 3	1	1	1	☒	1	1	☒	1	1	1
NIS bande L + JTIDS DTDMA	1	▒	▒	▒	☒	1	☒	☒	1	1

1 PERFORMANCES VALABLES, COMPATIBLES

■ INSUFFISANT, INCOMPATIBLE

▒ PERFORMANCES, COMPATIBILITE TANGENTE

☒ NE SONT PAS UTILISES SIMULTANEMENT OU N'EST PAS UTILISE POUR CETTE FONCTION

Compatibilité entre l'ensemble des systèmes dans la bande L

S I N T A C - 3

by

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FRANCE

1 - IMPROVEMENT TO THE CURRENT SYSTEM JTIDS/SINTAC-2

After a careful analysis of the JTIDS/SINTAC-2 performances and the NIS operational requirements, we came to the conclusion that the following improvements have to be added to JTIDS/SINTAC-2 terminals.

- MIDS side
  . simplify the acquisition circuits,
  . increase terminal capacity,
  . increase terminal utilization versatility with different message formats -compact or distributed,
  . maintain jamming protection for simplified terminals,
  . allow terminal simplification for small platforms.

- NIS side
  . use of the asynchronous channel for identification by Q and A,
  . have near to 100% availability for identification by Q and A,
  . be able to simplify the multifunction terminal to cover only the identification by Q and A function, if required, without diminushing the terminal, by keeping it interoperable with all the multifunction terminals.

These characteristics are added to the existing characteristics on the SINTAC-1 and SINTAC-2 for ease of installation on platforms, as the internal designation means it is no longer necessary to use the directional antenna for interrogation.

But that does not exclude the use of directional antenna, either for designation or only to increase the protection against jammers if necessary.

This is an additional capability of the system which :
  . allows IFF function on the platform on which good performances directional antenna cannot be installed,
  . improve in most case performances in separation capability,
  . increase reliability and security,
  . simplify the terminal (volume and cost).

NEW CONCEPT : SINTAC-3

- modification of the acquisition process : based on very large spread spectrum modulation according to the technique proposed for NIS,
- keeping the text modulation process without modification according to the JTIDS/SINTAC-2 technique,
- interoperability with JTIDS/SINTAC-2.

TERMINAL COMPLEXITY (figure 1)

This view showsthe relative complexities of the JTIDS/SINTAC-2 class II terminal and a SINTAC-3 terminal with a similar capacity and a similar anti-jam protection.

JTIDS/SINTAC-2 CLASS II TERMINAL WITH 1 RECEPTION CHAIN :

Using 8 receivers for acquisition -8 are needed for a good protection against jamming of the preamble- but limited to the reception of only one text at a time.

SINTAC-3 EQUIVALENT CAPACITY TERMINAL :

Using 1 reception chain split into two parts :

  . 1 for preamble acquisition,
  . 1 for text reception,

with the same protection against jamming due to the use of a 80 MHz bandwidth for the acquisition preamble.

<u>ANTI-JAM PROTECTION OF THE ASYNCHRONOUS CHANNEL</u> (figure 2)

The figure shows a schema of the protection against jamming of the asynchronous channel and the improvement gained by SINTAC-3 over JTIDS/SINTAC-2 solution.

<u>ABOVE</u> : large spread spectrum modulation and jamming range for all types of jammers, smart jammer included.

<u>BELOW</u> : narrow spread spectrum modulation and jamming range for a smart jammer.

Total energy of jamming is identical in both case*.

<u>HARDWARE COMPLEXITY AND CORRESPONDING INSTALLATION DIFFICULTY</u> (figures 3a et 3b)

- On the figure 3a are represented the operational configurations on which the aircraft can be utilized.
- On the figure 3b above are represented all equipments needed to fulfill the MIDS and NIS requirements on the aircraft :
  - . JTIDS/SINTAC-2 equipment,
    with upper and lower antenna,
  - . low band 1, receiver and
  - . low band 2, transceivers,
    with upper and lower common antennas,
  - . <u>high band</u> :
    - -- receiver with 3 upper and 3 lower antennas to have a good diagram,
    - -- emitter with directional antenna in the nose of the aircraft.
- On the picture below is represented the SINTAC-3 which has the required capability of MIDS and NIS using the information delivered by the navigation system already implemented on the aircraft, but not used by the dedicated systems (descripted above).

  In this system hard and soft arangements are such that the terminal can provide the required reliability.

<u>DISCRIMINATION ACCURACY</u> (figure 4)

The picture shows the advantages of the internal designation over the external designation for identification by Q and A.

RANGE : 80 NM

<u>EXTERNAL DESIGNATION WITH DIRECTIONAL ANTENNA</u> :

- with a beamwidth of 4°
  provides :
  - . lateral discrimination : 6 NM
  - . distance discrimination : 100 ft

<u>INTERNAL DESIGNATION USING A NAVIGATION SYSTEM</u> :

- with an average accuracy of 0.5 NM
  provides :
  - . lateral discrimination : 0.5 NM
  - . distance discrimination : 100 ft

Internal designation provides better discrimination accuracy and its installation and cost advantages are evident.

2 - SINTAC CONCEPT

<u>MESSAGE FORMAT</u> : figures 5 and 6

The message contains two parts :

- the preamble,
- the text.

The preamble has three functions :

- message acquisition,
- subchannel differentiation,
- predesignation of the correspondent(s) by zone, or individually.

---

* The wide band being reduced from 100 MHz to 80 MHz. Here is a reduction of 1 dB in gain approximately.

To perform these functions, the preamble is divided into two parts :

- preamble A, consisting of the acquisition frame,
- preamble B, consisting of two words :
    - . "MODE" defining the subchannel,
    - . "PREDESIGNATION" indicating the applicable zone or providing the individual code of the correspondent.

The text :

- gives the precision synchronization,
- clears up any ambiguities that the predesignation might have left,
- provides the message information.

- the preamble pulses are modulated through a wide band,
- the text pulses are modulated through a relatively narrow band, the same as for the JTIDS or SINTAC-2,
- the preamble and text pulses are transmitted in "pages" with :
    - . low filling rate,
    - . pseudo random (defined by channel) pulse position in the page.

The preamble and text filling rates are designed to :

- allow simultaneous interlaced transmission and reception without major loss, with also allows for possible interlaced messages reception in the same channel,
- to have sufficiently short messages to meet identification by Q and A requirements.

This message format :

- increases terminal availability,
- greatly simplifies channel organization,
- improves the compatibility between the independent systems on the same platform functioning in the same frequency (by the reduction of the mutual packet blanking of the messages during the transmission).

3 - TERMINAL COMPOSITION (figure 7)

A terminal consists of :

- acquisition receivers,
- text receivers,
- the transmitter,
- signal and message processing devices.

- ACQUISITION RECEIVERS :

A complete terminal includes :

- wide band acquisition receives in the asynchronous channel,
- wide band acquisition receives in the synchronous channels, organized on a shared time division basis,
- narrow band acquisition receives in the synchronous channels ; used mainly in peace time, but also during the transition period to assure interoperability with the current system (it can be backed up for this purpose).

- TEXT RECEIVERS :

- these, narrow band frequency hoping receivers, are identical to the receivers in the current system, except for the filling rate which is lower, and for the possibility of interlaced receptions.
- the number of these receivers depends on the terminal class,
- they are programmed only after the acquisition frame pulse is received in one of the available channels,
- they are used per message priority.

- TRANSMITTER :

A common transmitter is used for both types of modulation :

- wide band for the preamble,
- narrow band for the text.

- PROCESSING :

The signal and message processing is adapted to the new modulation, the new message format and the new channel organization.

4 - TERMINAL CLASSES

The SINTAC-3 terminal can be adapted to a very range of stations, from large C^3 stations down through all the intermediate platforms to the very short range man-pad weapon system.

It is regularly used as the multifunction system with omnidirectional antenna, but if required :

- it can be used with a directional antenna and for a reduced number of functions,
- or, if absolutely necessary, it can be used for a single function particularly as a specialized identification system, with the consequent configuration simplification, but with the fundamental characteristic that interoperability is provided among all terminal classes.

For this reason, it can be adapted perfectly well to the transition period when coexistence and compatibility must be provided between current military and civilian classes.

The various terminal classes are distinguished by :
- their transmission power and range,
- message processing capacity,
- technology depending on platform type : land, sea, air.

A certain number of terminals are shown in the pictures 8 to 12.

IDENTIFICATION SECURITY IN THE INTEGRATED MULTIFUNCTION SYSTEM IN RELATION TO THAT OF THE NIS

- The NIS is a specialized system using only the question and answer identification process. In cases of failure the identification is lost.
- the SINTAC used several identification processes and redundant equipment :
  1. Identification by "broad cast" :
     . P messages and indirect identification.
  2. Identification by Q and A :
     . both in the two channels families :
       - asynchronous channel,
       - synchronous channels.

The system normally uses internal designation with the position provided by the navigation system ; but if required the directional antenna is available for this purpose.

- The SINTAC identification capacity is very high :
  . identification has priority,
  . all terminal receivers can be switched instantaneously to identification required,
  . the terminal can respond to interlaced interrogations.

Therefore, concerning :
. redundancy and security,
. instantaneous capability.

the integratedmultifunction system offers advantages over the specialized system :
. the Q and A identification is relatively low as it concerns only unkown carriers
. five groups of three basic time-slots can be reserved in the synchronous channel for the Q and A identification (i.e. 15 time-slots) to meet load requirements. The time slot organization allows a $\pm$ 4 ms synchronization error. Access time is $\leq$ 200 ms.

The figuresbelow show :
. figure 13 : the time organization of the Q and A synchronous identification channel, with the tolerance of synchronization,
. figure 14 : the planned organization of all asynchronous and synchronous channels for various class II equipment.

MULTIFUNCTION SYSTEM - SECURITY OF OPERATION

REDUNDANCY IN SYSTEM,

REDUNDANCY AMONG SYSTEMS

In the SINTAC-3, all MIDS and NIS functions are performed in the L band (962-1215 MHz).

The objections are raised :
. that in the case of failure all functions are lost simultaneously,
. and that a single jammer can jam all the functions.

This comes down to a question of economy involving equipment redundancy and redundancy among systems :
. as far as failures are concerned, the integrated system configuration is such that it assures equipment redundancy under better economic conditions than the specialized system because the same redundancy can provide several functions,
. as far as concerns the jamming of the multifunction system by a single jammer, while two specialized systems in different bands require two jammers, the cost effectiveness ratio should be examined in both cases. These are given on the table below.

It can be seen that the protection of the multifunction system is equivalent to that of two specialized systems in two different bands because the cost of the identification function is marginal ; but the L band NIS alternative is unfavorable from this point of view because, in this case, the cost of the system actually doubles and both systems can be jammed by the same jammer.
The solution which offers the maximum operational effectiveness has to provide redundancy in the multifunction system operating in two different bands e.g. the U.H.F. or S band and L band, as shown in the

figure 15 opposite in which the only equipments specific to a band are the R.F. input and output stages.

In this case, redundancy of the equipments can be optimized ; it does not double the equipment whereas the jammer must be doubled.

The partial redundancy can also be realized between the communication systems particularly for the voice, as shown in the figure 16.

WAR, CRISIS AND PEACE TIME

PEACE TIME

- narrow band receivers only are used,
- narrow band receivers are fitted with a simplified acquisition circuit,
- acquisition process
  - . on a single frequency,
  - . on a single code,
  - . with a P.N. time distribution according to SINTAC-3 preamble organization.

CRISIS AND WAR TIME

- full SINTAC-3 capacity is used,
- wide band acquisition program is permanently running (even in peace time) and available without any notice.

TRANSITION PHASE - CONFIGURATIONS OF TERMINAL (figure 17)

1 - Current solution envisaged

Independent JTIDS/SINTAC-2 and NIS equipments.

2 - Solutions based on SINTAC-3

2.1. Transition phase :

. JTIDS/SINTAC-2 (not modified), and in addition,

. SINTAC-3 : wide band emission-reception on the preamble for acquisition and NIS capability.

2.2. Final phase :

SINTAC-3 complete with MIDS and NIS capabilities.

CONCLUSION - ADVANTAGES OF THE SINTAC-3

- terminal simplification,
- increase in system capability at constant cost,
- jammer protection in the asynchronous channel,
- jammer protection of simplified terminals,
- 100% availability of the asynchronous identification channel,
- flexible channel organization : asynchronous and synchronous channels by using sub channels, indicated in the preamble,
- reduction of the load text receivers and message processing circuits, by use of predesignation in the preamble,
- message format : preamble and text with sufficiently low filling rate to allow simultaneous interlaced transmission and reception, but sufficiently dense to have a relatively short message, thus allowing the short identification time required for the air-air configuration and for the battlefield,
- optimization transmission discretion :
  - . the wide band preamble is more discreet than the narrow band preamble,
  - . the narrow band text remains unchanged, but it transmitted with frequency jumps at each pulse.
- the message format is such that preamble B defines (by the "MODE") the type of message or sub channel and (by the "DESIGNATION") the involved correspondent or zone, this provides :
  - . high availability without a rigid organization,
  - . important reduction of the load of useless text processing.
- entry in net
  - . the use of the asynchronous channel considerably reduces the time for net entry, and also simplifies the procedure.
- localization by P messages
  - . the greater capacity of the SINTAC-3 and the possibility of receiving interlaced messages mean that :
    - either the periodicity of P message reception and the localization precision can be increased, which could be very useful in the anticollision function,
    - or the transmission periodicity could be reduced, thereby reducing for the same period of the renewal of receptions, the network load.
- ease of system utilization
  - . this channel organization is simplified by the message format which allows simultaneous interlaced transmission and reception,
  - . this is very important for identification, but also for voice communication where it is possible to talk and listen simultaneously,
  - . it is also important to relayed transmissions where a single channel with several sub channels can be used without organization, and with 100% availability.

- compatibility with current IFF/SSR and TACAN/SME is satisfactory because only the preamble is transmitted in wide band and it is short enough not to jam the IFF or TACAN
- during the transition period, interoperability with the JTIDS (SINTAC-2) can be ensured by :
  - . either adding the necessary wide band acquisition circuitry and consequent software to the JTIDS (SINTAC-2),
  - . or by adding narrow band acquisition circuitry to the SINTAC-3 (for class 2 with 2 receivers per antenna) with the consequent software.
- peace time, war time operation
  On peace time only the narrow band acquisition is used :
  - . the wide band channel program runs continously but is used only for identification by Q and A, where the load is low.
  - . this means the system is available for instantaneous changeover to war time operation without forewarning, the only constraint being operational responsability.
  - . this procedure ensures maximum system security and discretion in peace time.

ADDENDUM : SYSTEM COMPATIBILITY IN THE L BAND

Since the text of this lecture was prepared, we have examined the problem of compatibility between the various systems used in the L band :

- military systems already existing : TACAN and IFF MK 10,
- civil systems being developped : DABS,
- future military systems being studied and developped :
  - . NIS L BAND,
  - . JTIDS - TDMA and DTDMA,
  - . SINTAC-3

We show here the matrix which recapitulates the results obtained with the operational configurations examined. They seem to us significant and call for more thorough examination.

It can be seen that :

1 - The existing systems, TACAN and IFF MK 10 (12) are compatible with other systems but their performance or protection against jamming is insufficient and that is why new systems must be sought,

2 - The DABS is also compatible with other systems. However, there could be problems with the NIS L BAND which would depend on the design and use of the latter. This is a civil system which does not satisfy military requirements for security or protection against jamming, system for which only the problem of compatibility arises.

3 - The NIS L BAND is considered however as it provides a high level of secrecy and protection against jamming. The Band x Time (BxT) product for the pulses is high and, in independent operation, its communication load appears to be incompatible with the JTIDS TDMA.
It is a question of blanking (dead time) the receivers of all systems in the L BAND during transmission from any one of them.
With the compact, relatively long, messages of the NIS, the jamming of the JTIDS TDMA by packets exceeds the error correction capacity of the JTIDS and the jammed messages are lost.
In the case of the JTIDS DTDMA, the messages are distributed and the jamming by packets is limited to a small number of symbols that the error correction code can reproduce ; this gives the compatibility shown between the NIS L BAND and the JTIDS DTDMA.

4 - The JTIDS TDMA, like the JTIDS DTDMA, can be used for identification, which can be done in various ways : P messages, indirect identification and interrogation and response but, in this last case, the full performance required by the NIS is not obtained as is shown in the table. It is necessary, in a certain number of configurations, to complete this with a specialized system, which complies with the NIS specifications, while the whole remains compatible.
Such a system on the NIS side might be the NIS S BAND, which is compatible with both the JTIDS TDMA and the DTDMA but it causes difficult problems in the transition period because of the need for additional antennas in the S BAND. From this point of view, there is an advantage in using the L BAND for both systems, the MIDS and the NIS.

5 - A solution may be found in the combination, JTIDS DTDMA + NIS L BAND. The two systems are compatible but some of the performance, shown shaded in the table, is dubious. It would be advisable to optimize this assembly.

6 - The SINTAC-3 uses distributed messages for all functions for both the NIS identification by interrogation and response, and the MIDS. The blanking time is short and its distribution is random so that not more than one symbol is ever lost at once and the total loss in a message remains small, which is compatible with the error correction code. The message loss rate is very small, which is fully compatible with the operational requirements of both the NIS and the MIDS. The SINTAC-3 messages although distributed, are sufficiently short for message overlapping to be rare and, if it happens, it seldom exceeds 2 to 3, which simplifies system management considerably.

The SINTAC-3 is in the global system category : NIS L BAND + JTIDS DTDMA optimized :

- the message is distributed both for the NIS and the MIDS. This gives a reduction in the jamming by packets and in messages loss and makes the NIS and MIDS compatible in the same band (L BAND),
- the preamble is short thanks to the very wide band modulation. This reduces the message length and the number of overlapping messages. It simplifies system management.
- preamble B is used for preselection of the sub channel ("MODE") and of the carrier concerned by the message ("PREDESIGNATION"). This lightens text processing considerably (a very small proportion being intended for the carrier).
- decoding of the preamble is of the passive type (the code expected is detected) as in MK 10 identification (mode A of the SSR). This gives a high separating power (theorically 1 chip) both in interrogation and response.
  It is possible to decode a large number of overlapped messages, the limit being fixed by the choking of the messages and the number of circuits provided for decoding.

b- During the transition period however, the SINTAC, on interrogation, may be called on to send the predesignation message in relative directive coordinates with respect to the interrogator. Their decoding can only be active (decode the data sent, as in mode C of the SSR). In this case, with PPM modulation (used in the SINTAC-3) the separating power decreases (to the pulse length) and the number of acceptable overlaps does not exceed 2 to 3.

. this could be just sufficient on the interrogator side,

. on the response side, there is no problem. The decoding is passive and the adequate separating power allows a large number of overlapping messages.

To counter this objection to the SINTAC-3, we have considered a new version which retains the high separating power, equal to one chip, even when preamble B is actively decoded.

This is a possibility which increases even further the flexibility of the SINTAC-3 in use but which complicates the receiver slightly and will only be used of it is really justified operationnaly.

CONCLUSION

1 - This study of compatibility in the L BAND shows that only two systems are likely to satisfy the requirements of the NIS and MIDS simultaneously in a carrier :

1 - NIS L BAND + JTIDS DTDMA

2 - SINTAC-3 V-2 or, if necessary, V-3.

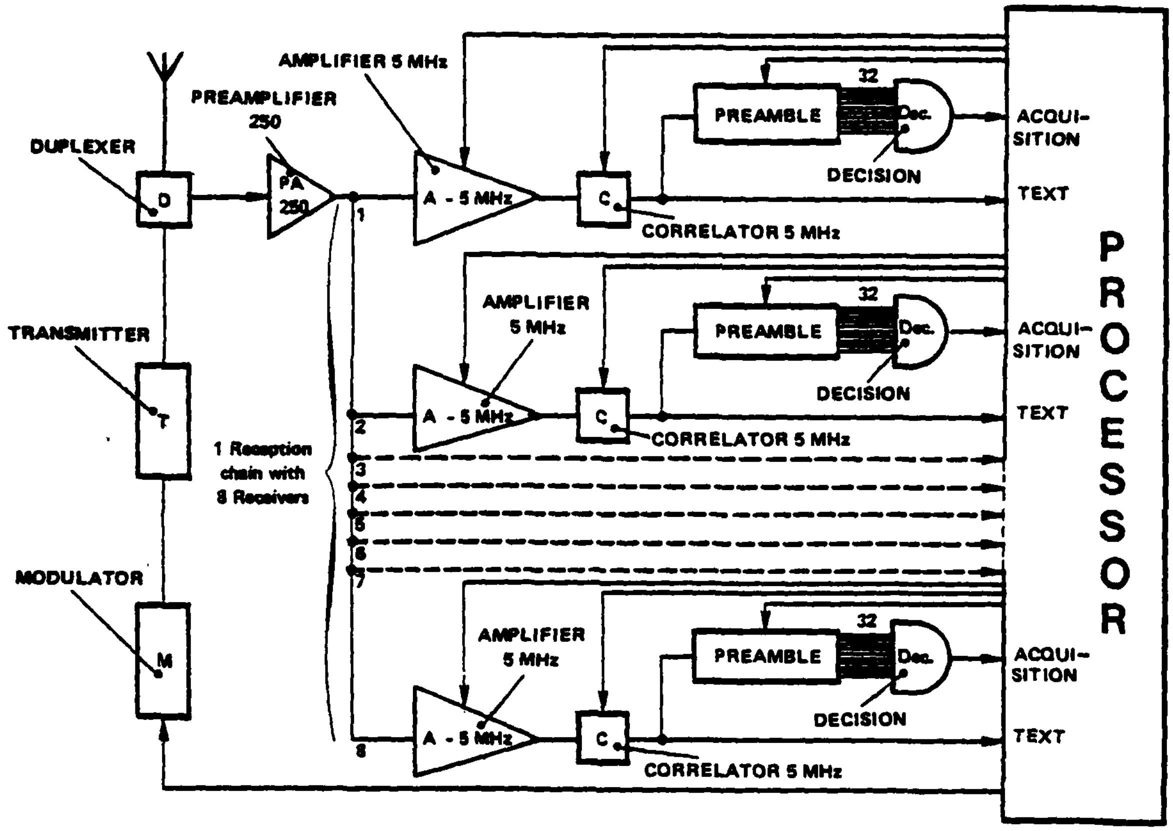
The SINTAC-3 appears as the optimized version of the first one. However, in both cases, a more detailed study of the system is required.

2 - From this study, it appears that it is also necessary to examine the compatibility between all the systems used simultaneously in a carrier and then define them taking this compatibility into account.

This may lead not only to an adaptation of the systems but also to a modification of the system specifications to take their coexistence into account.
This overall study could concern not only the compatibility of the systems but also their complementarity and redundance and cover the theme of this congress "THE TECHNOLOGY OF HIGHLY INTEGRATED SYSTEMS".

This integration should be extented to the study of the various systems in the spirit of their integration. It must therefore also concern the design of the sensors and not limit itself to their integrated use.

There is an optimization to be made in that from the point of view both of security and of availability as well as of the best cost/effectiveness ratio of the system during the life of the carrier.



SINTAC 3 TERMINAL WITH EQUIVALENT CAPACITY

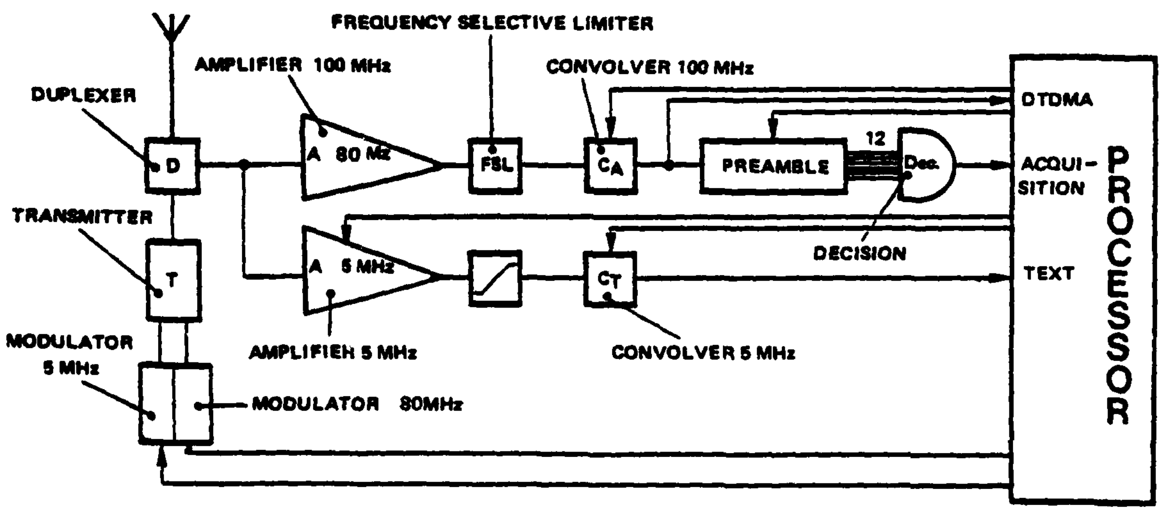


Fig.1 Example of JTIDS/SINTAC 2 terminal

PROTECTION AGAINST JAMMING (dB)		
JAMMING BAND (MHz) / MODULATION BAND (MHz)	5	100
5	6 dB	19 dB
100	19 dB	19 dB

Fig. 2b Fig. 2a

Figure 2a

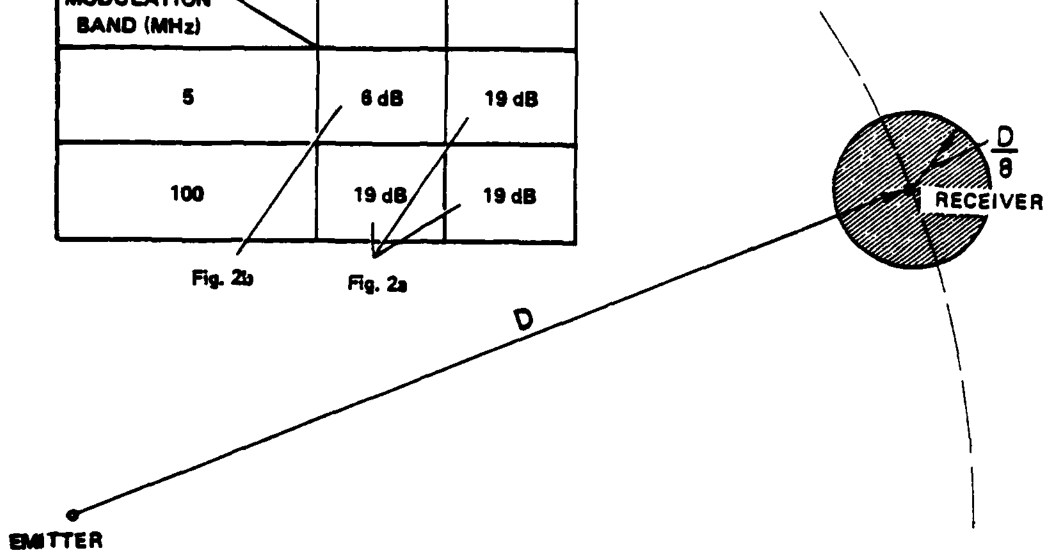


Figure 2b

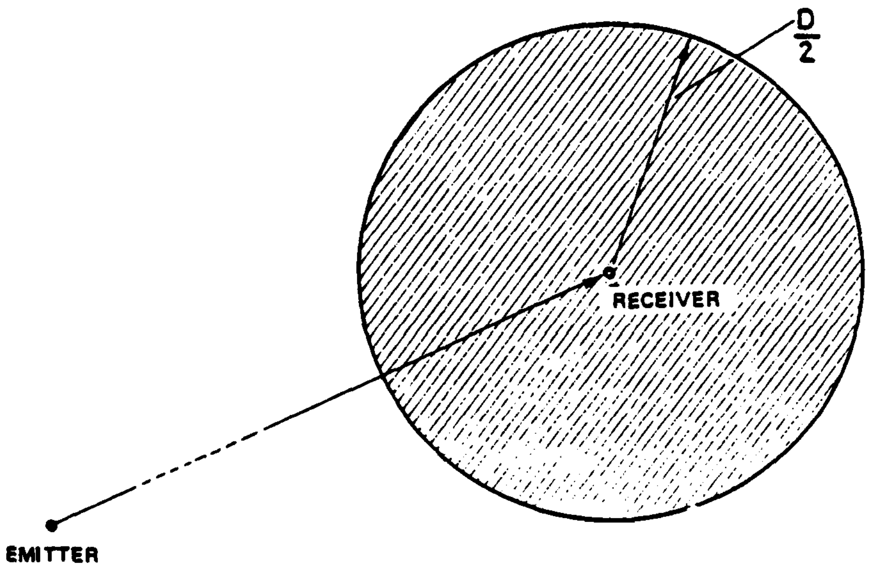


Fig.2 Jamming

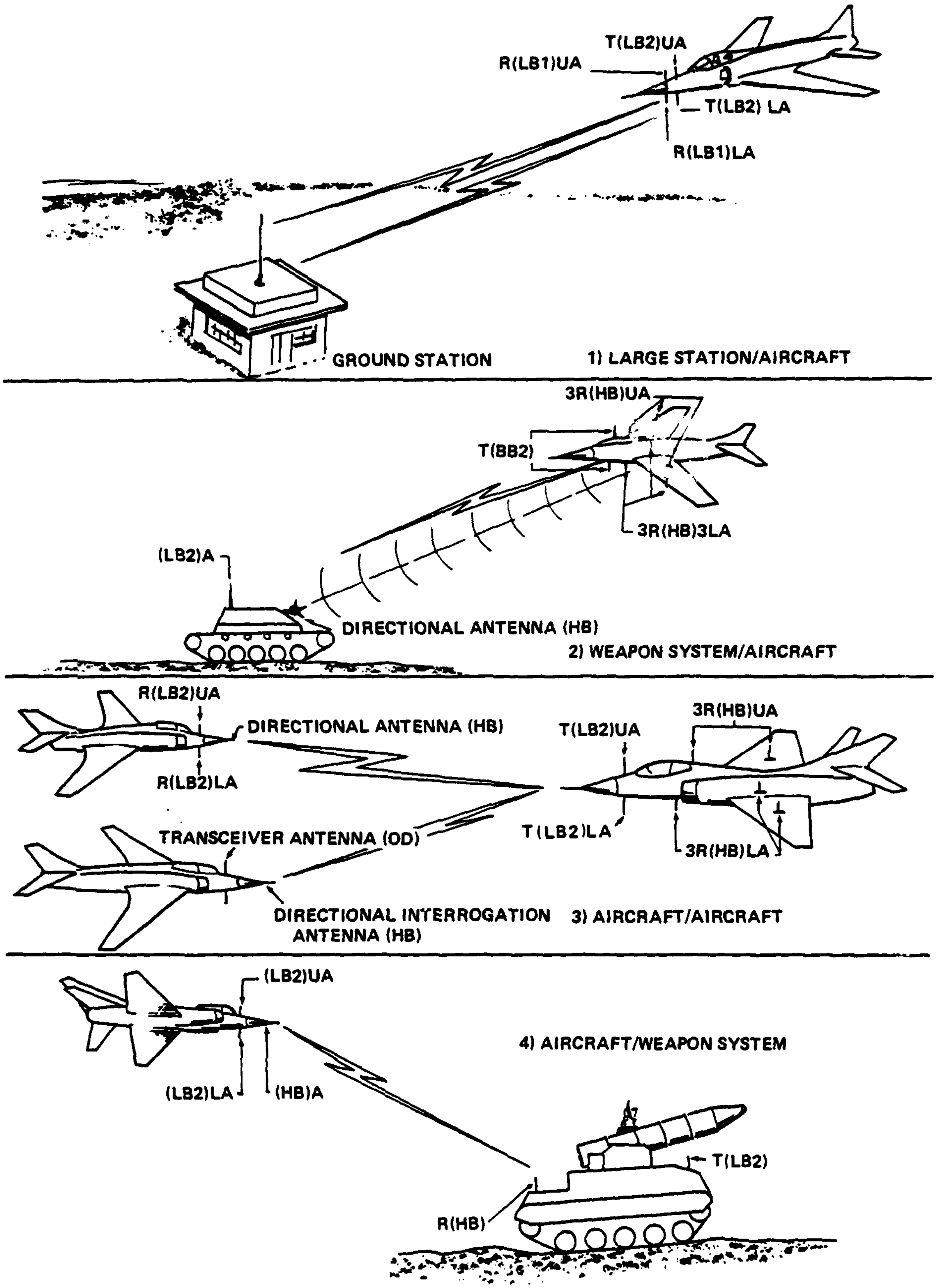


Fig.3(a) Identification configurations

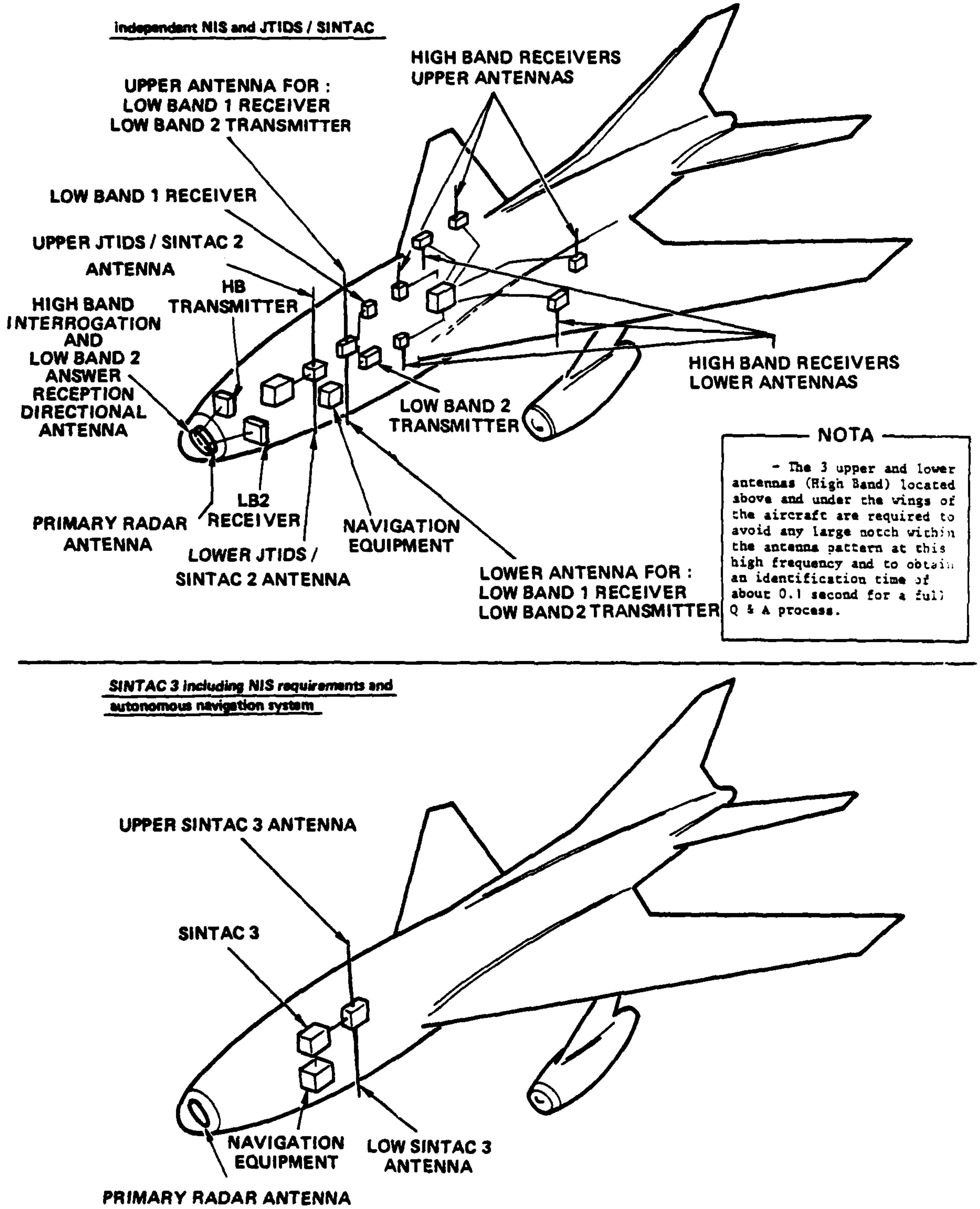


Fig.3(b) On board JTIDS/SINTAC 2 and NIS systems

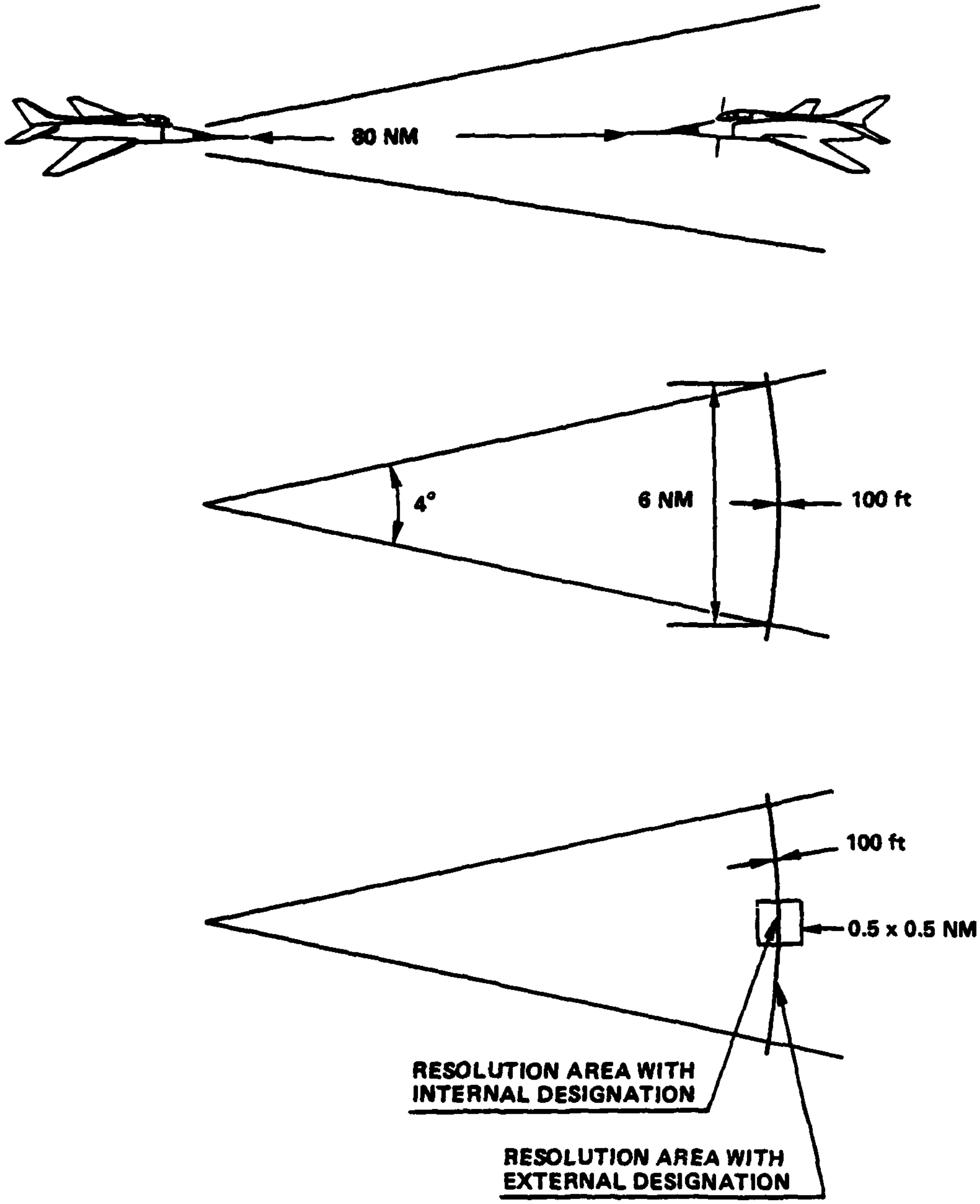
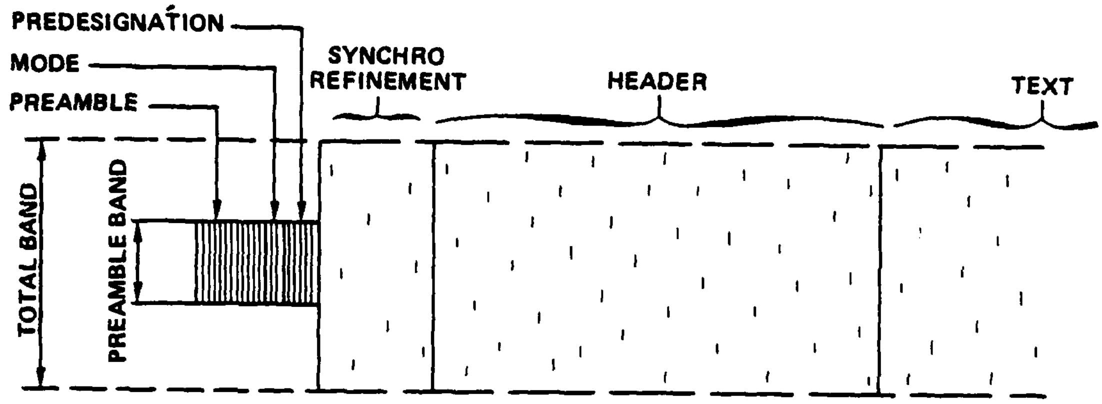


Fig.4 Compared resolution with internal and external designation for air to air identification

VERSION 2 : INTERNAL DESIGNATION



VERSION 2 : EXTERNAL DESIGNATION

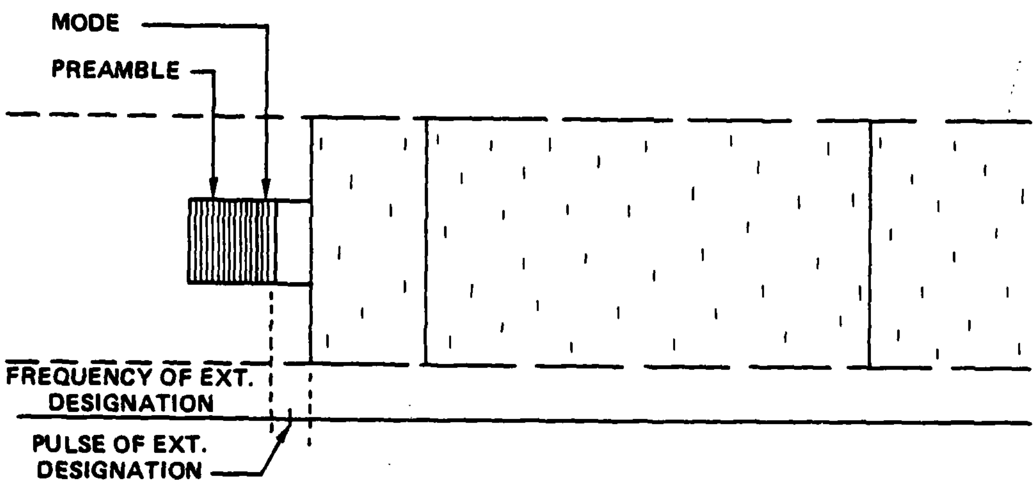


Fig.5 Message format

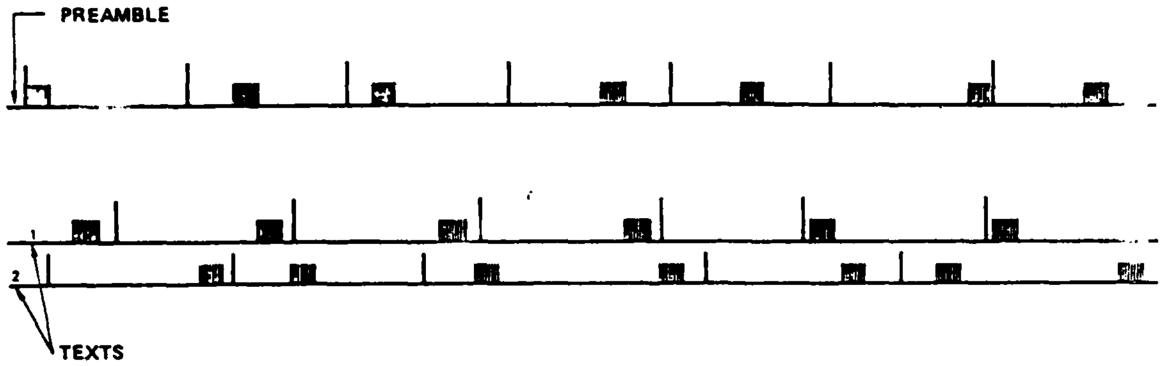


Fig.6 Jamming of the preamble by the text

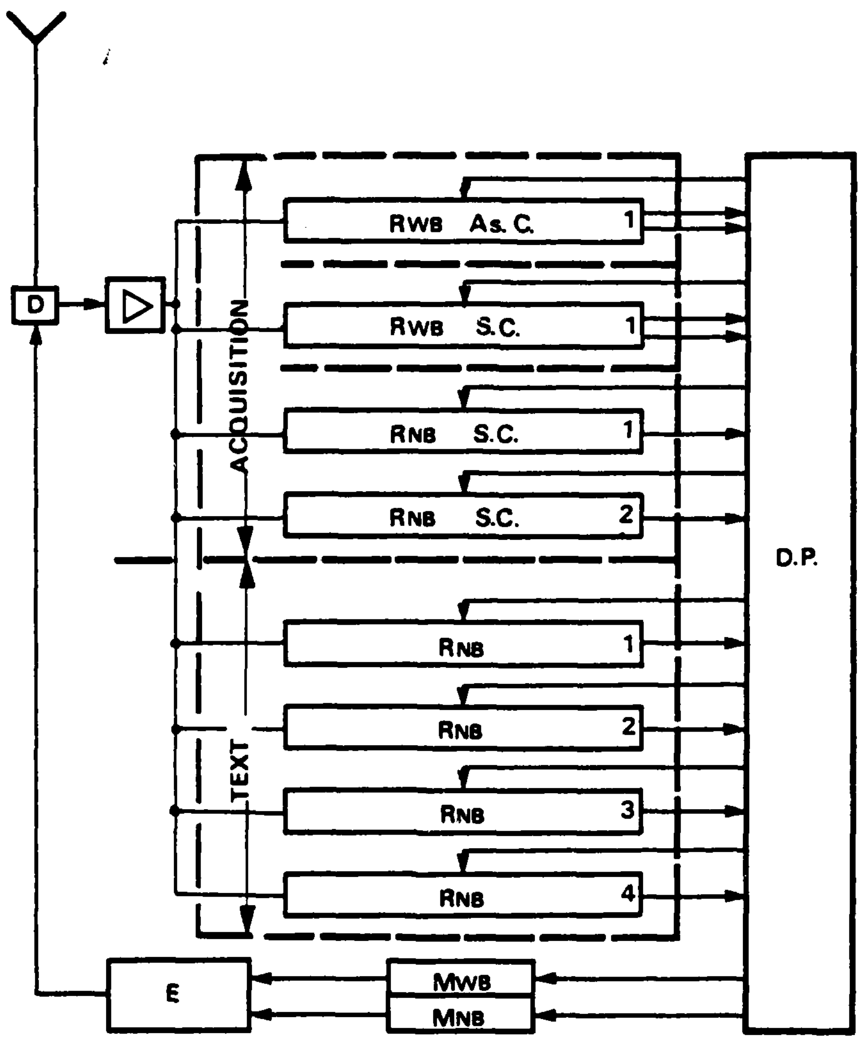


Fig.7 Complete terminal

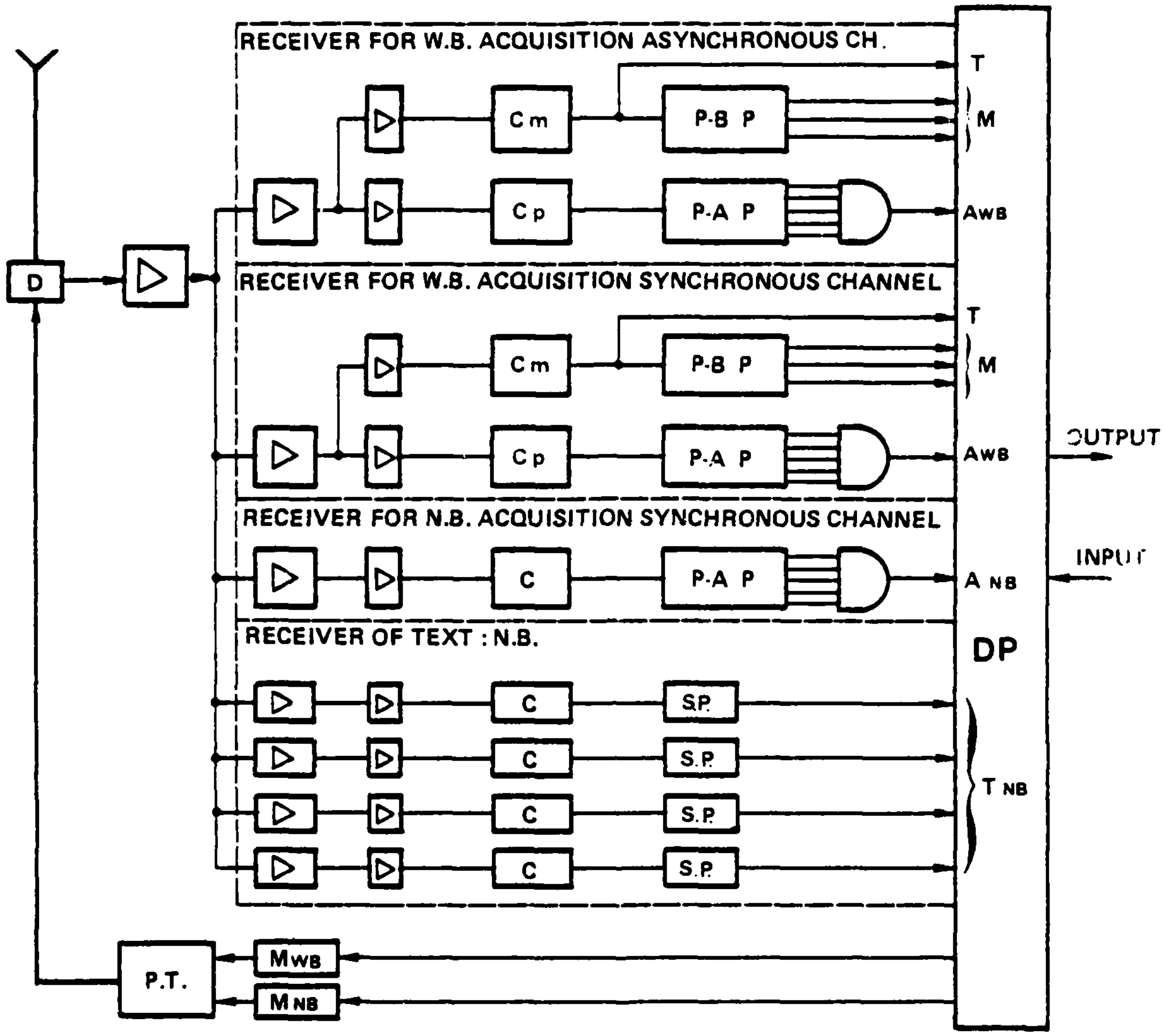


Fig.8 Terminal A2

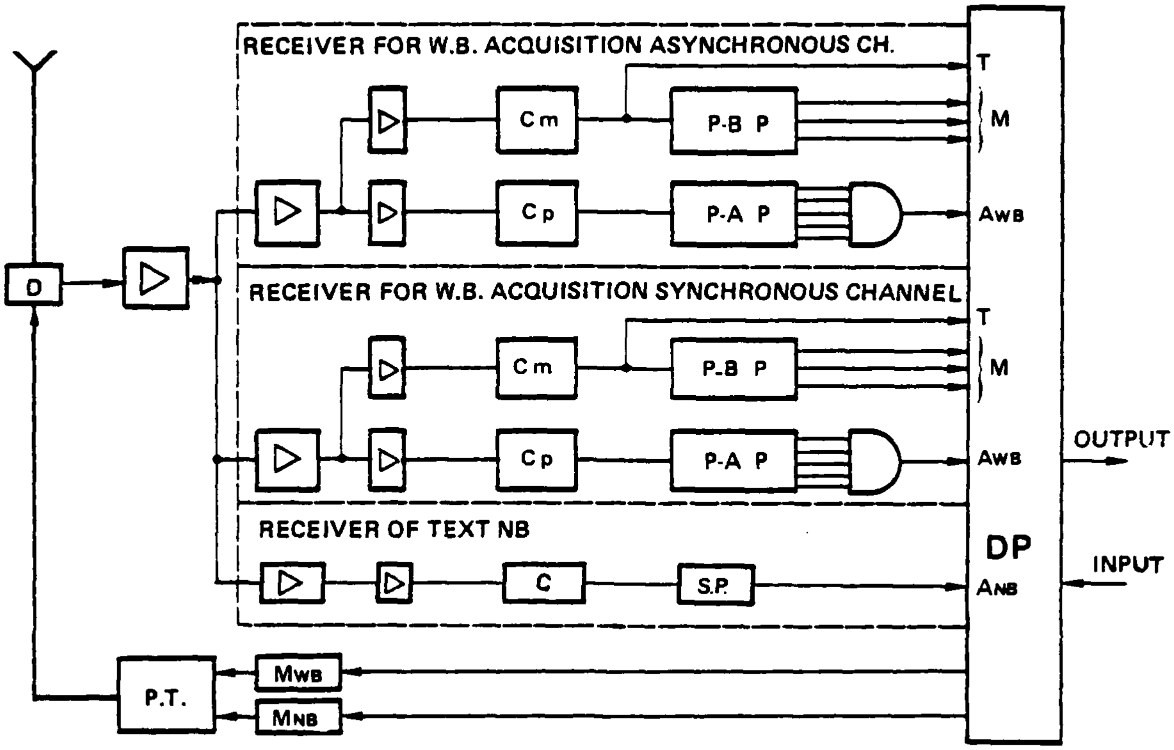


Fig.9 Terminal B4

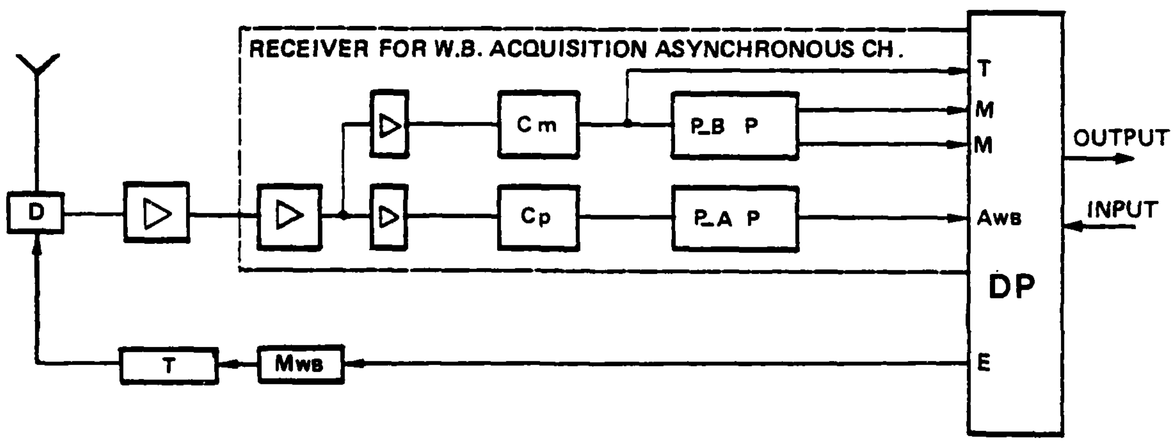


Fig.10 Terminal B6

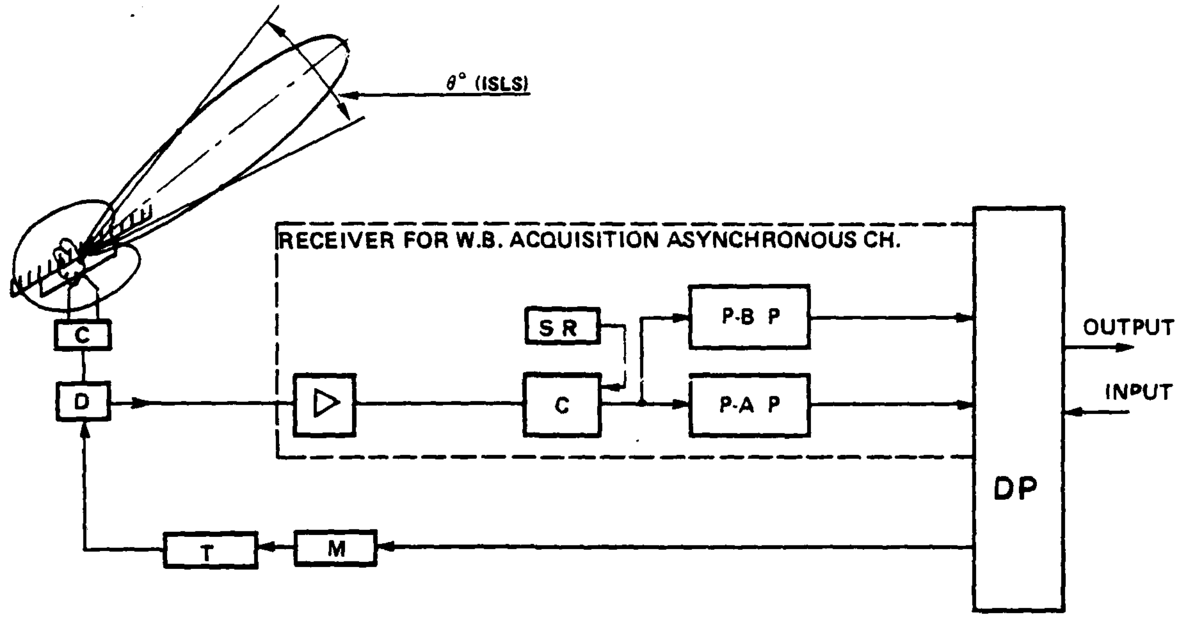
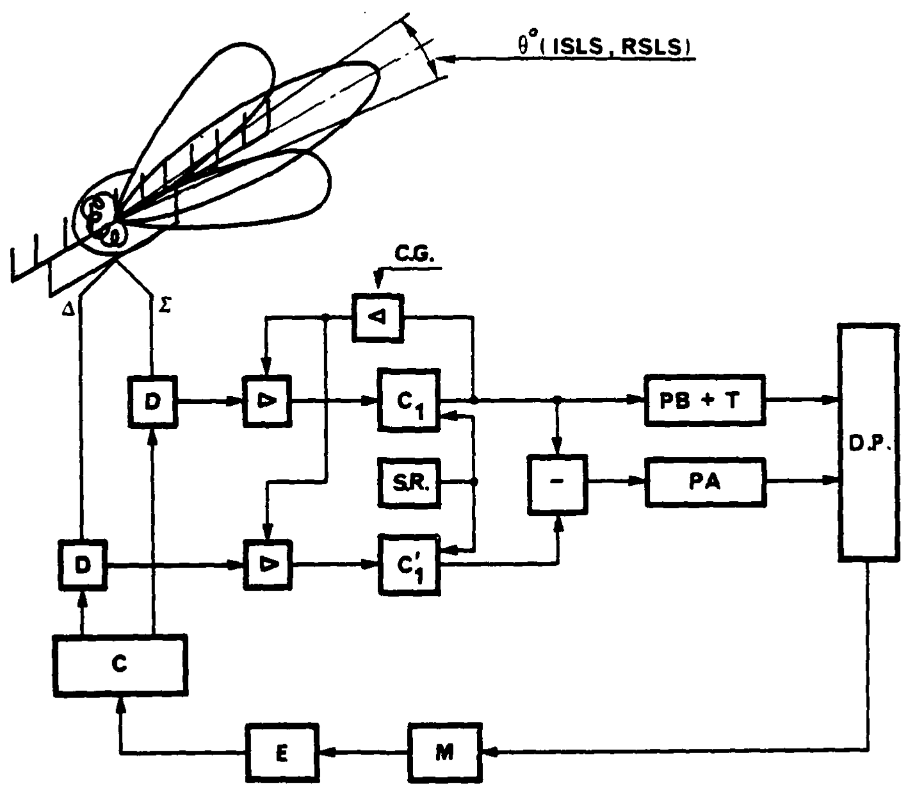
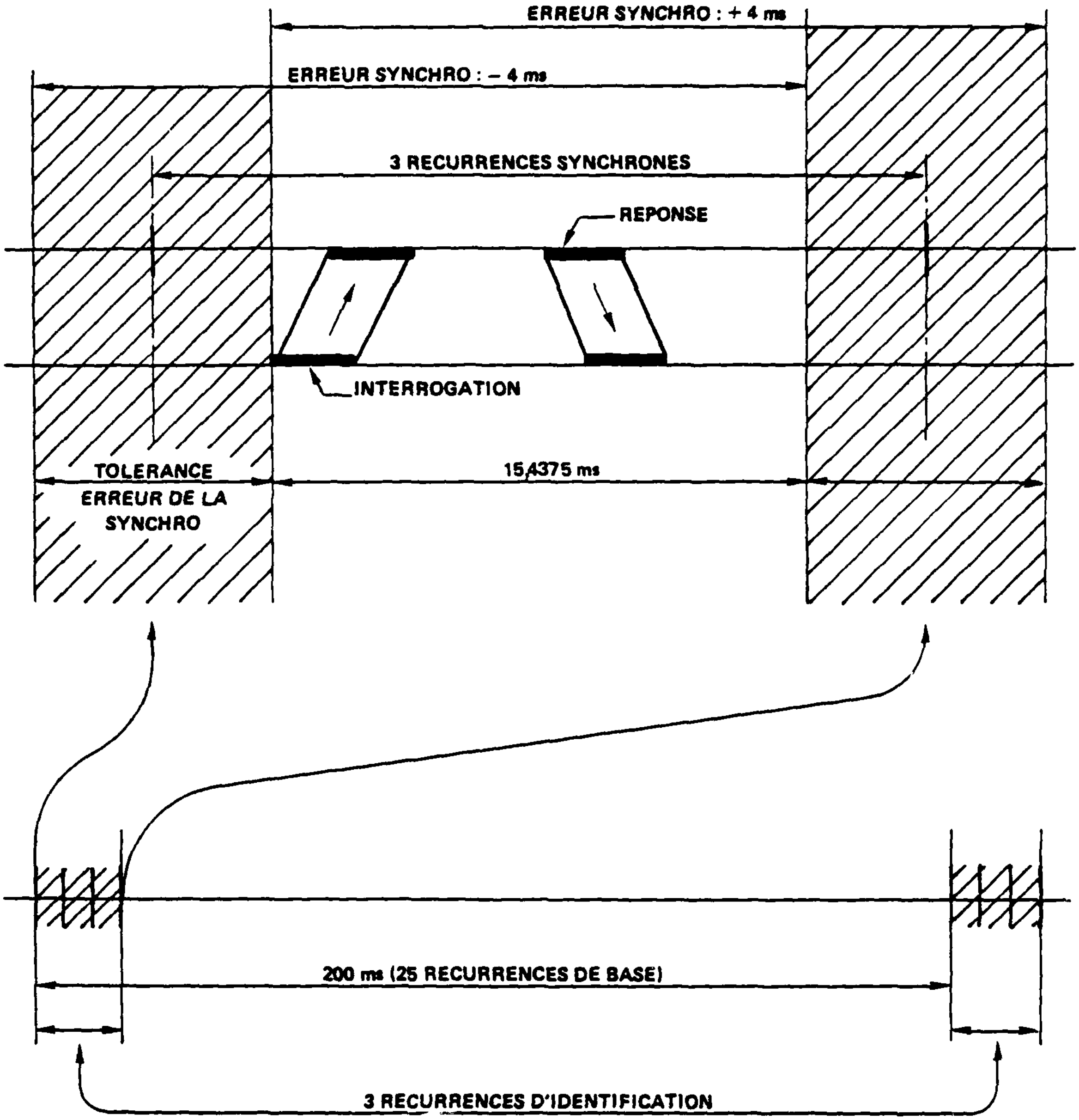


Fig.11 SINTAC 3 Terminal B7 -- For utilization with directional antenna and ISLS



(G.C.) : Gain control of the 2 amplifiers by the Σ channel
(R.S.) : Reference signal common to both convolvers
(C_1, C'_1) : 2 identical convolvers mounted on the same substrate

Fig.12 SINTAC 3 B7 (RSLS) – For utilization with directional antenna and ISLS-RSLS devices



ORGANISATION DU CANAL SYNCHRONE D'IDENTIFICATION

Fig.13 Synchronous channel identification

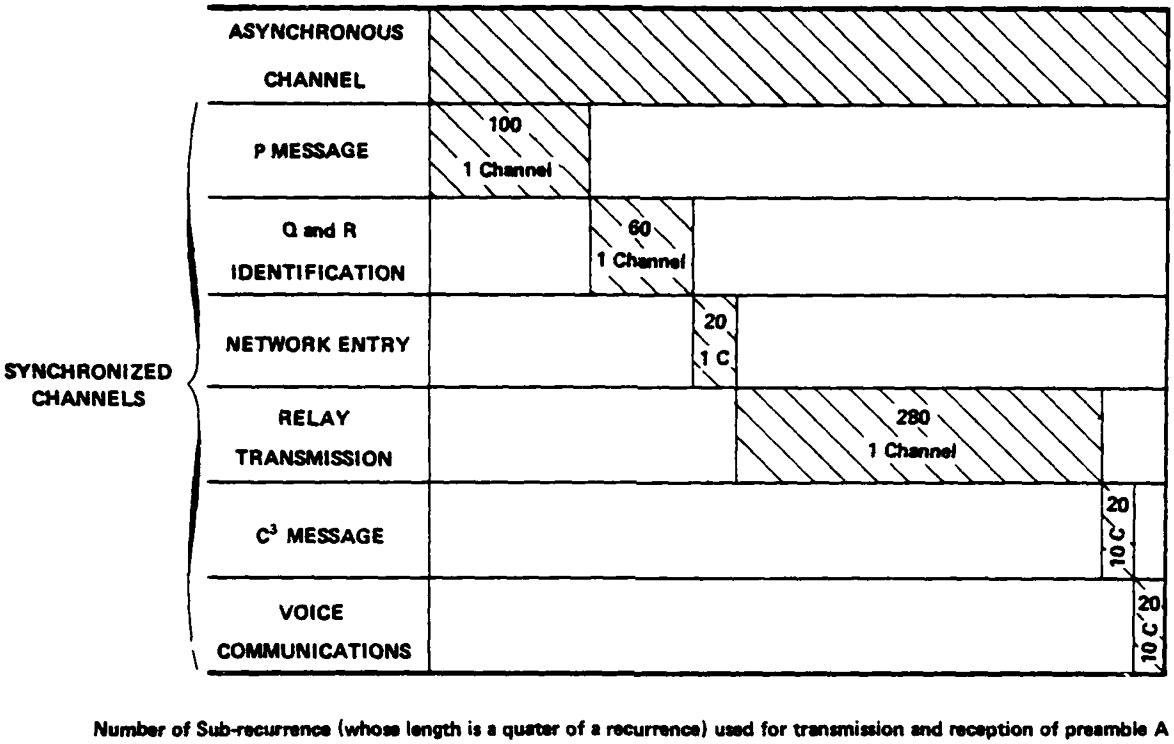


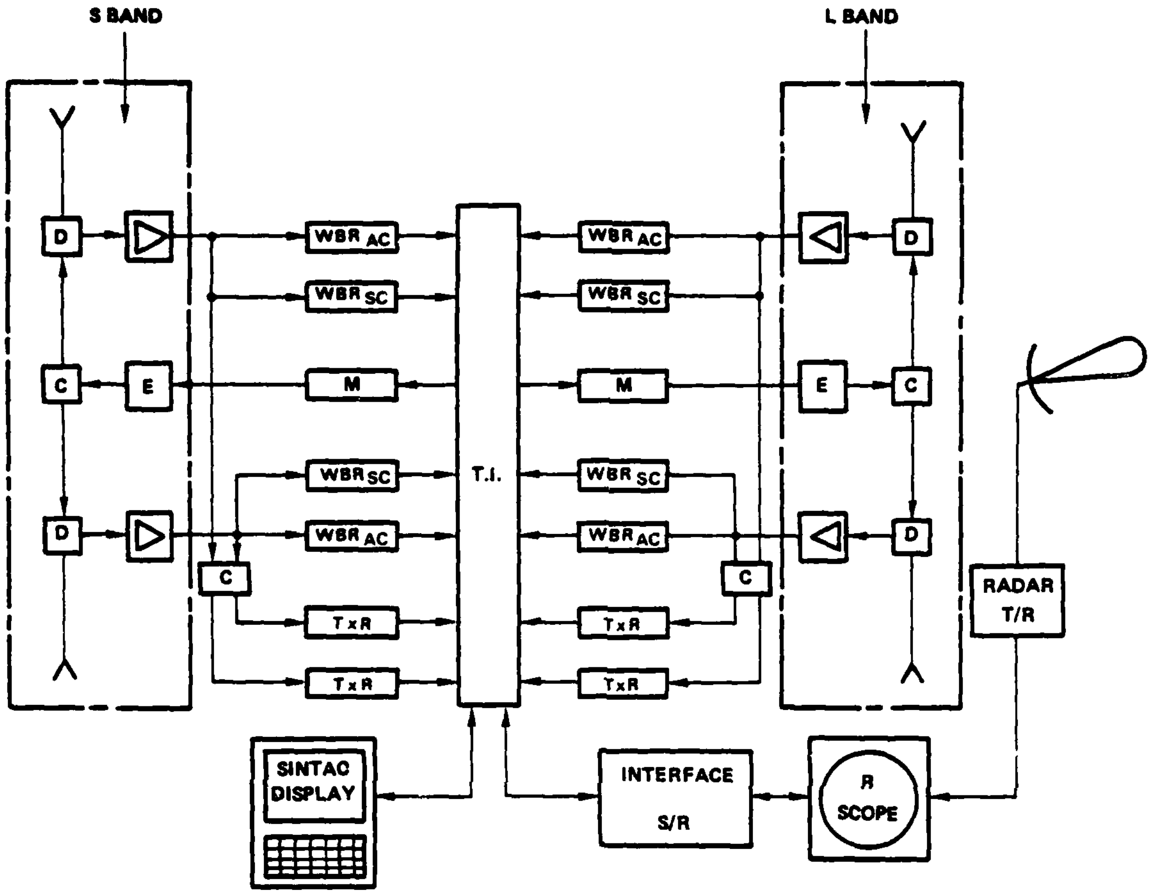
Fig.14 Channel organization time allocated to each channel

- OPERATION SAFETY
- HOSTILE JAMMING

	SYSTEM	JAMMER	SYSTEM/JAMMER RATIO
1	JTIDS LB NIS SB	LB JAMMER SB JAMMER	2/2
2	JTIDS LB NIS LB	LB JAMMER LB JAMMER	2/1
3	SINTAC 3 { MIDS + NIS } LB	LB JAMMER	1/1

LB = L BAND SB = S BAND

Fig.14(a) Cost/efficiency of the various one and two band system versus jammers



- THE WIDE BAND ASYNCHRONOUS AND SYNCHRONOUS CHANNEL RECEIVERS CAN BE EXCHANGED
- TEXT RECEIVERS CAN BE EXCHANGED
- THE H.F. STAGES ONLY ARE SPECIFIC THE S AND L BANDS
- THIS TERMINAL PERFORMS ALL THE COMMUNICATION, NAVIGATION BROADCAST AND Q AND R IDENTIFICATION FUNCTION IN THE L AND S BANDS IN ACCORDANCE WITH CHANNEL ORGANIZATION AND SAFETY REQUIRED

Fig.15 S and L band terminal block diagram

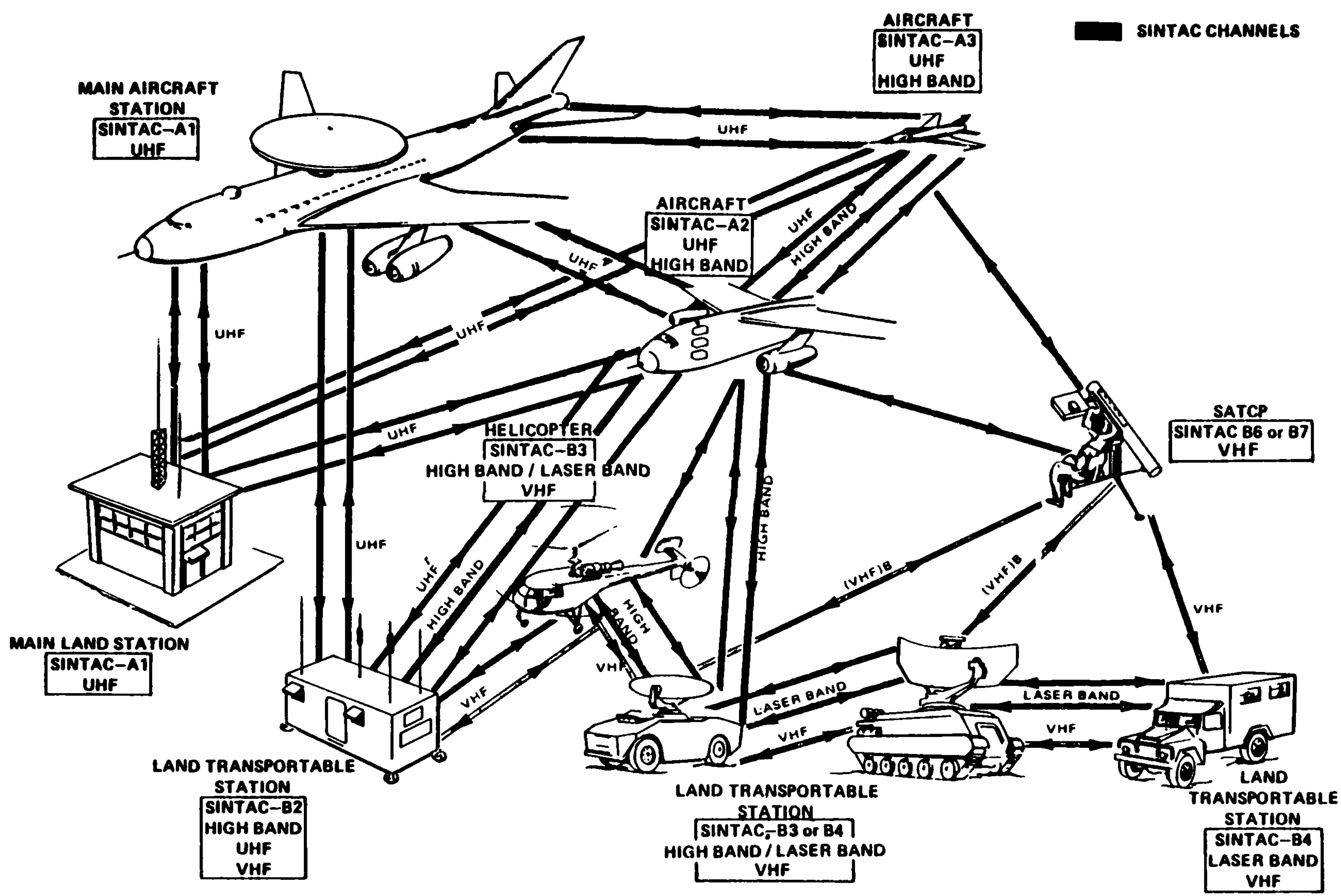


Fig.16 Communication types in accordance with the carrier

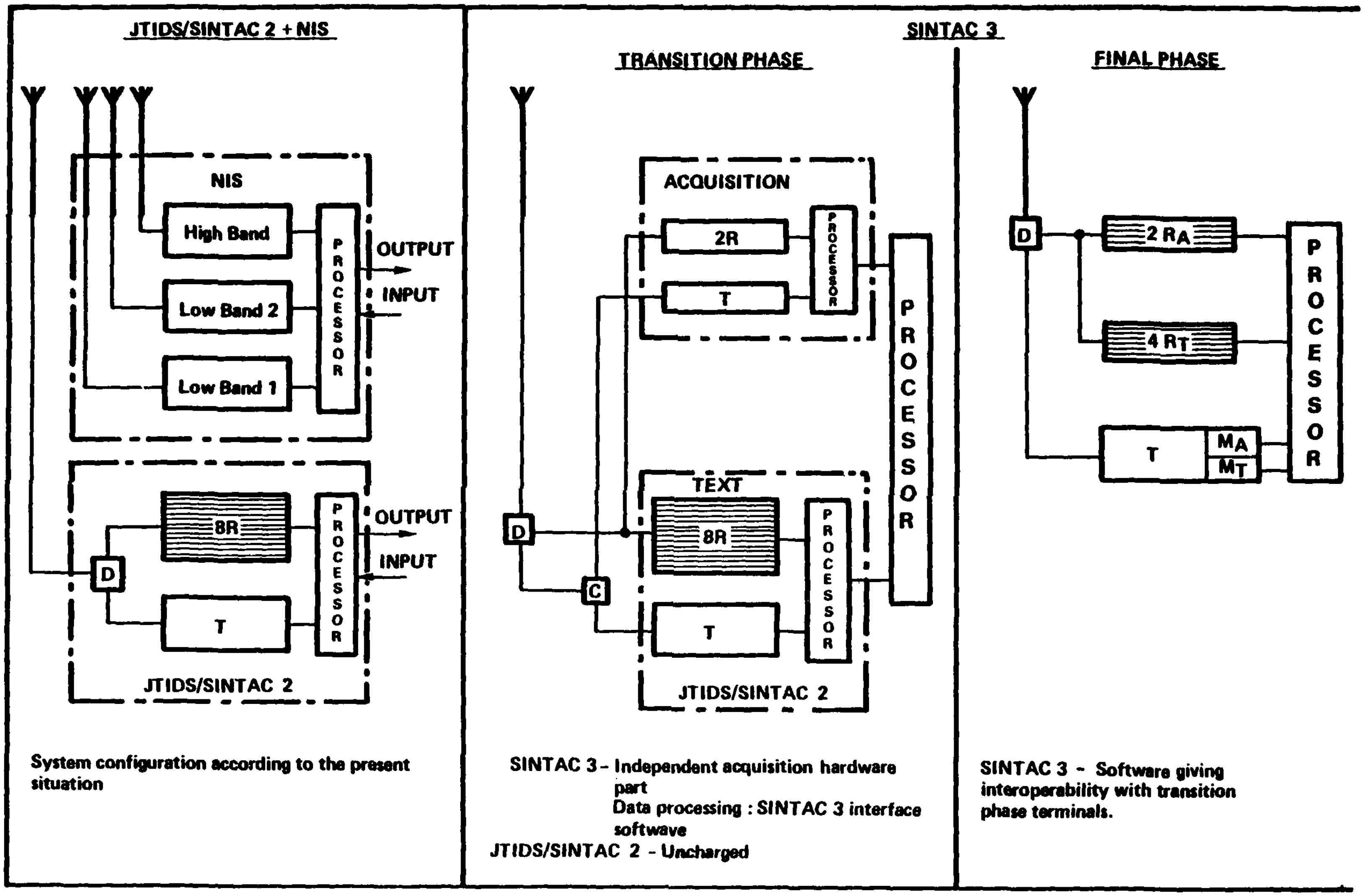


Fig.17 Terminal configurations

SYSTEM	COMPATIBILITY								PERFORMANCES	
TRANSMISSION \ RECEPTION	TACAN	IFF MK10	DABS	NIS band L	JTIDS TDMA	JTIDS DTDMA	SINTAC-3 V-2	SINTAC-3 V-3	NIS	MIDS
TACAN	1	1	1	1	1	1	1	1	⊠	■
IFF MK10	1	1	1	1	1	1	1	1	■	⊠
DABS	1	1	1	▨	1	1	1	1	⊠	⊠
NIS L band	▨	▨	▨	▨	■	1	⊠	⊠	1	⊠
JTIDS TDMA	1	1	1	■	▨	⊠	1	1	■	1
JTIDS DTDMA	1	1	1	1	⊠	1	1	1	■	1
SINTAC-3 V-2	1	1	1	⊠	1	1	▨	⊠	▨	1
SINTAC-3 V-3	1	1	1	⊠	1	1	⊠	1	1	1
NIS L band + JTIDS DTDMA	1	▨ 1	▨ 1	▨ 1	⊠	1	⊠	⊠	1	1

□ 1 GOOD PERFORMANCES (COMPATIBLE)

■ BAD PERFORMANCES (INCOMPATIBLE)

▨ 1 SOME DEGRADATION IN PERFORMANCES

⊠ NOT IN USE AT THE SAME TIME OR FOR THIS FUNCTION

Compatibility between all L band systems

INTEGRATED NAVIGATION-TF/TA-SYSTEM BASED ON STORED TERRAIN DATA PROCESSING

by

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SUMMARY

This paper is a contribution to the solution of the problem, how terrain-following-flight can be made more reliable and optimal in the sense of smoothing against the terrain. The main source of information in the discussed system is a terrain data base aboard the aircraft, in which the height values of the overflown area are stored coherently. A system description will be given including the monitoring function. A hardware - and software design of the combined navigation - and tf/ta-flight control system is developed, adapted to a modern avionic-system-architecture.

1. Introduction

The tremendous progress in microprocessor technology and digital data processing have caused a trend towards such highly sophisticated avionic system with decentralized system-architecture.

This trend has resulted from the progress made in microprocessor-technology and in development of new mass-memory-devices. In this context one could remember the availability of high speed single-chip-microprocessors and highly integrated bubble-memory-circuits for application in digital avionic systems.

Classical real-time solution to problems involving complex mathematical relationships can be performed by parallel computation and parallel processing in a microprocessor system with decentralized architecture.

In modern military aircraft there is an increasing tendency of achieving the capability, to fly in very low heights during adverse weather conditions and in any electromagnetic environment. If an aircraft remains low at near sonic speed above all occuring types of terrain, it is difficult to be detected and very difficult to hit.

With decreasing heights there is a reduced vulnerability risk due to hostile defence systems, but an increasing risk of collision, especially under adverse weather conditions in a mountainous area.

To fly as low as possible at near sonic speed above occuring types of terrain means that the terrain profile in front of the aircraft must be known very accurate.

An automatic control is done in state-of-the-art-system by computing on optimal flight path from terrain-following-radar-measurement. But there are inherent physical limitations.

By aiding this terrain-following-radar with a terrain data base aboard the avionic-system the false-alarm-rate may be decreased.

2. TERPAC-System

The system, which offers this characteristic , is called TERPAC. TERPAC is an abbreviation of TERRAIN-PARAMETER-COMPARSION (fig. 1). The comparison is done in the position-fix-mode by matching the measured terrain-signatures against terrain-reference-signatures stored in the terrain mass-memory. In the case of terrain-profile-correlation (PROKOR) the actual position of the aircraft is computed by matching the scanned terrain-signals with digitized terrain values, stored in the TERPAC-mass-memory. In the case of scene-correlation (SZEKOR) the comparison is performed between an actual image of the terrain-contour taken with an image-making sensor and a stored reference image.

Whether the position-fix-system operates in a single-fix - or in a continuons mode depends on the significance of the overflown terrain and on the functional status.

If failure is indicated, the system will be resynchronized by switching over from the continous-mode to the single-fix mode.

On the other hand the terrain-following-system will be provided with the stored terrain data in the fourth mode in order to optimize the reference flight-path against the actual flight-path by means of software. The critical point on the terrain ahead of the vehicle, predicted by the TERPAC-System, and the optimal flight-path are computed.

The optimal flight-path is as close to all stored terrain-grid-points as practical for controlling the minimum clearance above the terrain.

In fig. 1 the main characteristics of the TERPAC-System have been highlighted:

- automatic, because the processing is automatically performed in the preplanned mission waypoint-computation

- autonomous, because the system is independent upon external stations and data-links like GPS or other classical radio navigation system

the further advantages

- all weather-, night capability, high anti jam resistance, low detectibility correspond to the behaviour of the downlooking radar-altimeter-sensor.

In order to extract the terrain-profile ahead, the position of the aircraft must be known very accurate with respect to the terrain in front of it.

In the TERPAC-System both the position and the predicted terrain-following-profile are calculated in a common terrain-pre-processing stage.

Fig. 2 shows the principle of terrain-data-management within three tf/ta cycles n, n + 1, n + 2. Corresponding to the travelling progress the numbers characterize the terrain-frames, which will be processed by the tf/ta-subsystem in a predective manner and by the correlation-subsystem in a recursive manner.

The tf/ta-frame n, generated from the terrain mass memory around a predicted position, will be matched by the correlation system two cycles later. The typical frame-length will be determined according to the overflown type of terrain.

The block-diagram in fig. 3 shows the principle data flow and the control functions of the TERPAC-system.

The terrain-map-mass-memory and the digitized terrain-data stored in it serve as the basic data-source both for the position-fixing-system and for the terrain-following/ avoidance-system. In addition the digitized terrain-data may be pre-processed for projection on a digital-map-display instead of an analogous device or for obstacle warning to the pilot.

The TERPAC-processor as a central processing unit computes in a recursive manner the actual position by correlation-algorithms and in a predicting manner the reference flight-path by optimizing algorithms.

The differential output signals generated in the tracker software module by comparison of the predicted reference flight path with the actual flight path are fed to the flight control system.

Switching over from one mode of operation and the inherent sensor subsystem to another will be controlled prior to data pre-processing.

The position-fixing-subsysteme consists of:

- inertial measurement unit
- radar altimeter subsystem
- data pre-processing modul
- TERPAC-processor
- terrain-mass-memory
- kalman-filter

Fig. 4 shows the principal data-flow of the combined navigation and correlation-system. The overflown contour is calculated from the differential output signal of baro-sensor and the platform vertical channel and from the radar altimeter outputs. The actual position of the vehicle corresponds to the correlation-maximum, computed in the terpac-unit.

Not only the difference between the indicated and the actual position is processed in the kalman-filter, but the difference between the predicted height above ground and the actual radar altimeter reading.

By estimating the errors of the inertial unit and feeding back it to the navigation computation, the behaviour of the aided navigation system will be improved. The dynamic models of the inertial-system and of radar altimeter- and positions-fix-measurement are implemented in the kalman-filter.

3. Position-fix-mode

The values of clearance altitude and the altitude obove datum-plane, continuously measured by the radar altimeter and the baro aided vertical channel of the intertial measurement unit (fig. 5), are simultaneously stored in a pre-processing stage. After pre-processing (noise-filtering, data reduction, time synchronisation) the measured terrain profile is generated as differential signal and fed to the TERPAC-correlator stage.

This processing method guarantees that the motion of the vehicle in the vertical plane is eliminated, ensuring that additional terrain elevations cannot be simulated. These additional terrain-elevations could reduce the resulting position-fix accuracy of the TERPAC-aided intertial navigation system.

The additive interfering noise signals will degrade the signal to noise ratio and thereby increase the false-fix-probability.

Comprehensive filtering methods are necessary both for correlation and terrain following computation to reduce this influence. The result of such signal processing is shown in fig. 6.

The terrain profile as a difference-signal between barometric and radar-altimeter-measurement is derived from raw-data of flight experiments.

The problem of terrain-noise-reduction is that of extremely low-pass-filtering. Because the whole terrain-information is contained in low spatial frequencies, compared to higher frequencies of additive terrain-noise.

Furthermore this error-behaviour will be modelled in the kalman-filter-algorithms.

The filtered terrain profile in the middle of fig. 6 is a good aproxmation of the profile at the bottom, generated from topographical map of 1:50.000 scale. The low pass filter will be designed with an adaptive corner-frequency, adapted to the spatial domain spectrum and the velocity of the aircraft.

Besides the motion in the elevation-plane the aircraft will be operated by the pilot in the horizontal plane. In contrast to a preplanned mission with preplanned flight-path in missile-guidance, the TERPAC-system must guarantee maximum flexibility and will be subordinated to the basic navigation system.

To operate during horizontal maneuvers and variable flight-paths is the key to TERPAC-position-fix-mode.

The principle of navigation update consists of correlation performance with a so called flight-template, generated from the measured inertial position-values within the update area.

Representing the height-values on a geographic-grid, the digital map of the update-area will be preprocessed from the terrain data stored in the terrain-mass-memory. It will be constructed around a principle track, predicted from the last coarse-informations.

The size of the update area will be adapted automatically to the time-dependent error-behaviour of the inertial measurement unit and the travelling time between two update areas.

By this means it is guaranteed, that the indicated flight-template doesn't lie out of the update area.

The correlation process is initiated after crossing the waypoint. The flight template is displaced along and across the flight direction until the best fit occurs.

The position of the correlation maximum corresponds to the actual position of the aircraft.

After each desplacement the height values under the flight-template will be extracted from the terrain-data-base and put together to a new virtual correlation track.

Position fixes are calculated in the continous-mode within overlapping update areas. The size of the update-area varies because of decreasing position uncertainty. Like a sliding correlation-window the update area follows the flight-path.

The position-fix-system operates in two modes. If a failure occurs or the radar-altimeter is out of track, the system will be resynchronized in a single-fix-mode. In this case the update area is expanded.

4. Terrain-following/avoidance-continuous mode

The main components of the tf/ta-subsystem (fig. 7) are:

- tf-controller
- terrain-mass-memory
- radar-altimeter
- failure-logic
- tf-, mapping-radar
- tf/ta-, mapping-display

In the well known ski-type-algorithm only information about the terrain in the present line-of-sight of the terrain-following-radar is used. In contrast to it, the tf-controller takes full advantage of the stored map by using the over-the-hill data. The result is an optimal fit to the terrain with better descenting into valleys instead of flight-path-angle control.

In developing the terrain data aided terrain-following-system, great emphasis must be put on finding a safe, reliable and confident failure logic. To achieve this, the control-loops have to be redundant together with sophisticated data checks.

If a failure occurs in the tf-radar-system, the logic decides, whether the radar-system will be aided on the level of distance - or angle measurement.

In the case of total disturbance the tf-controller calculates continously the distance to the nearest obstacle in front of the aircraft by using the stored terrain data and the radar-altimeter-signals.

The radar-altimeter is a back-up sensor in the tf-system. It provides altitude-information, which is transformed into a command-signal, when no returns from the forward-looking-radar are available and the terrain-data are not reliable.

On the other hand a pull-up-command is initiated, when the actual height is below the selected clearance height.

In the manual or automatic tf/ta-flight-control-system it is important, to indicate the upcoming terrain-situation to the pilot via display. The examples in the following figures demonstrate, that the synthetic terrain-characteristics can be generated from the stored terrain-data-base.

The pilot monitors the correct function of the system and its reponse to terrain ahead, observing the E-Scope during terrain-following-control (fig. 8) The terrain heights including the secure flight-path are displayed against the distance in logarithmic scale. Indicating valleys out of the line of sight is an advantage of this synthetic display.

The horizontal plane-display in fig. 9 is a good tool for manual or automatic ta-control.

Corresponding to the aircraft-altitude avoidance-regions will be calculated. The advantage in this case is, that the hidden-lines behind the hills are presented. Selecting the avoidance-regiones the ta-control-flight will be performed by the pilot.

The synthetic Gray-Scale-Display in fig. 10 offers the possibility of manual or automatic tf/ta-control. A perspective of the terrain ahead is simulated by gray-scales decreasing with distance.

Furthermore additional symbols for command, flight-path, position and altitude of the aircraft are shown. In fig. 11 the advantages of a navigation tf/ta-flight control-system based on stored terrain-data are summarized:

- The system operates either in a back-up-mode by time synchronuous aiding the tf-radar-system or in a stand-alone-mode during operational phases of silence in a hostile jamming environment.

- Also reliable recognition of obstacles is guaranteed in the case, when radar-returns are not available.

The results of optimal flight path approximation against terrain are:

- reduced maximum g-loads

- soft-ride commands

A possible hardware- and software-design of the combined navigation- and tf/ta-flight-control-system, adapted to a modern avionic-system-architecture, is shown in fig. 12.

The key elements of this architecture are:

- hierarchilly structured bus-system

- decentralized, autonomous microprocessor modules

The signals coming from flight-computer, the radar-altimeter and the terrain-following-radar are transmitted into the terpac-subsystem via the avionic bus.

The terrain-processor controls the data-transmission in both directions and provides the tf/ta- and navigations-subsystem with the preprocessed terrain-data. If the data-channel-capacity is not sufficient, the data will be transmitted via an inter-face-unit IFU.

For monitoring the terrain-datas on a multi-function-display or a head-up-display, the preprocessed terrain data will be transmitted via direct memory access to the display-management-stage.

A ground based data-management is needed, to transfer the terrain data from the terrain data base to the terrain-mass-memory aboard the aircraft. Two methods can be distingnished:

At first the terrain data base of a whole area of about 150.000 km^2 is permanently stored in the terrain mass memory. In this case only the flight-computer must be programmed by inserting the waypoint-coordinates before a mission.

In the other case the digital terrain data are preprocessed during the mission-planning-phase and transferred together with other information into the terrain mass memory of the aircraft via a datalink.

METHOD

- TERPAC--processor matches measured terrain--signatures against terrain--reference--signatures stored in the mass memory and compares TF/TA--flight path with stored terrain data

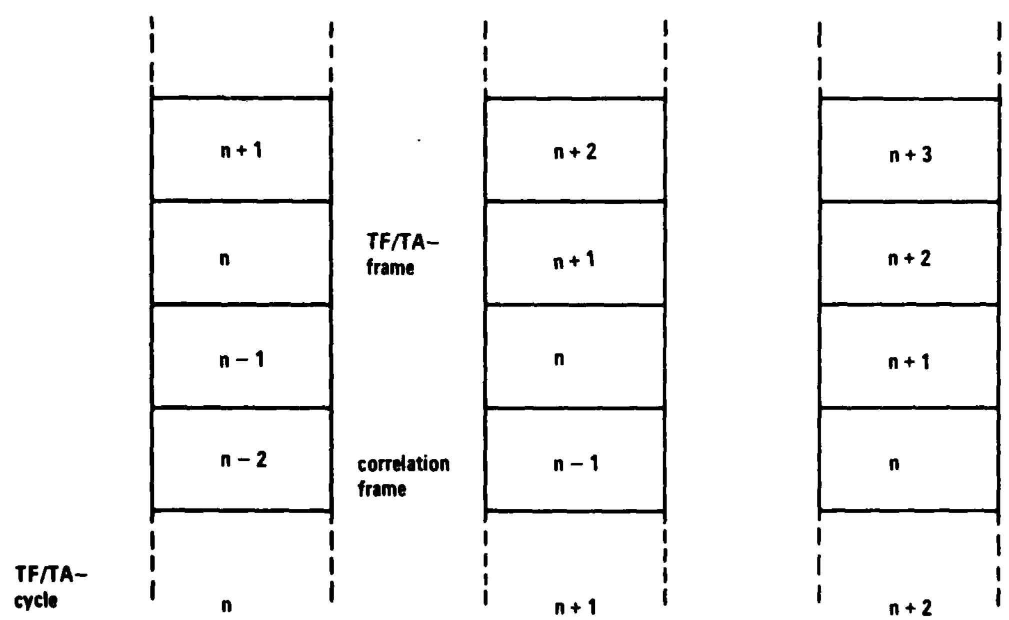
MODES OF OPERATION

- *inertial mode*
- position--fix--mode (single, continuous) -- PROKOR --
- SZEKOR--position--fix--mode
- terrain--following/avoidance--continuous--mode

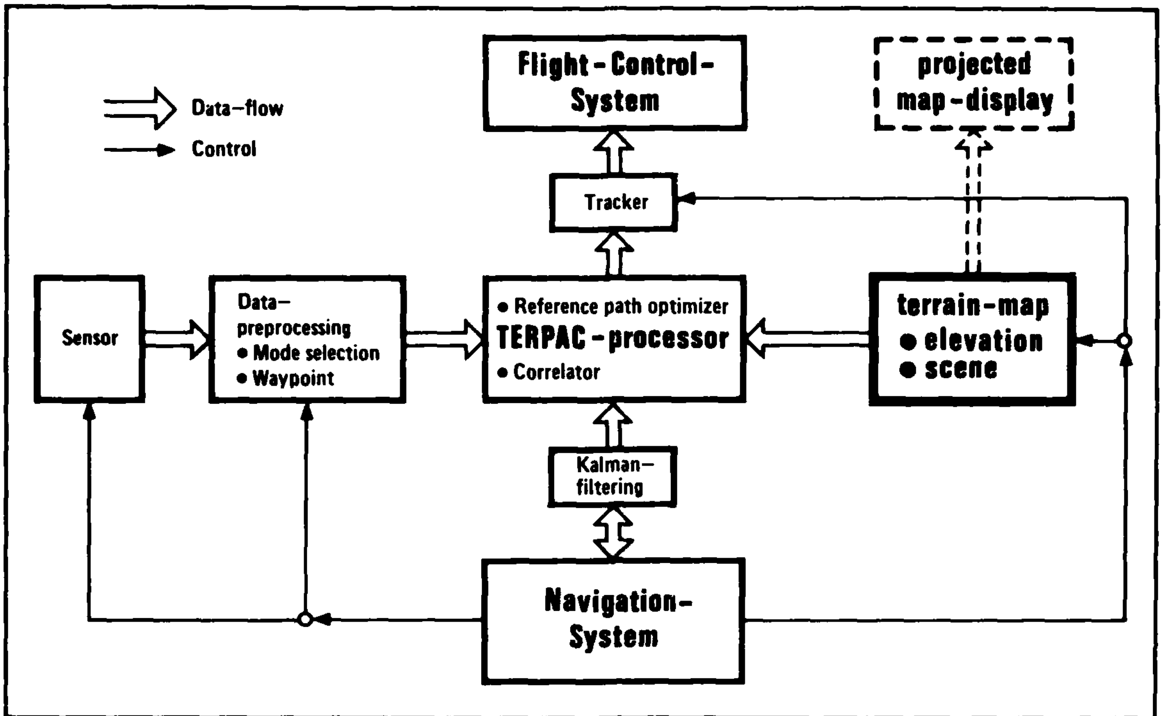
CHARACTERISTICS

- automatic
- autonomous
- all--weather--, night capability
- high anti--jam resistance, low detectibility

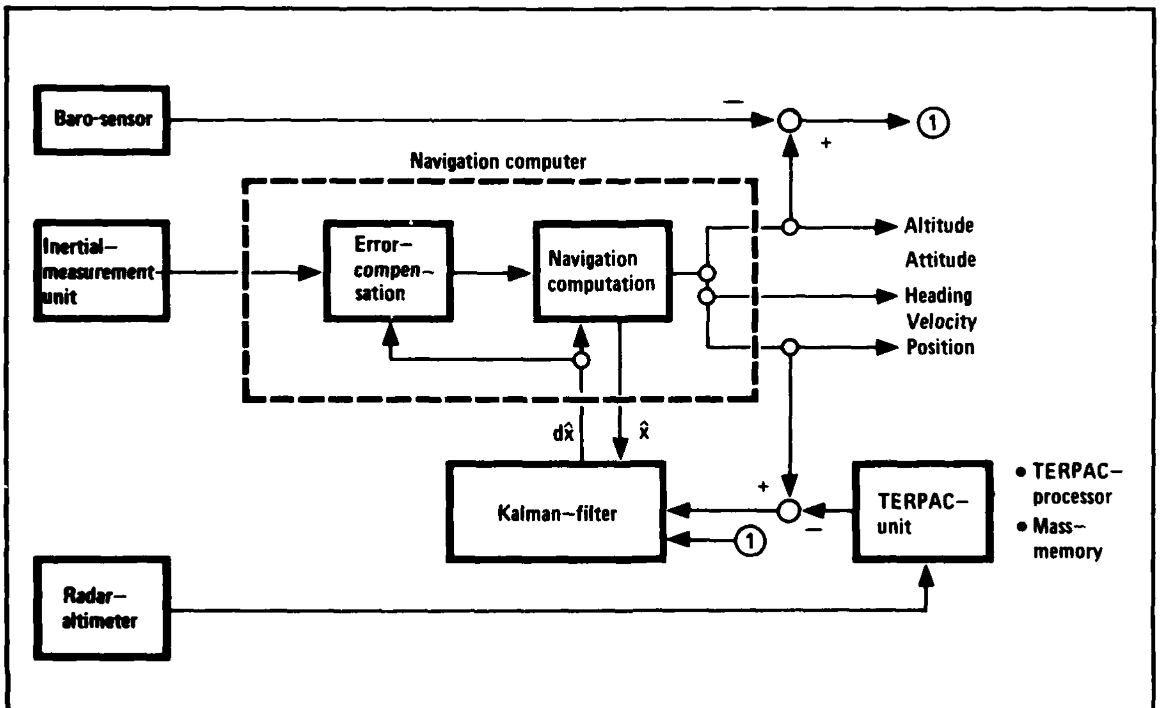
TERPAC | TERPAC (TERRAIN - PARAMETER - COMPARISON) | Fig.1



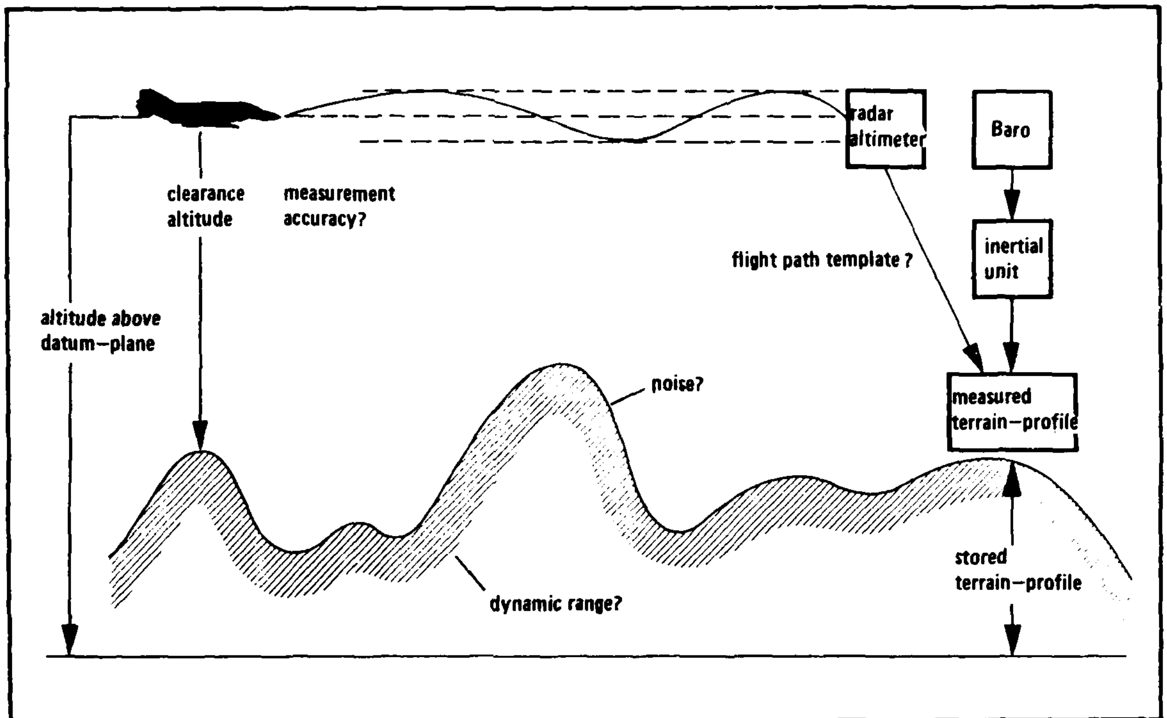
TERPAC | PRINCIPLE OF TERRAIN DATA PROCESSING | Fig.2



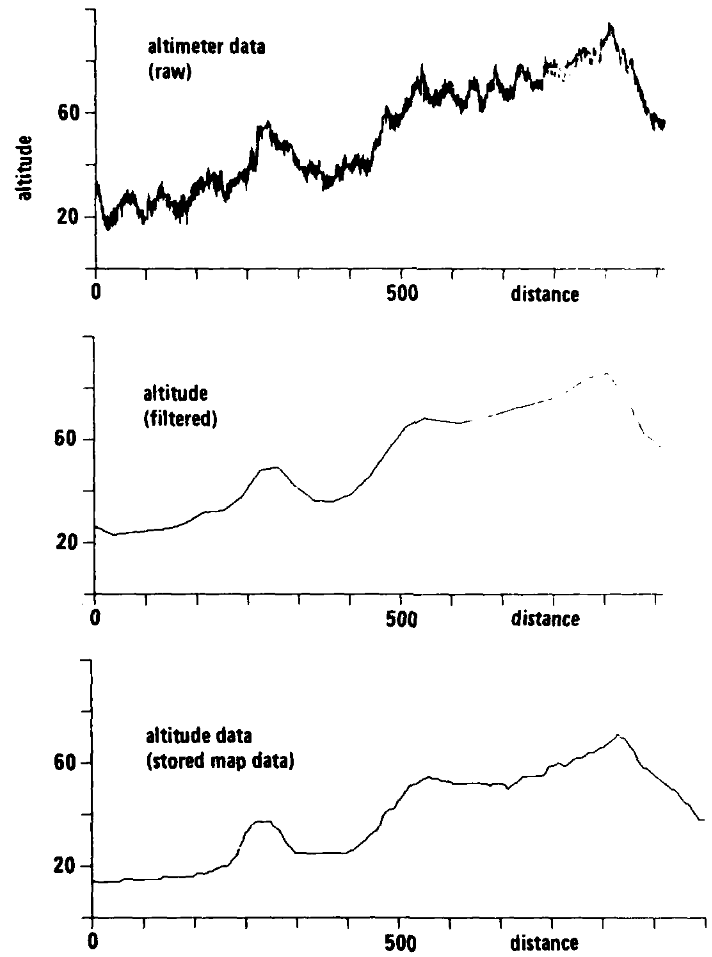
TERPAC	NAVIGATION–/FLIGHT CONTROL SYSTEM	VFW Fig.3



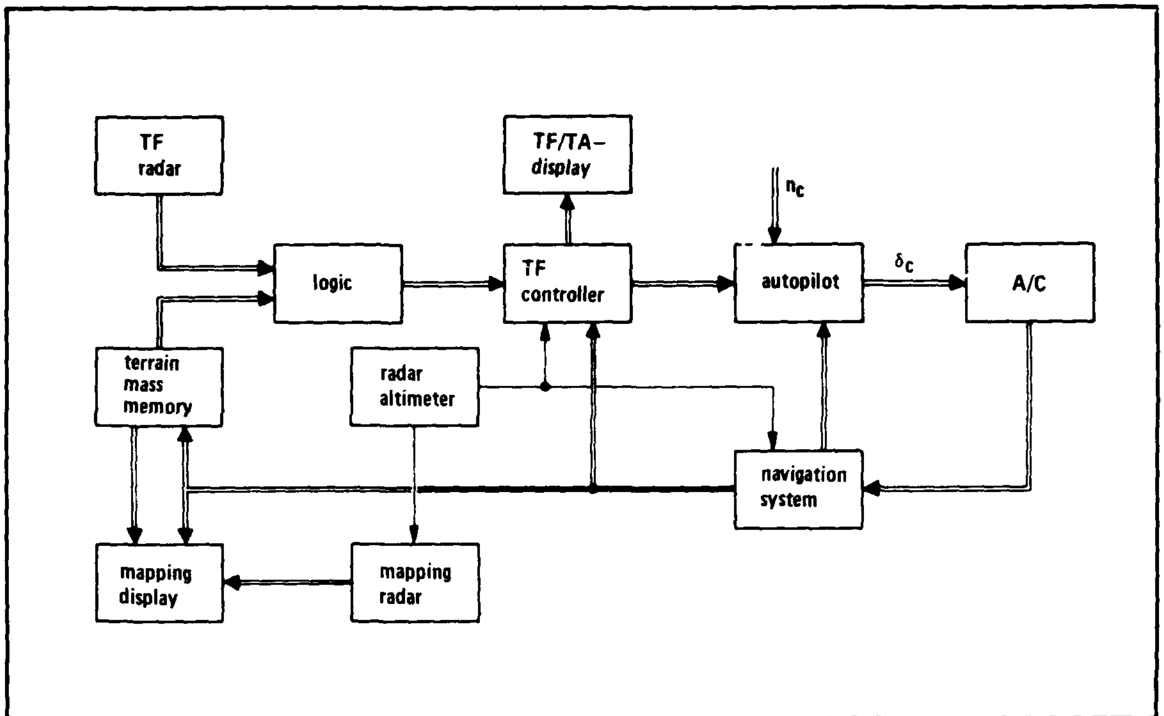
TERPAC	NAVIGATION–SYSTEM: DATA FLOW	VFW Fig.4



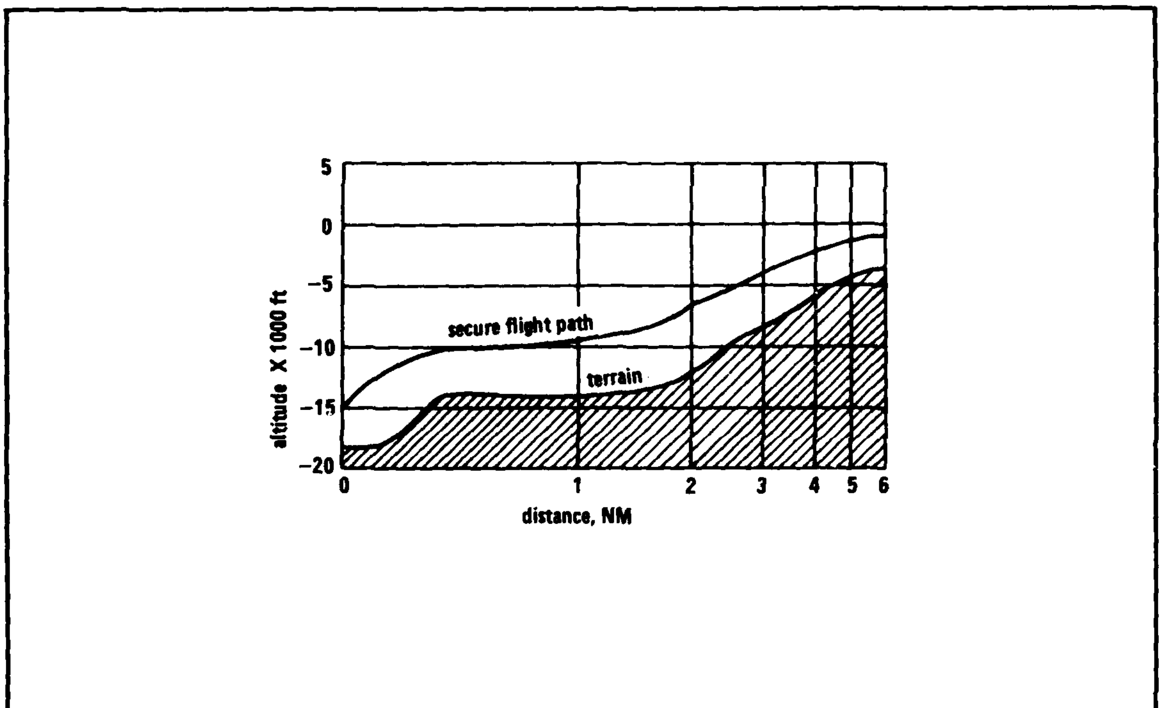
TERPAC	TERRAIN–PROFILE–MEASUREMENT	VFW Fig.5



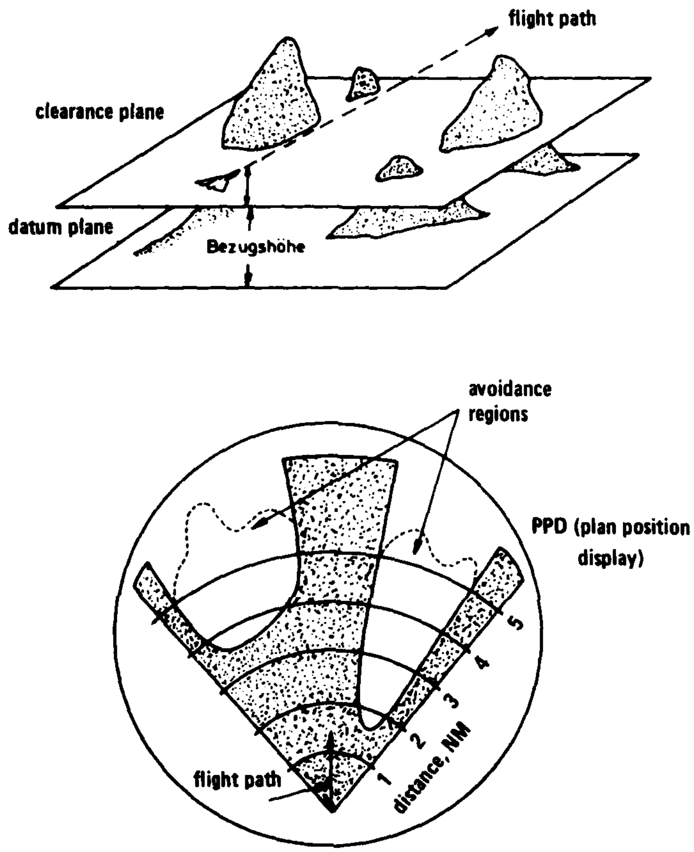
TERPAC	ALTITUDE DATA MEASUREMENT/SIGNAL PROCESSING	VFW Fig.6



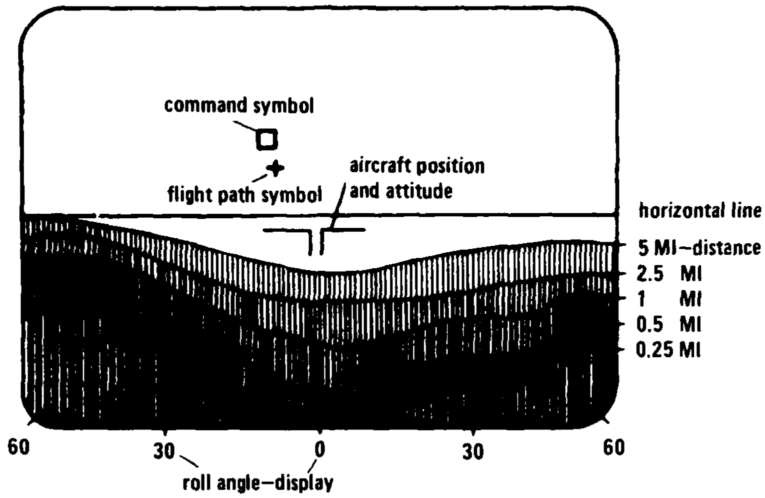
TERPAC	TERRAIN AIDED TF/TA–FLIGHT–CONTROL SYSTEM	VFW Fig.7



TERPAC	E–SCOPE–DISPLAY FOR TF–CONTROL	VFW Fig.8



TERPAC	PLANE DISPLAY FOR MANUAL OR AUTOMATIC TA–CONTROL	VFW Fig.9

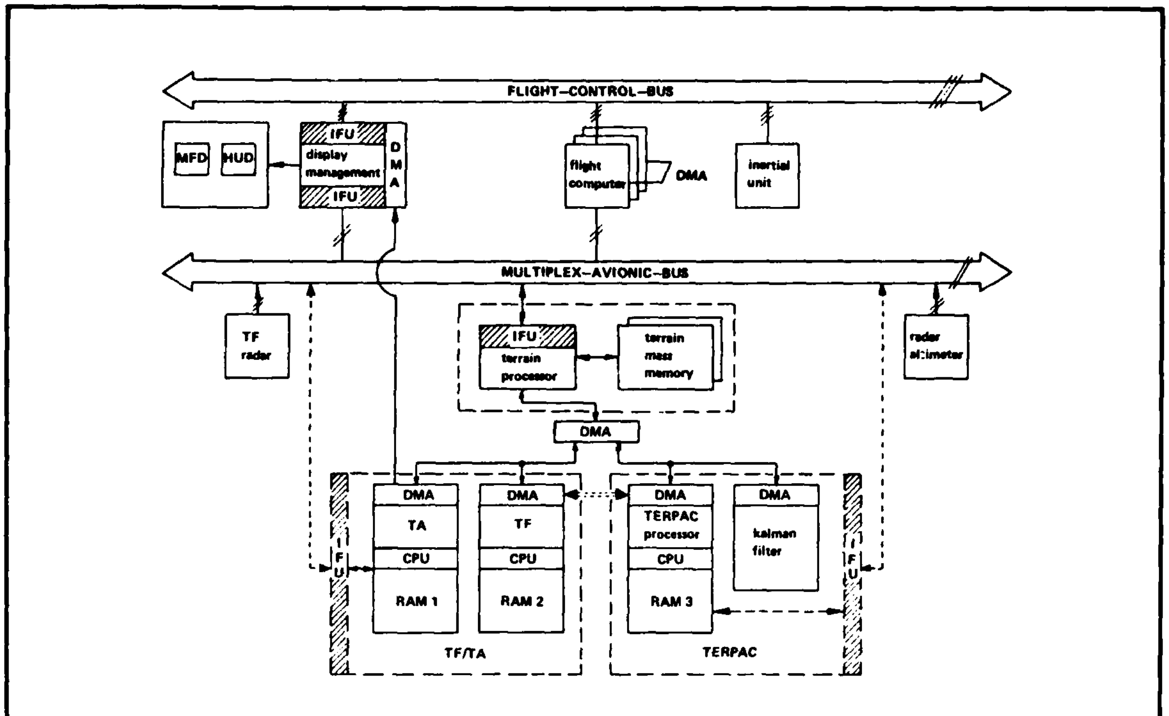


TERPAC	GRAY–SCALE DISPLAY FOR TF/TA–CONTROL	VFW Fig.10

ADVANTAGES

- back–up–mode of TF–radar–system (shadowing)
- stand–alone mode during operational phases of silence
- reliable recognition of obstacles
- reduced maximum g–louds
- soft ride commands
- reduced detectability and ECM–vulnerability
- optimal flight–path–approximation against terrain (valleys)

TERPAC	TF/TA–FLIGHT–CONTROL–SYSTEM BASED ON STORED TERRAIN DATA	VFW Fig.11



TERPAC	COMBINED NAVIGATION–TF/TA FLIGHT CONTROL SYSTEM BASED ON STORED TERRAIN DATA (BUS ARCHITECTURE)	VFW Fig.12

LA NAVIGATION OPTIMALE INERTIE-CORRELATION D'ALTITUDE
UNE SOLUTION ATTRAYANTE AU PROBLEME DE LA NAVIGATION POUR AVION D'ATTAQUE AU SOL

par
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(présenté au 33ème symposium du GCP AGARD en octobre 1981 à Athènes)

RESUME

Cet exposé concerne l'application du recalage de l'inertie par corrélation d'altitude aux avions de pénétration et d'attaque au sol tout temps. Il est basé sur une étude en cours, sur contrat du gouvernement français. Il rappelle brièvement les principes du positionnement par corrélation d'altitude. Il analyse les paramètres influant sur la précision du point. Pour l'application sur avion d'attaque au sol, des algorithmes spéciaux omnidirectionnels ont été conçus pour permettre une grande liberté en cap au-dessus de la zone de recalage. L'exposé résume ensuite les principales caractéristiques d'utilisation, intéressantes pour l'avion de pénétration et d'attaque au sol tout temps : recalage en 3 dimensions, précision élevée, autonomie, discrétion, insensibilité au brouillage, automaticité, faibles contraintes de pilotage et, dans le cas du recalage omnidirectionnel, souplesse de définition et de modification de la mission. Il rappelle également brièvement le principe et les avantages du recalage optimal de l'inertie par filtrage de KALMAN, à partir de la corrélation d'altitude. Les améliorations qui en découlent pour la position et la vitesse peuvent être essentielles pour un suivi de terrain précis et pour la mise en oeuvre des armes, l'initialisation et l'alignement en vol de missiles tactiques.
Il montre la possibilité et l'intérêt, technique et économique, de pouvoir intégrer dans l'unité inertielle les traitements de l'inertie pure, de la corrélation, du filtrage optimal et même de l'attaque air/sol. Il décrit le scénario de mise en oeuvre d'une telle solution au cours d'une mission d'attaque au sol tout temps. Il présente d'abord la préparation de la mission avec sélection des terrains immédiatement avant vol ou à partir d'un fichier de terrains préparés à l'avance et couvrant tout un théâtre d'opération. Il montre l'intérêt de l'emploi sur l'avion d'une mémoire à cassette, de type à bulles magnétiques par exemple. Il présente ensuite le déroulement de la mission proprement dite et la souplesse de modification possible.

OPTIMAL INERTIAL NAVIGATION USING TERRAIN CORRELATION
AN ATTRACTIVE SOLUTION TO THE GROUND ATTACK AIRCRAFT NAVIGATION PROBLEM

SUMMARY

The subject of this paper is the use of terrain correlation for all weather penetration and ground attack aircraft. It is based on a contract from the French government. Positioning by terrain correlation is briefly reviewed. For the ground-attack-aircraft application special algorithms allow large heading freedom over the updating area. The parameter sensitivity of the updating accuracy is analysed. This includes the terrain characteristics (and hence the extensive terrain selection procedure required), the cartography accuracy and the altitude measurement accuracy. The main operational features for ground attack aircraft are summarized : tri-dimensional updating ; high accuracy facilitated by low altitude flight ; self-contained, secure, jam-resistant, automated operation ; a low level of heading constraint to and over the updating area ; savings in weight, volume, electrical power and cost using sensors already needed on the aircraft (i.e. INS, and radio and barometric altimeters) ; moderate level of sophistication and hence reliability. The only specific need is a cassette memory of intermediate size. The principle of inertial system optimal updating is also briefly discussed. Temporal carryover from position and velocity updates can be crucial for accurate horizontal terrain following and weapon delivery initialization and in-flight alignment of tactical missiles. The possibility and the technical and economical interest of integrating correlation and optimal filtering in the inertial unit are shown. A typical all weather ground attack mission scenario is described. This includes the extensive mission preparation, the nominal flight plan and possible alternatives, update area selection using ground based equipment and map data bases, and the data assembly and cassette loading. Mission preparation can be eased and shortened by using files of adequate update areas which have been preselected and cover all the theater of operation and by using mass memory in the aircraft (such as magnetic bubble memory). The mission execution itself includes the nominal flight plan execution, the automatic update warning, execution and control, possible flight plan modification, or even an update of opportunity. A possible operational system architecture is described with the necessary equipment.

1. INTRODUCTION

Le recalage de l'inertie par corrélation d'altitude est connu dans son utilisation sur les missiles de croisière américains. Cet exposé concerne son application aux avions de pénétration et d'attaque au sol tout temps.

La nécessité de pouvoir réaliser des missions de pénétration et d'attaque au sol tout temps en territoire hostile s'impose de plus en plus aux états-majors. Le perfectionnement des moyens de défense aérienne et de guerre électronique exige pour la survie que cette pénétration se fasse en suivi de terrain à l'altitude la plus basse possible et à la vitesse la plus élevée possible, et qu'elle soit précise, autonome et discrète. Enfin, la mise en oeuvre des armes sur l'objectif doit également être précise.

Ce type de mission est donc très exigeant car il accumule les difficultés de la survie aux défenses aériennes ennemies et de la sécurité du vol en suivi de terrain tout temps. Depuis quelques années, les progrès techniques permettent de réaliser des systèmes qui répondent plus ou moins bien à chacun de ces besoins. Les avions utilisés sont au moins bi-places, à cause de la charge de travail. Ils sont généralement bi-réacteurs. La quantité d'électronique embarquée est grande. Tel est le cas des avions de pénétration tout temps F111, TORNADO.

Pour la prochaine génération, compte tenu des progrès techniques et technologiques, les état-majors souhaitent pouvoir utiliser des avions de taille et de coût plus modérés, c'est-à-dire des monoplaces, monoréacteurs. Ceci suppose une réduction de la charge de travail, donc une automaticité plus grande, ainsi qu'un volume d'électronique plus faible. Ils souhaitent aussi des progrès en discrétion, car la survie est essentielle, mais aussi en fiabilité et en précision de navigation.

L'objet de cet exposé est de montrer qu'un système intégré de navigation optimale inertie-corrélation d'altitude pourrait être un élément de réponse à ces objectifs ambitieux, en ce qui concerne la discrétion, la précision, l'autonomie, la souplesse et l'automaticité de la mise en oeuvre, la fiabilité et l'économie de matériel.

Cet exposé s'appuie sur un contrat du Service Technique des Télécommunications et des Equipements Aéronautiques (STTE) du gouvernement français dont l'objet est l'étude et l'expérimentation en vol d'un tel système pour avion de pénétration et d'attaque au sol tout temps.

2. PRINCIPE DE POSITIONNEMENT PAR CORRELATION D'ALTITUDE

2.1 Généralités (Voir planche N° 1)

Le principe du positionnement par corrélation d'altitude consiste à relever sur quelques kilomètres le profil de l'altitude du terrain survolé et son orientation dans le plan géographique, puis à rechercher le profil correspondant sur une carte d'altitude de la zone concernée. La carte est stockée sous forme numérique dans une mémoire embarquée. Si le profil relevé était sans erreur, il coïnciderait exactement avec celui de la carte en mémoire, à condition que la position indiquée par l'inertie soit également sans erreur. En réalité, l'erreur de l'inertie n'étant pas nulle, pour trouver un profil identique au profil relevé, il est nécessaire de faire une certaine translation dans le plan horizontal. Cette translation que l'on peut mesurer correspond évidemment à l'erreur de position de l'inertie et le profil identifié en mémoire donne la position vraie.

En pratique, la carte d'altitude de la zone survolée est disponible en mémoire point par point, selon un maillage régulier caractérisé par la taille de sa "cellule" élémentaire, voir figure 1. L'altitude du terrain survolé est calculée par différence entre l'altitude "absolue" baro-inertielle, fournie par l'inertie, et l'altitude relative de l'avion par rapport au terrain, fournie par la radio-sonde, voir figures 2et 3. L'altitude du terrain survolé est mémorisée au fur et à mesure sous forme de points régulièrement espacés. A chaque point sont associés les coordonnées horizontales données par l'inertie. La recherche du profil identique de la carte en mémoire est réalisée par calcul du degré de coïncidence, ou de "corrélation", du profil relevé avec tous les profils parallèles de la carte, obtenus par des translations successives dans les deux dimensions. A chaque translation, le calcul fait correspondre une valeur de corrélation. A l'ensemble correspond une surface de corrélation. La recherche du maximum de cette surface fournit les deux composantes de la position vraie. Comme on le verra au paragraphe 6.2, il est possible de déterminer une indication du degré de confiance de cette position "vraie", très utile pour le recalage optimal de l'inertie.

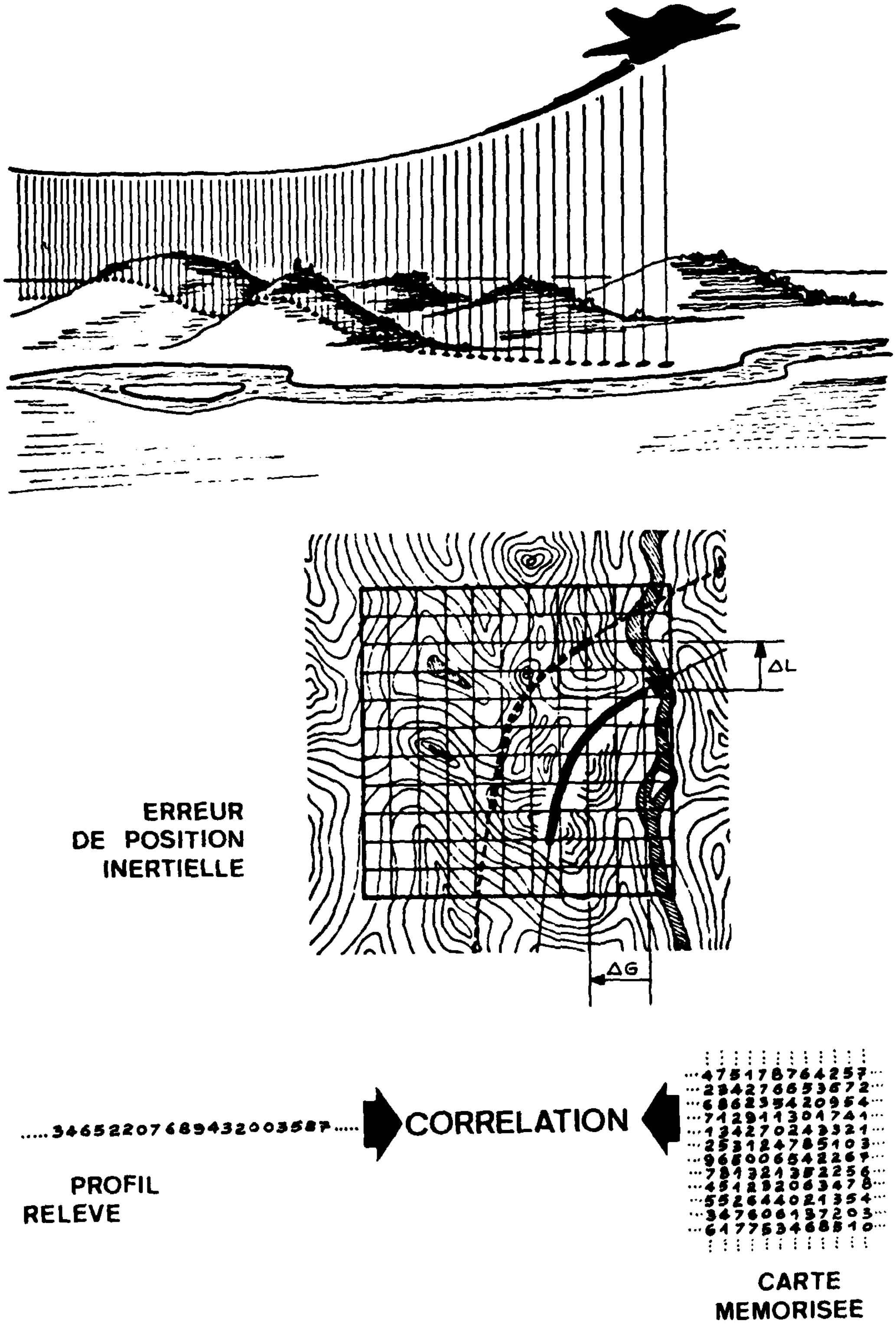


Figure 1 : PRINCIPE DU POSITIONNEMENT PAR CORRÉLATION D'ALTITUDE

2.2 Corrélation d'altitude omnidirectionnelle

Dans le cas d'une trajectoire préprogrammée, la zône de recalage peut être rectangulaire et orientée selon la route désirée prévue au départ. Son survol est réalisé en ligne droite. Sa largeur correspond à l'erreur de navigation maximale possible, qui dépend de l'inertie et du temps écoulé depuis le précédent recalage. La longueur est au minimum égale à la longueur de corrélation nécessaire pour déterminer, sans ambiguité, le profil de la carte identique à celui relevé, voir paragraphe 4.1.

Cette définition de la zone de recalage peut convenir aux missions de pénétration entièrement préprogrammées, dont la préparation est réalisée à l'avance dans le détail. Cependant, sur un avion piloté, il peut être souhaitable de disposer de plus de liberté dans le choix de l'axe d'approche de la zone ou dans les évolutions possibles sur la zone. Ceci peut être souhaitable par exemple, pour permettre en vol la modification de la mission face à des circonstances imprévues, ou la prise en compte d'objectifs d'opportunité, et pour simplifier au sol la préparation de la mission.

Un algorithme a été spécialement développé pour répondre à ces besoins. Il autorise le survol de la zone à une route quelconque et selon des évolutions importantes. La zone mémorisée est alors carrée comme le montre la figure 4. La surface à mémoriser et donc le volume mémoire nécessaire augmentent dans ce cas d'un facteur de l'ordre de 2,3 ce qui ne pose pas de difficulté particulière avec les dernières technologies de mémoires.

3. PRINCIPALES CARACTERISTIQUES DU RECALAGE PAR CORRELATION D'ALTITUDE

Le recalage par corrélation d'altitude présente un certain nombre de particularités intéressantes. Tout d'abord, il permet un recalage de la position dans les <u>3 dimensions.</u> En effet, en plus du positionnement précis dans le plan horizontal,connaissant l'altitude du terrain survolé, on peut reconstituer l'altitude de survol de l'avion, à partir de la hauteur radio-sonde. La précision du recalage de l'altitude est celle de la radio-sonde à l'altitude de survol, entâchée de l'erreur de cartographie au point considéré. Un autre aspect intéressant concerne l'<u>autonomie</u> du recalage. Aucun équipement extérieur à l'avion n'est nécessaire : toutes les informations nécessaires au recalage, y compris la carte du terrain survolé, sont contenues dans l'avion. Sur le plan de la <u>discrétion</u>, la radio-sonde est considérée comme difficilement détectable à partir du sol, en raison de sa faible portée dans le plan horizontal (émission dirigée vers le bas). Il en résulte aussi une grande <u>insensibilité au brouillage</u>, de même pour le baro-altimètre et la centrale inertielle.
Sur le plan opérationnel, on peut mettre en évidence un certain nombre de qualités particulières : simplicité de fonctionnement, grande liberté de survol du terrain et facilité de mise en oeuvre. Le fonctionnement du recalage en vol ne nécessite aucun mode particulier de la part de l'inertie, du baro-altimètre, de la radio-sonde, ni des autres équipements avion. La sécurité de la mission et du vol basse altitude en suivi de terrain est renforcée. Le mode de recalage par corrélation d'altitude peut donc être mis en oeuvre dans toutes les phases de la mission. Le survol du terrain sélectionné pour le recalage ne nécessite pas <u>de contraintes particulières de pilotage</u>, ni en cap d'entrée sur la zone, ni en tenue d'altitude, ni en évolution. Une grande liberté de manoeuvre peut ainsi être laissée au pilote. La mise en oeuvre est relativement simple : en vol, le <u>fonctionnement</u> est entièrement <u>automatique</u> et la préparation du vol se résume au choix des zones de recalage (voir paragraphe 6.1). Enfin, la <u>précision</u> du recalage est <u>excellente</u> : 70 m CEP en utilisant une carte numérisée au pas de 100 m et potentiellement mieux encore dès qu'une numérisation plus fine des cartes le permettra. Cette précision est obtenue dans le cas de missions à très basse altitude. Dans les autres cas, la précision va en diminuant quand l'altitude augmente.

Enfin, la banque de données utilisée pour la préparation de la mission est relativement simple puisqu'elle consiste en cartes mémorisées d'altitude. Elle ne nécessite <u>pas de traitements particuliers en fonction du cap ni de l'altitude d'approche</u>. Il n'y a pratiquement <u>pas de problème de mise à jour</u>, car la corrélation d'altitude utilise des caractériques globables du terrain qui sont permanentes et non des points remarquables, tels que des constructions, qui peuvent évoluer ou être vulnérables.

En résumé, cette technique de recalage offre un grand nombre d'avantages par rapport aux autres moyens de recalage possibles et s'avère particulièrement bien adaptée aux avions d'armes modernes, pour les missions de pénétration très basse altitude et d'attaque au sol tout temps.

4. PARAMETRES INFLUANT SUR LA PRECISION DU POINT

Les paramètres qui interviennent dans la précision du point peuvent être rattachés aux principaux éléments suivants :

- le terrain,
- le relevé d'altitude en vol,
- la carte d'altitude en mémoire.

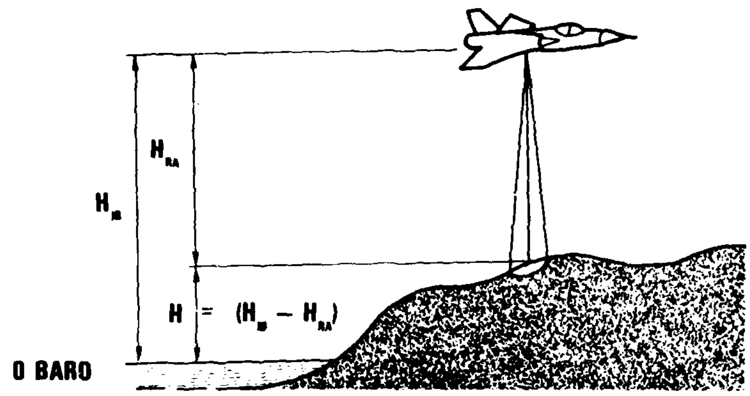


Figure 2 : CALCUL DE L'ALTITUDE DU TERRAIN PAR DIFFÉRENCE ENTRE L'ALTITUDE "ABSOLUE" INERTIE-BARO ET L'ALTITUDE RADIO-ALTIMÈTRE

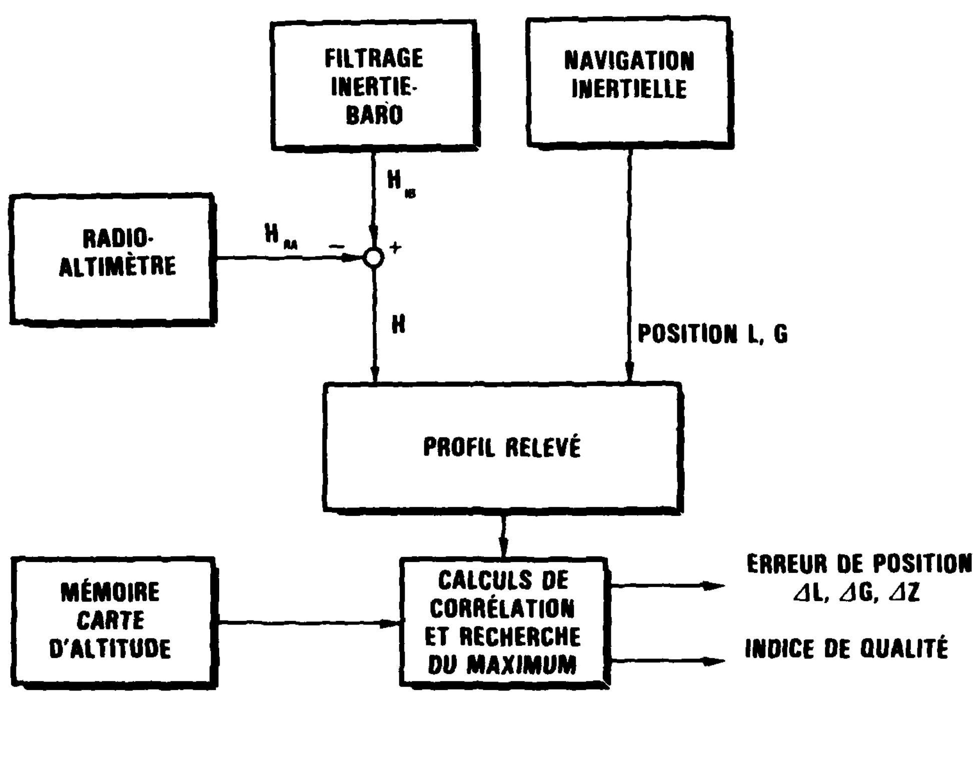


Figure 3 : SCHÉMA FONCTIONNEL DU POSITIONNEMENT PAR CORRÉLATION D'ALTITUDE

4.1 Le terrain

En première analyse ce sont les caractéristiques spectrales (spatiales) des mouvements du terrain qui comptent. Pour que la précision du point soit inférieure ou égale à une cellule, qualitativement :

- l'amplitude des mouvements du terrain doit être significative, c'est à dire nettement supérieure à la précision et à la résolution du relevé (radio-sonde et inertie) et à celles de la carte en mémoire,
- la longueur L du profil à corréler doit être nettement supérieure aux longueurs d'onde λ des mouvements significatifs du terrain,
- les longueurs d'onde λ des mouvements significatifs du terrain doivent être nettement supérieures à deux fois la taille de la cellule (théorème de Shannon).

En résumé, il est nécessaire que :

$$L \gg \lambda \gg 2 \text{ cellules}$$

Pour éviter les faux recalages, c'est à dire ceux dont la précision est supérieure à une cellule, il faut également que le terrain ne contienne pas de profils identiques distants de plus d'une cellule.

Une sélection complète du terrain est donc essentielle pour garantir les performances du calcul du point par corrélation d'altitude. Cette sélection nécessite des traitements numériques et doit être réalisée à l'aide d'un système informatique adapté, voir paragraphe 7.

Cette première analyse a écarté un certain nombre d'aspects du terrain tels que nature de sol, végétation, constructions, qui sont analysés au paragraphe 4.2 par leurs effets sur l'erreur du radio-altimètre.

4.2 Le relevé d'altitude en vol

La précision du relevé d'altitude dépend dans le plan vertical de celles du radio-altimètre et du filtrage inertie-baro. Dans le plan horizontal elle dépend de la précision de l'inertie.

. Radio-altimètre

En théorie le radio-altimètre mesure la distance la plus courte dans le champ de l'antenne entre l'avion et le sol, mais en pratique il intègre grossièrement la surface couverte par son lobe. Une dimension de cellule inférieure à la surface au sol couverte par le lobe du radio-altimètre est donc inutilisable. La précision d'un radio-altimètre est schématiquement fonction de la nature du sol survolé, de l'altitude, de l'attitude avion, de l'ouverture d'antenne et de la vitesse avion. Le vol à très basse altitude est favorable à une bonne précision. Le modèle du radio-altimètre est représenté généralement par une erreur de facteur d'échelle, supposée constante dans le temps et seulement dépendante de la nature du terrain survolé.

. Filtrage inertie-baro

Un filtrage inertie-baro délivre l'équivalent d'une altitude absolue, décalée de l'erreur barométrique à long terme. Son excellente résolution, son excellente précision à court terme et sa bande passante élevée, par rapport à la dynamique avion, sont l'apport de l'accéléromètre vertical de l'inertie. Ces caractéristiques remarquables sont très utiles pour mesurer de façon précise les variations instantanées d'altitude de l'avion en suivi de terrain. En pratique les erreurs instantanées d'un filtrage inertie-baro performant sont négligeables pour la corrélation d'altitude, de même que son erreur à long terme.

. Inertie

L'inertie fournit la position estimée du profil et sa restitution dans le plan horizontal. L'erreur de restitution est une erreur de mesure supplémentaire pour la corrélation. Elle dépend de l'erreur de vitesse géographique et de la durée T du relevé corrélé, ou encore de la vitesse vectorielle relative et de la longueur L du profil corrélé :

$$\text{Erreur de restitution} = |\delta\vec{V}|\ T = \frac{|\delta\vec{V}|}{V} \cdot L = \frac{|\delta\vec{V}|}{V} \cdot N.C.$$

où N est la longueur corrélée, exprimée en nombre de cellules.

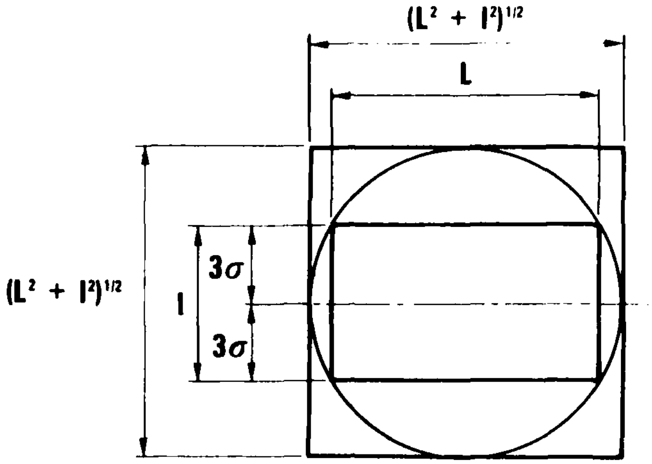


Figure 4 : SURFACE DES ZONES POUR CORRÉLATION D'ALTITUDE "MONODIRECTIONNELLE" ET "OMNIDIRECTIONNELLE"

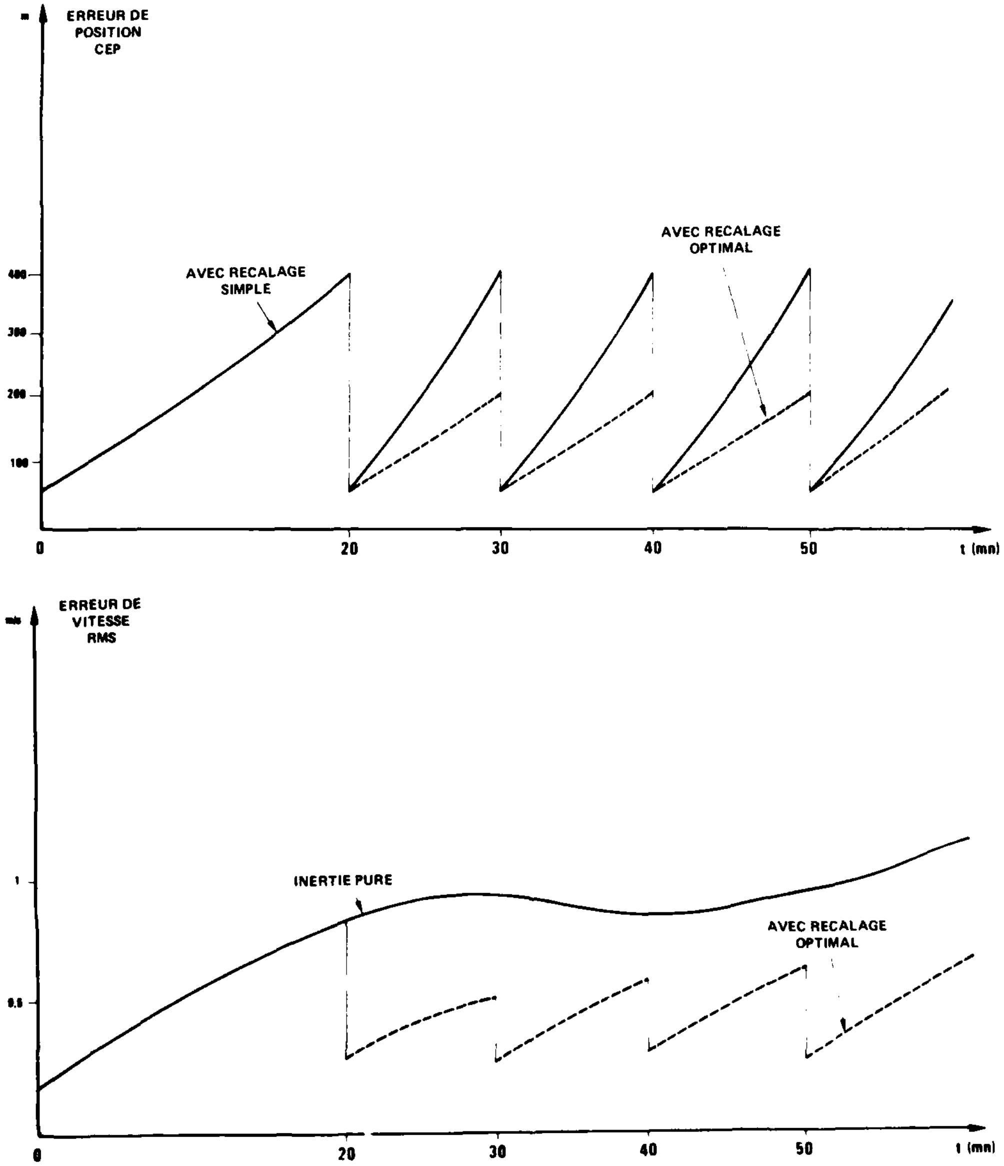


Figure 5 : PERFORMANCE DU FILTRE OPTIMAL DANS LE CAS D'UNE MISSION TYPE DE PÉNÉTRATION ET D'ATTAQUE AU SOL INERTIE - CORRÉLATION D'ALTITUDE

C est la dimension d'une cellule.

Pour garantir une erreur de restitution inférieure à une cellule, il faut vérifier que :

$$|\overrightarrow{\delta V}| \quad \frac{V}{N}$$

Pour N = 50 par exemple et une vitesse de 400 noeuds, la précision de la vitesse Nord et Est doit être meilleure que 4 m/s, c'est à dire de l'ordre de 4 m/s à 2 ou 3 . Une telle précision correspond à un système inertiel de bonne performance, c'est à dire de la classe 1NM/H CEP et 1 m/s à 1 .

4.3 La carte d'altitude en mémoire

Les caractéristiques importantes de la carte d'altitude en mémoire sont la précision de l'altitude au point considéré et la dimension de la cellule.

. Précision de la carte d'altitude

En pratique la carte est réalisée, soit à partir de cartes à grande échelle existantes, soit par photogrammétrie à partir de photographies aériennes ou satellites. On peut noter que l'utilisation fait surtout intervenir les altitudes différentielles, ce qui est favorable, et qu'une précision supérieure à celle du relevé en vol est inutile.

. Dimension de la cellule

La taille de la cellule conditionne directement la résolution du point donc sa précision. La précision théorique possible est d'environ 0,7 fois la dimension de la cellule. Compte-tenu de la résolution des cartes actuellement disponible et de celle de la radio-sonde, une cellule de 50 à 100 mètres est, aujourd'hui, la mieux adaptée au recalage précis à très basse altitude.

5. NAVIGATION OPTIMALE *INERTIE-CORRELATION* D'ALTITUDE

5.1 Principe

L'utilisation la plus efficace de l'information donnée par la corrélation d'altitude consiste à réaliser une hybridation par filtrage de Kalman avec l'inertie. Les erreurs de position dans les trois dimensions, fournies par la corrélation, sont traitées comme des observations de l'erreur de position de l'inertie. Le filtre en déduit une estimation "optimale" des erreurs inertielles. Cette estimation est utilisée pour recaler l'inertie.

L'efficacité et les performances de cette hybridation sont intéressantes à cause de la possibilité de définir un assez bon modèle des erreurs de l'inertie, des caractéristiques très basse fréquence de l'évolution de ces erreurs, et de la grande précision des points de corrélation d'altitude, même si la fréquence de ces points est relativement faible.

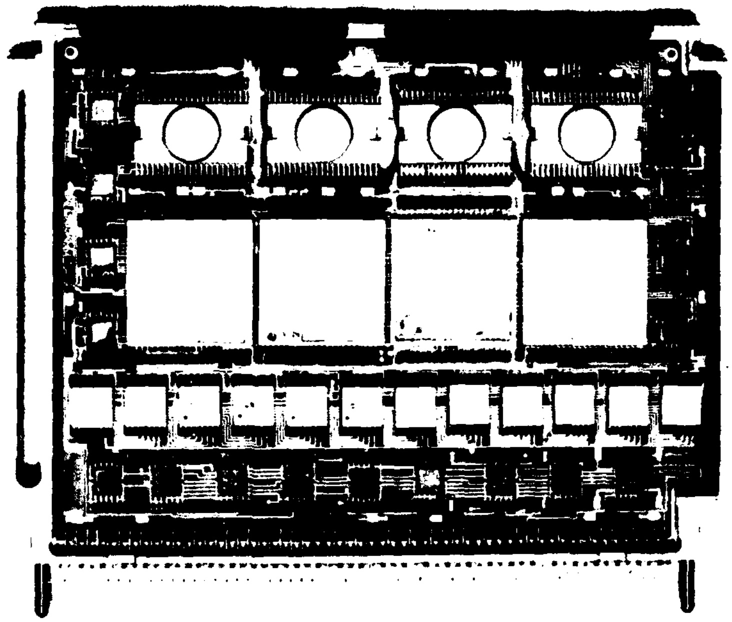
La figure 5 montre un exemple de performances simulées. Elle indique, pour une mission de pénétration type, l'évolution de précision de position et de vitesse, au fur et à mesure de quatre recalages par corrélation d'altitude, de 70 m CEP de précision, espacés de 10 minutes, et dans le cas d'une inertie de la classe 1 NM/H CEP. L'amélioration de la précision de la position, mais aussi de la vitesse, obtenue par la navigation optimale inertie-corrélation d'altitude est un élément important pour la réalisation précise tout temps de la pénétration, de la mise en oeuvre des armes conventionnelles ainsi que de l'initialisation et de l'alignement des missiles tactiques.

5.2 Intégration dans le système inertiel

Les fonctions :

- navigation inertie pure dans le plan horizontal,
- inertie-baro dans le plan vertical,
- corrélation d'altitude,
- filtrage de Kalman,

sont très dépendantes les unes des autres. Elles échangent beaucoup d'informations entre elles, mais globalement peu avec l'extérieur. Certains de ces échanges sont bouclés ; ils sont généralement assortis de contraintes temps réel assez sévères liées à la dynamique avion d'armes.



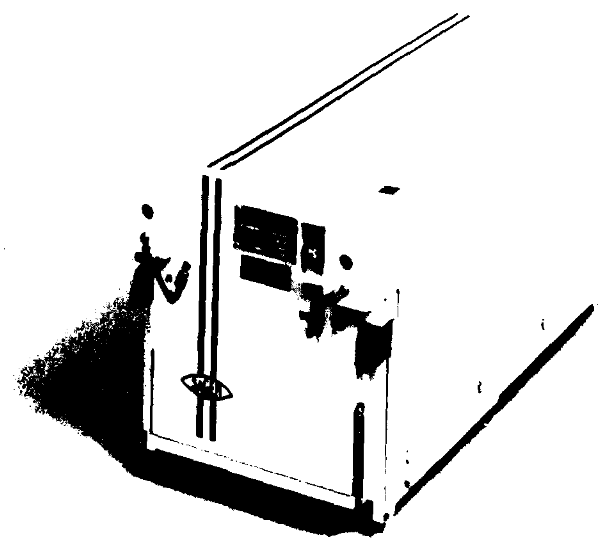


Figure 6 : MICROCALCULATEUR UT 382-50 DE LA CLASSE 400.000 OPÉRATIONS PAR SECONDE, EN UNE SEULE CARTE, ET UNITÉ DE NAVIGATION INERTIELLE MULTI-FONCTIONS DU SYSTÈME ULISS

Il est donc naturel que les traitements correspondants soient réalisés dans le même calculateur, sans oublier que les charges de calcul et les volumes mémoires sont relativement significatifs.

Or la définition moderne des fonctions et des performances d'un système de navigation inertiel pour avion d'armes demande dans le système un calculateur assez puissant, à cause de la complexité des traitements nécessaires. La technologie électronique disponible permet de couvrir largement ces besoins, sans pénaliser le volume du système.

Ainsi l'unité inertielle multi-fonctions du système ULISS possède un micro-calculateur rapide, de la classe 400.000 opérations par seconde, en une seule carte, et une capacité mémoire de 64.000 mots de 16 bits. Cette unité inertielle est compacte puisque son volume est inférieure à 16 litres et son poids inférieur à 16 kilogrammes, voir figure 6. La charge de calcul et le volume mémoire disponibles permettent d'y intégrer sans difficulté la corrélation d'altitude et le filtrage de Kalman.

D'autres intégrations fonctionnelles ont déjà été réalisées dans le système ULISS, telles que navigation inertielle, calculs aérodynamiques, gestion de bus multiplexé 1 Mégabit, ainsi que l'attaque air/sol, bombe, canon, roquette, missile. Cette dernière fonction est également très liée aux informations inertielles de position, vitesse et attitude pour les calculs de balistique, de présentation et d'acquisition dans le viseur, de pointage radar ou télémètre. Compte tenu des progrès électroniques, il est raisonable de dire que, dans peu d'années, il sera possible d'intégrer dans un système type ULISS l'ensemble de ces fonctions plus la corrélation d'altitude et le filtrage de Kalman, comme le suggère la figure 7. Une telle intégration pourrait être une bonne solution pour les futurs avions d'attaque au sol tout temps, tant sur le plan opérationnel que sur le plan de l'économie de matériel.

6. DESCRIPTION DE LA MISE EN OEUVRE OPERATIONNELLE

Ce paragraphe décrit de façon fonctionnelle la mise en oeuvre du recalage de la navigation par corrélation d'altitude pour une mission-type d'avion de pénétration ou d'attaque au sol tout temps.

6.1 Préparation de la mission

La mission-type d'un avion de pénétration peut être définie de la façon suivante: enchaînement de plusieurs segments de navigation (15 par exemple), chacun de ces segments étant défini par le but à atteindre en fin de segment (voir figure 8). On peut envisager un objectif principal (but n° 15), un objectif secondaire (but n° 15') avec choix de l'objectif après le 7ème segment. Il faut envisager aussi un déroutement possible après le 10ème segment par exemple et retour vers le but n° 14". On pourrait imaginer un plus grand nombre de trajectoires possibles pour une même mission principale, avec un plus grand nombre de déroutements possibles par exemple.

A partir du plan de vol ainsi défini, il s'agit de définir les zones de recalage le long des trajectoires possibles de pénétration, en respectant un intervalle de temps déterminé (7 à 12 mn par exemple) entre les recalages et en choisissant le dernier point de recalage le plus près possible de l'objectif à atteindre (but n° 15 ou 15'). Un nombre typique de zones de recalage est de 4 pour une trajectoire : zones (1) à (4) pour la mission principale, zones (1), (5), (6), (7) pour la mission secondaire et zone (8) pour le déroutement possible. L'ordre de grandeur de 8 zones de recalage pour une mission-type est donné purement à titre indicatif. Pour assurer une grande sécurité de navigation et obtenir une bonne performance de recalage, il est nécessaire d'effectuer le meilleur choix possible de ces zones de recalage. La liberté de choix est possible le long de chaque segment de navigation. Les critères de choix sont essentiellement : relief suffisant, végétation non gênante (forêts par exemple), pas de constructions importantes (villages, viaducs, chateaux d'eau, etc ...). L'application de ces critères de choix montre qu'il ne présente pas de contrainte particulière pour des opérations en Europe par exemple. Cette affirmation a été confirmée au cours de vols expérimentaux sur le territoire français.

La précision et la "fiabilité" du recalage dépendant directement de la qualité de la sélection du terrain, en pratique, les opérations de sélection de la zone de recalage consistent à analyser les paramètres statistiques du terrain ainsi que l'impossibilité de rencontrer de faux recalages. Ces opérations nécessitent un traitement informatique élaboré, réalisé sur un système de préparation au sol.

Dans ce qui précède, la sélection de terrain a été supposée faite dans le cadre [illegible] préparation de la mission juste avant le vol. Cependant, la préparation de mis[illegible] pourrait être allégée par l'utilisation d'un fichier de zones de recalage convena[illegible] préparé à l'avance et couvrant l'équivalent de plusieurs missions (5 à 10 par [illegible] même de tout un théâtre d'opérations (100 à 200 zones pour 1.000 km^2 par [illegible] fichier pourrait être préparé à l'avance et stocké dans une cassette, sup[illegible] les opérations de sélection immédiates avant-vol. Les technologies [illegible] de développement permettent de satisfaire ces besoins sans difficulté [illegible] de faible encombrement.

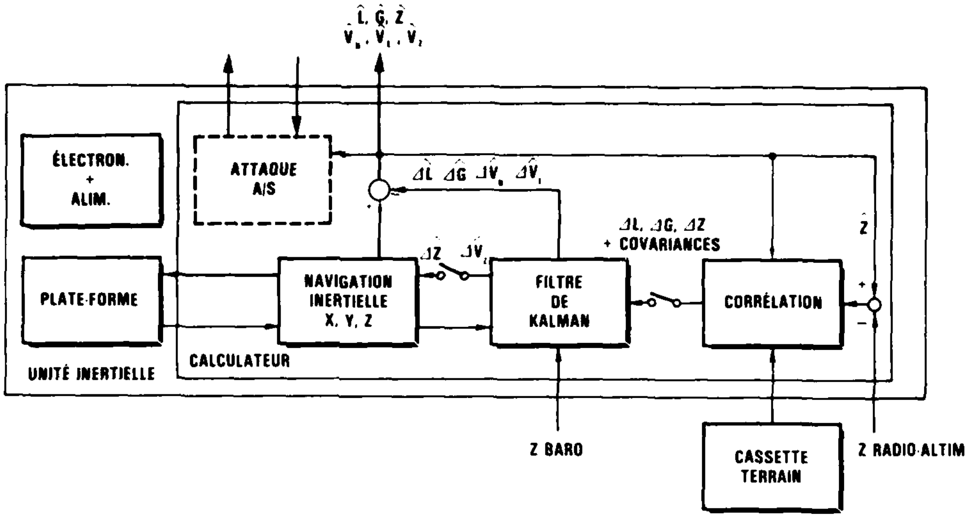


Figure 7 : UNITÉ INERTIELLE MULTI-FONCTIONS INTÉGRANT LA NAVIGATION OPTIMALE INERTIE - CORRÉLATION D'ALTITUDE ET L'ATTAQUE A/S

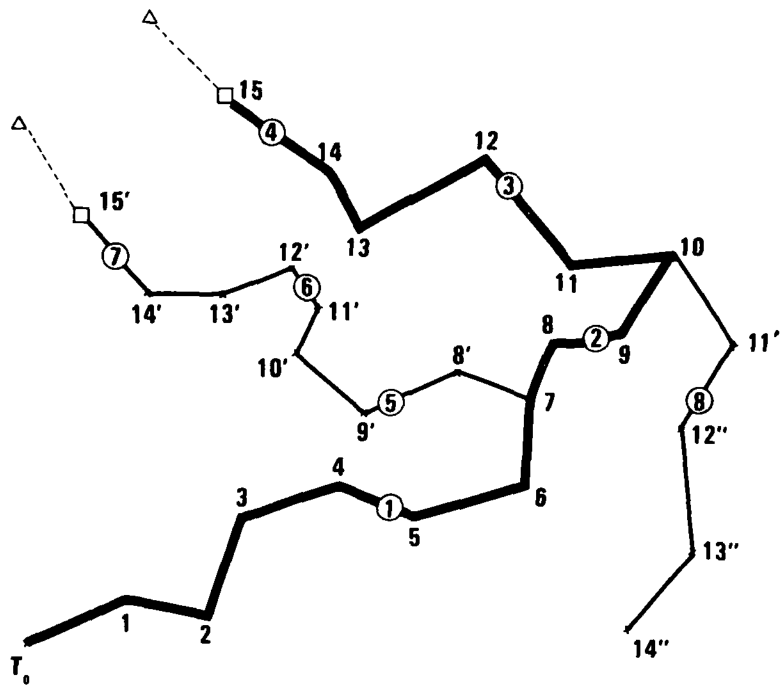


Figure 8 : MISSION - TYPE D'UN AVION DE PÉNÉTRATION :
- MISSION PRINCIPALE : BUTS 1 A 15
- MISSION SECONDAIRE : BUTS 1 A 7 PUIS 8' A 15'
- MISSION AVEC DÉROUTEMENT : BUTS 1 A 10 PUIS 11" A 14"

ET ZONES DE RECALAGE 1 A 8

6.2 Mise en oeuvre du recalage

. Opérations avant survol de la zone

La sélection du mode de recalage par corrélation d'altitude peut se faire dès le départ de la mission. En effet, le déclenchement du processus de recalage est réalisé de façon automatique en fonction du rapprochement de l'avion vers la zone de recalage. L'enchaînement des buts de navigation, sélectionné manuellement ou automatiquement, permet de prévoir s'il y aura ou non passage sur une zone de recalage et de lancer ou non la préparation automatique du processus, en particulier le chargement du terrain numérisé en mémoire de travail. Cette préparation est déclenchée sur le résultat d'un test de la distance avion-zone de recalage.

. Présentation et survol de la zone

La zone de recalage est identifiée par un point de référence qui constitue le point de passage souhaité sur le terrain. La présentation de l'avion sur la zone est visualisée au pilote sur l'indicateur tête haute. Au cours du survol de la zone, aucune contrainte particulière de pilotage n'est nécessaire. Le pilote dispose d'une indication présente tant que l'avion survole la zone, lui signalant que le processus de recalage est en cours.

. Validation du recalage

Une fois les calculs de corrélation effectués, les résultats de recalage sont visualisés en écarts de position sur l'indicateur tête basse ou tête haute. En plus des écarts de position, l'algorithme de corrélation fournit un indice de qualité du recalage qui a l'une des 3 significations suivantes :

- bon recalage (inférieur à 1 cellule),
- recalage moins précis, mais suffisant pour recaler la navigation (inférieur à 2 ou 3 cellules par exemple),
- faux recalage important très probable.

On peut automatiser la prise en compte de l'indice de qualité en acceptant le recalage dans le 1er cas et en le refusant dans le 3ème, la décision de recaler incombant au pilote dans le 2ème cas. On pourrait alléger totalement la charge du pilote et rendre entièrement automatique la prise en compte du recalage (par exemple en refusant le recalage dans le 2ème cas).

7. INTEGRATION DANS LE SYSTEME DE NAVIGATION ET D'ARMES

7.1 Configuration proposée

La figure 9 présente une configuration possible pour avion de pénétration et d'attaque au sol tout temps à l'horizon 85-90. Les principaux équipements sont connectés à un bus multiplexé redondant. Les équipements en soute comprennent :

- un radio-altimètre numérique,
- 2 jeux de capteurs anémométriques,
- un ensemble inertiel redondant constitué de 2 systèmes inertiels multi-fonctions.

Chaque système inertiel multi-fonctions intègre à l'inertie les fonctions :

- calculs anémométriques,
- filtrage inertie-baro,
- corrélation d'altitude,
- filtrage de Kalman,
- attaque Air/Sol,
- gestion bus.

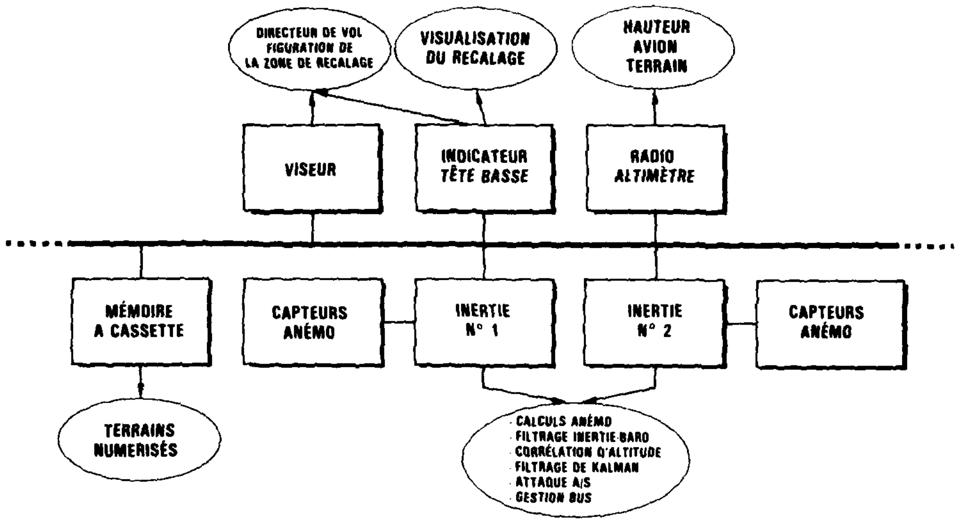


Figure 9 : CONFIGURATION PROPOSÉE POUR AVION DE PÉNÉTRATION
ET D'ATTAQUE AU SOL TOUT TEMPS A L'HORIZON 85-90

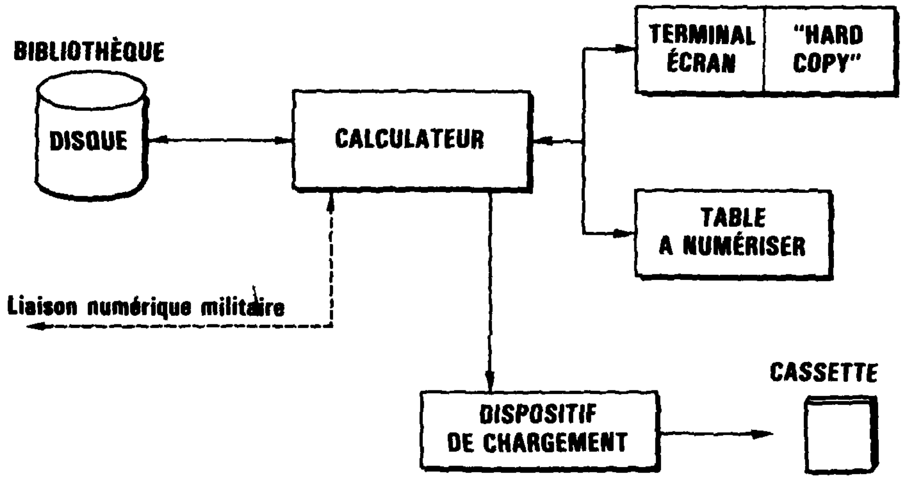


Figure 10 : INSTALLATION SOL POUR LA PRÉPARATION
DE MISSION ET LA SÉLECTION DE TERRAIN

Les équipements en cockpit comprennent :

- un viseur tête haute,

- un viseur tête basse,

- une mémoire à cassette.

Cette configuration montre sur le plan matériel les avantages suivants :

a) Aucun équipement sophistiqué supplémentaire n'est nécessaire mis à part la mémoire à cassette qui peut servir à bien d'autres besoins : plan de vol, armement, maintenance, etc .

b) La possibilité de réaliser une intégration fonctionnelle élevée avec l'inertie.

c) Un bilan réduit de volume, masse, consommation électrique, particulièrement intéressant dans le cas d'un mono-réacteur.

d) Une configuration relativement simple, donc fiable.

7.2 Caractéristiques des équipements pour la corrélation

Ce paragraphe présente les caractéristiques importantes des équipements nécessaires à l'utilisation opérationnelle de la corrélation d'altitude.

. Radio-altimètre

Les principales caractéristiques du radio-altimètre sont la précision de la mesure de hauteur, la sensibilité à la nature du terrain et l'ouverture du diagramme d'antenne. La précision de quelques pour-cents des radio-altimètres modernes est suffisante. Certains radio-altimètres parmi les plus récents, tel que le modèle numérique AHV-12 de TRT, donneraient même une précision de 1 % quelle que soit la nature du terrain. Cette réduction de sensibilité à la nature du terrain est un élément important qui peut contribuer à élargir les critères de sélection de terrain.

. Altitude barométrique

L'altitude barométrique est utilisée par l'inertie pour élaborer l'altitude de référence de l'avion, appelée altitude baro-inertielle. Les performances absolues d'altitude barométrique ne sont pas critiques pour la corrélation d'altitude car son erreur est typiquement un biais, non gênant puisque la grandeur à mesurer est le profil du terrain survolé et non son altitude absolue. La stabilité à court terme du biais d'altitude baro n'est pas gênante non plus puisque les variations d'altitude de l'avion sont mesurées par l'accéléromètre inertiel.

. Viseurs tête haute et tête basse

Il s'agit de présenter au pilote une figuration lui permettant de le renseigner sur la position de l'avion par rapport à la zone de recalage, de voir le résultat de la corrélation et, si nécessaire d'intervenir sur la validation du recalage. Dans le cas de l'avion monoplace, il est peut être préférable d'avoir une validation automatique, avec une indication simple du recalage accepté ou refusé.

. Mémoire à cassettes

Le volume mémoire nécessaire correspond au nombre maximal de zones de recalage prévu pour une mission. En se référant à la mission-type décrite au paragraphe 6.1, il faut mémoriser au moins une dizaine de zones. En considérant un besoin de 3 K mots de 16 bits, soit 48 K bits par zone omnidirectionnelle, le besoin de mémoire est évalué à 0,5 M bits environ par mission. Des mémoires à bulles magnétiques utilisées sous forme de petites cassettes de 16 M bits sont en cours de développement. Leur capacité permettrait de contenir l'équivalent de plusieurs missions ou même de tout un théâtre d'opérations, (10 M bits pour 200 zones).

. Système à inertie

Les caractéristiques nécessaires pour une corrélation précise d'altitude ont été abordées au paragraphe 4.2 et correspondent à la classe de performance 1 NM/h et 1 m/s.

8. INSTALLATION POUR LA PREPARATION DE MISSION

La préparation de la mission est réalisée au sol avec une installation particulière. A partir du plan de vol, il s'agit de sélectionner les zones de recalage à prévoir au cours du vol et de charger la cassette avion avec les terrains choisis. Cette installation dispose d'une bibliothèque de cartes numérisées avec la résolution maximale disponible (aujourd'hui 3" d'arc).

On peut concevoir une installation indépendante pour la sélection de terrain ou réalisant la préparation complète de la mission.

La configuration de l'installation proposée est représentée sur la figure n° 10. Les traitements nécessaires sont réalisés par un mini-calculateur, de type industriel par exemple. Des périphériques relativement simples complètent l'installation. Par ailleurs, l'installation est prévue pour être raccordée à un réseau numérique militaire.

L'installation doit permettre de réaliser en temps réel la sélection des terrains et leur chargement en cassettes avec un temps de réaction court. Enfin, l'exploitation de cette installation est étudiée pour être utilisée de façon simple par un opérateur non spécialisé, par exemple avec un dialogue en langage conversationnel.

9. CONCLUSION

L'exposé a montré les caractéristiques remarquables de la navigation optimale inertie-corrélation d'altitude :

- sur le plan opérationnel, la précision de la position et de la vitesse dans les trois dimensions, la discrétion, la résistance au brouillage, l'autonomie, la grande liberté de pilotage autorisée et le fonctionnement automatique, qui sont nécessaires pour la survie et pour l'efficacité,
- sur le plan matériel, la possibilité et l'intérêt d'une intégration fonctionnelle élevée dans l'inertie, et l'utilisation d'équipements simples indispensables par ailleurs à l'avion, qui conduisent à une solution fiable et économique en volume, masse, câblage et puissance,
- sur le plan de la préparation de la mission, une complexité modérée qui peut être encore réduite par l'utilisation d'une mémoire de masse sous forme d'une petite cassette en cours de développement.

Ces caractéristiques remarquables font de la navigation optimale inertie-corrélation d'altitude une solution attrayante pour les avions de pénétration et d'attaque au sol tout temps de la génération 85-90.

INTEGRATED FLIGHT AND FIRE CONTROL DEMONSTRATION ON AN F-15B AIRCRAFT: SYSTEM DEVELOPMENT AND GROUND TEST RESULTS

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Summary

The Integrated Flight and Fire Control (IFFC) program is a USAF Advanced Development Program to design, build, test, and evaluate an improved weapon delivery system. This improvement is accomplished by an automatic coupler and modified flight control system to steer out tracking errors calculated by a director fire control system using information from an ATLIS II electro-optical tracker. The F-15B aircraft is the test bed aircraft.

Simulation and analysis results indicate that the IFFC system has excellent air-to-air gunnery accuracy with decreased pilot workload. These results also indicate that attacker survivability in air-to-ground gunnery and bombing encounters can be increased by using maneuvering approaches without compromising air-to-ground weapon delivery accuracy.

Extensive ground testing of IFFC hardware and software has taken place and is described herein. The IFFC system is currently being evaluated in a 15 month flight test program which began in June 1981. Early testing has centered around verifying proper system operation. Details of the flight test program plan are presented in this paper.

1. Introduction

The USAF is sponsoring a program to design, build, test and evaluate an Integrated Flight and Fire Control (IFFC) system on an F-15B fighter aircraft. IFFC involves the blending of pilot control with automatic control to achieve improved weapon delivery accuracy and survivability in air-to-air gunnery, air-to-ground gunnery, and bombing. Automatic control is achieved by generating command signals to the flight control system based on steering errors determined by the fire control system. Figure 1 compares weapon line control for the baseline F-15B and the F-15B with the IFFC system.

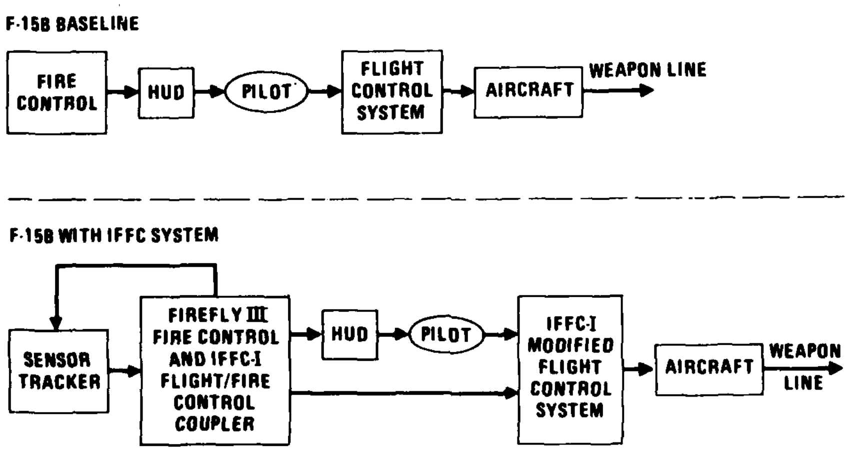

Figure 1. Weapon Line Control Comparison

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The overall IFFC program consists of two contracted efforts: IFFC I, being performed by McDonnell Aircraft (MCAIR) and sponsored by the Flight Dynamics Lab (AFWAL/FIGX); and FIREFLY III, being performed by General Electric (GE), Aircraft Equipment Division, and sponsored by the Avionics Lab (AFWAL/AART). The IFFC I program is: (1) providing the coupling between the flight and fire control systems, (2) modifying the standard F-15 control augmentation system (CAS) to accept coupler command inputs and to tailor flight control response for weapon delivery, and (3) integrating the overall system. The FIREFLY III program is: (1) providing an electro-optical sensor/tracker pod (ATLIS II), (2) processing tracking data to generate target state information, and (3) using target data to implement a director fire control system. The FIREFLY III program is discussed in detail in Reference (1).

The primary objective of the overall IFFC program is to demonstrate the feasibility of improving weapon delivery accuracy and attacker survivability through the use of IFFC techniques. This paper presents an overview of the IFFC program including system description, tracking/flight control features, safety provisions, system operational description, the onboard simulation feature, predicted weapon delivery performance, and test and evaluation plans.

2. IFFC System Description

A block diagram of the F-15B/IFFC system equipment and their interfaces with existing F-15 equipment is presented in Figure 2. Three existing line replaceable units (LRUs) are modified and six new ones are added.

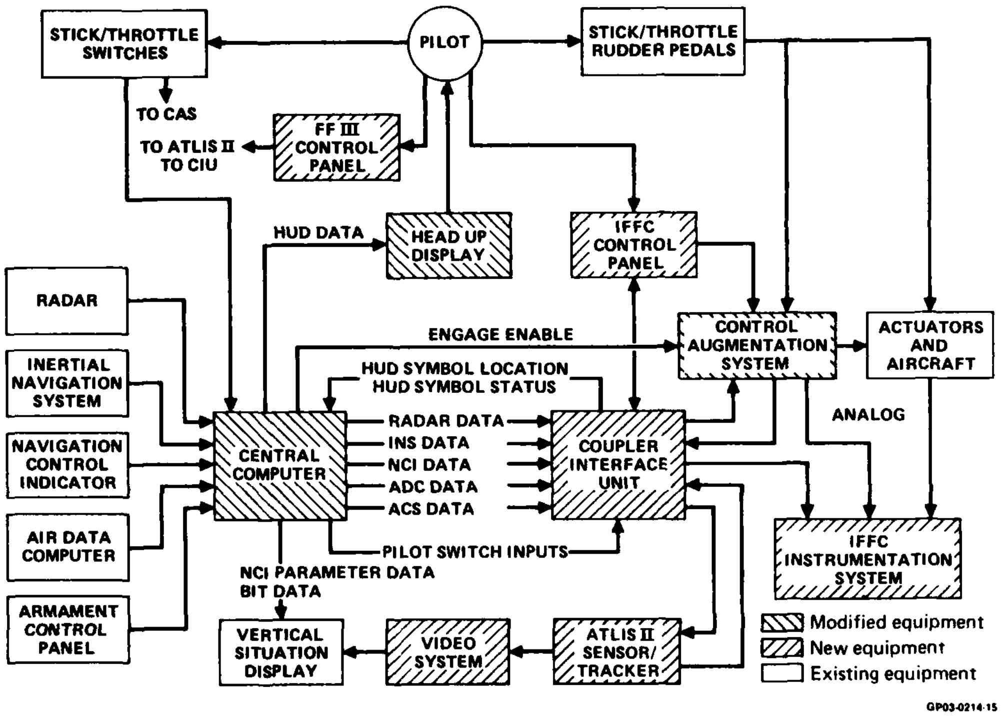


Figure 2. F-15B Equipment and Information Flow

Coupler Interface Unit (CIU) - The CIU is the heart of the IFFC system. It consists of a central processor, a 32K memory, a MIL-STD-1553A multiplex bus terminal, and analog-to-digital (A/D), digital-to-analog (D/A), and discrete input/output circuitry. The MIL-STD-1553A multiplex bus conveys digital information to and from the CIU, the F-15 central computer (CC), the sensor/tracker, and the IFFC airborne test instrumentation system. The D/A and A/D circuitry transmits data to and receives data from the CAS and aircraft control surfaces. The discrete input/output circuitry receives mode and control signals from aircraft systems and control panels. It transmits these status signals for display to the pilot.

The CIU contains both FIREFLY III and IFFC I software. The FIREFLY III software consists of modules for the Kalman filter target state estimation, the director fire control system, the bombing system, the ATLIS II tracking loop rate aiding, the coordinate transformations, the attacker aircraft state estimation, and the sensor preprocessors. The IFFC I software consists of modules for the gunnery and bombing coupler control laws, the Inflight Integrity Management (IFIM) system, the Built in Test (BIT), and the Onboard Simulation (OBS) modes.

F-15 Central Computer (CC) - The CC communicates with the existing F-15 subsystems via the existing multiplex bus. It has been modified to communicate with the new IFFC subsystems over the added MIL-STD-1553A multiplex bus. The modified CC acts as the 1553A bus controller. Also, the Controls and Displays module of the CC has been modified to accept Head Up Display (HUD) symbology information from the CIU. It combines this information with existing F-15 HUD symbology and transmits the combined data to the modified HUD Signal Data Processor.

F-15B Control Augmentation System (CAS) - The CAS has been modified to provide flight control responses tailored for the weapon delivery tasks and to accept command signals from the CIU. Details of the CAS modification are presented in Section 3.

ATLIS II Sensor/Tracker - ATLIS II is an electro-optical imaging tracker with laser ranging capability. It is carried in a pod approximately 8 feet long, 12 inches in diameter and weighing 320 pounds. This pod will be carried on the left forward sparrow missile station. The ATLIS II has an area correlation tracking mode as well as several contrast tracking modes. Its line-of-sight can be controlled by the pilot or slaved to the radar or INS for initial target acquisition. Following lockon, the ATLIS II tracker will be rate aided using attacker aircraft and target state information from the CIU.

Controls and Displays - The F-15B/IFFC forward crew station is shown in Figure 3. The primary IFFC controls are the IFFC Control Panel, the F-15B/IFFC Throttle, the F-15B/IFFC Stick Grip, the FIREFLY III Control Panel, and the Navigation Control Indicator (NCI). The primary IFFC displays are the Vertical Situation Display (VSD) and the Head Up Display (HUD).

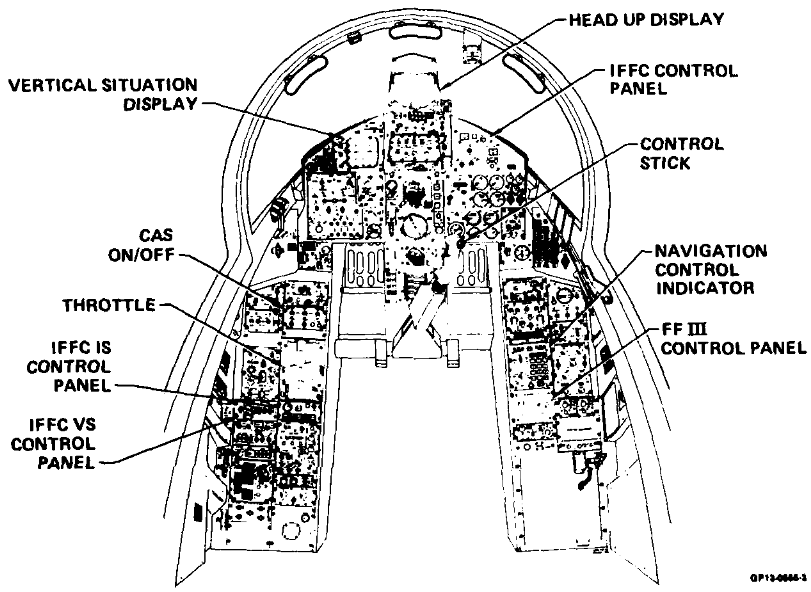


Figure 3. F-15B - IFFC Forward Crew Station

IFFC Control Panel - The IFFC control panel, Figure 4, serves as the central control and status display panel. It has hardwired connections with the CIU, the CAS and the CC. The master power switch controls power to the IFFC Control Panel and CIU. The IFFC ON/OFF switch assigns some of the throttle and stick switch functions, and changes HUD and VSD displays alternately to IFFC (ON) and normal F-15 (OFF). It does not engage the IFFC coupler and, hence, sends no commands to the flight control system.

The weapon mode lights identify which IFFC weapon delivery mode the pilot has selected. The velocity switch provides information to the modified CAS for use in converting acceleration limits to pitch rate limits. The simulation switch is used to enable the Onboard Simulation (OBS) mode which includes display of a simulated target on the HUD. The Initial Operating Condition (IOC) light illuminates when an IOC for the OBS target run has been selected; the RUN light illuminates when the simulated target sequence is in progress. The configuration select button is used to transfer configuration parameter data, which the pilot has entered into the CC using the NCI, from the CC to the CIU.

The BIT button initiates the IFFC Built-in-Test. It flashes while the test is in progress, remains on if a failure is detected, and extinguishes if no failure is detected. The IFFC warning light illuminates when a failure of any IFFC I element has been detected which would prevent coupling of the flight and fire control systems. The POD warning light illuminates when a failure has been detected in the FF III sensor/ tracker pod.

Throttle and Stick Control Functions - The F-15B/IFFC throttle functions are shown in Figure 5. Only the functions of the secondary control buttons are changed from the F-15B. They are controlled by the ON/OFF switch on the IFFC Control Panel. The only F-15B stick grip switch function that changes in the IFFC mode is the paddle switch. In the IFFC modes, it provides the pilot with an emergency IFFC disengage capability.

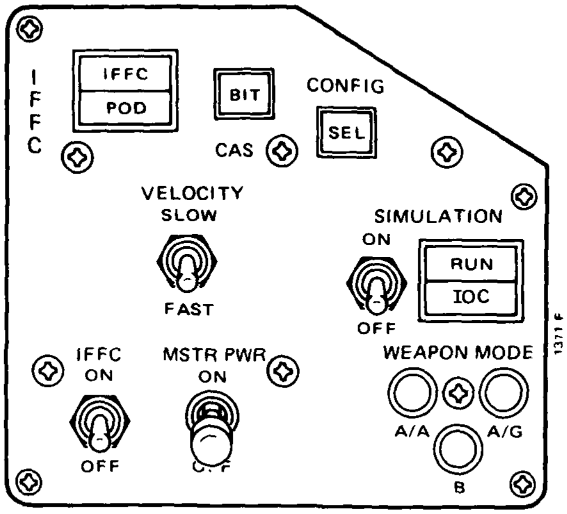
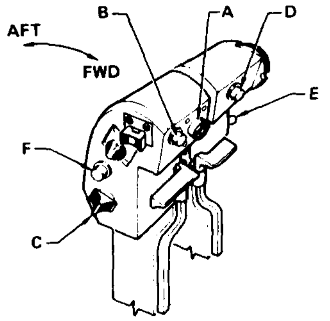


Figure 4. IFFC Control Panel GP13-0555-4



A - TARGET DESIGNATOR CONTROL (TDC)
- UP-DOWN, LEFT-RIGHT FOR RATE COMMAND TO SENSOR TRACKER LINE-OF SIGHT
- DEPRESS AND HOLD FOR:
  - < 1 SEC S/T LOCK-ON*
  - > 1 SEC BREAK LOCK, RETURN TO SLEW

B - COUPLE COMMAND
- ALTERNATE ACTION - COUPLE/UNCOUPLE

C - MODE SELECT
- AFT - AAG
- CENTER - AGG
- FORWARD - BMG

D - SIMULATION CONTROL
- 1st DEPRESS AND RELEASE START SIMULATION
- 2nd DEPRESS AND RELEASE STOP SIMULATION, RETURN TO INITIAL CONDITIONS AND HOLD

E - FIELD-OF-VIEW CONTROL
- AFT AND RELEASE, INCREASES FOV
- FORWARD AND RELEASE, DECREASES FOV

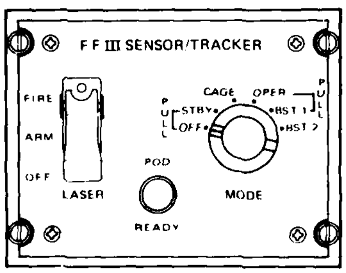
F - FIELD-OF-VIEW
- DEPRESS AND RELEASE, DECREASES FOV

A - Depress - Momentary Push Button Side Force-Rate Command to S/T
B - Momentary Push Button
C - 3 Position
D - Momentary
E - 3-Position, Return to Center
F - Momentary Push Button

*Also used to transfer track from point to area track

Figure 5. F-15B - IFFC Throttle Functions GP03-0214-4

FIREFLY III Control Panel - Figure 6 shows the FIREFLY III control panel and describes the ATLIS II operating modes as selected by the mode switch. The POD light is illuminated when the POD is ready, typically about 30 sec after selection of the STBY mode. A flashing light indicates a pod over-temperature condition. The laser switch is a three position guarded switch. In the OFF position, the laser can neither be armed nor fired. In the ARM position, the laser is armed and can be fired by: (1) BIT (into the retro-reflector), or (2) Automatic vidicon to laser boresight request (into the retro-reflector), or (3) automatic laser fire command on the mux bus, provided all safety interlocks have been satisfied (along the existing tracker line of sight).



OFF
- POD STOWED
- POWER OFF

STBY
- POD STOWED
- POWER ON

CAGED
- POWER ON
- POD UNSTOWED
- LOS CAGED

OPERATE
- LOS CONTROLLED BY CIU
- TRACK MODES: CRT, TAC, TPT, LST

BST 1
- TPT LOCK-ON STRAIGHT AHEAD (LOCKED OUT WHEN AIRBORNE)

BST 2
- TPT LOCK-ON 90° LEFT (LOCKED OUT WHEN AIRBORNE)

GP03-0214-5

Figure 6. Firefly III Control Panel

Navigation Control Indicator - The pilot can use the F-15 Navigation Control Indicator (NCI) to change IFFC parameters. Coded parameters are stored as "destinations" in the CC. The pilot depresses the Configuration Select (CONFIG SEL) button on the IFFC control panel (Figure 4) to transfer these parameters from the CC into the CIU. The NCI itself has not been modified to achieve this feature. Only the CC software which interprets inputs from the NCI has been modified.

IFFC Instrumentation System - The IFFC Instrumentation System is the principal data acquisition system aboard the aircraft. It will collect analog data from system elements such as the CAS, and digital data from the MIL-STD-1553A multiplex bus. It can record approximately 250 IFFC variables and telemeter the same information to a ground receiving station.

IFFC Video System - The IFFC Video system allows the sensor/tracker video to be recorded on a video tape recorder and displayed to the pilot on the Vertical Situation Display (VSD) as shown in Figure 7. The video system also includes two video cameras and a second video tape recorder to record the HUD and VSD in a split screen format.

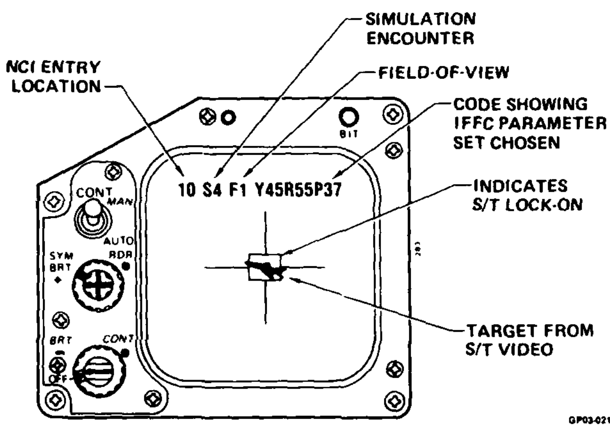


Figure 7. Vertical Situation Display - IFFC Mode

3. IFFC I Tracking/Flight Control Features

The IFFC I system generates automatic tracking control input commands to the IFFC modified CAS for the three weapon delivery modes; air-to-air gunnery, air-to-ground gunnery, and bombing. As shown in Reference 2, the F-15 CAS has been modified to accept these IFFC tracking control (outer loop) commands and to tailor flight control (inner loop) performance for each of the three weapon delivery modes.

Air-to-Air Gunnery (AAG) - The technique for automatic tracking control in the elevation axis for AAG employs proportional control of the elevation tracking error aided by the pitch rate of the line-of-sight to the target. The pitch coupler and flight control system thus matches the attacker's pitch rate to that of the line-of-sight while simultaneously nulling the elevation tracking error. The technique for automatic tracking control in the traverse axis employs the roll channel for nulling large tracking errors and the yaw channel for precise and rapid response tracking. The roll coupler and flight control system roll the aircraft such that the resulting tracking error is primarily in the pitch plane. This technique is similar to that used by fighter pilots for manual control. The yaw coupler and flight control system then operate on the remaining relatively small (less than 100 mr) traverse error by matching the attacker's yaw rate to the line-of-sight rate while simultaneously nulling the traverse tracking error using proportional control. An overview block diagram of the AAG tracking and flight control laws is given in Figure 8.

The bandwidths of the pitch, roll and yaw flight control (inner) loops of the CAS have been increased to accommodate the IFFC tracking control (outer) loops. This was done by increasing forward loop gains in all three channels and scheduling them with dynamic pressure. This scheduling provides a more nearly uniform response over the IFFC flight test envelope. In addition, the pitch CAS has been reconfigured from the blended normal acceleration and cancelled pitch rate feedback system used in the baseline F-15 to an uncancelled pitch rate feedback system which provides more precise pointing (attitude) control. Also, the IFFC yaw channel eliminates the lateral acceleration feedback, the roll-rate-times-aircraft-angle-of-attack feedback, and the cancelled yaw rate feedback. (These features are provided in the F-15 mode for coordinated flight. Removing them provides more precise pointing capability and aids the flight control system in matching the yaw rate of the line of sight.) In addition, a lead-lag compensation and a roll-rate-times-gun-elevation-angle feedback is added in the yaw channel to aid the aircraft in rolling around the gunline.

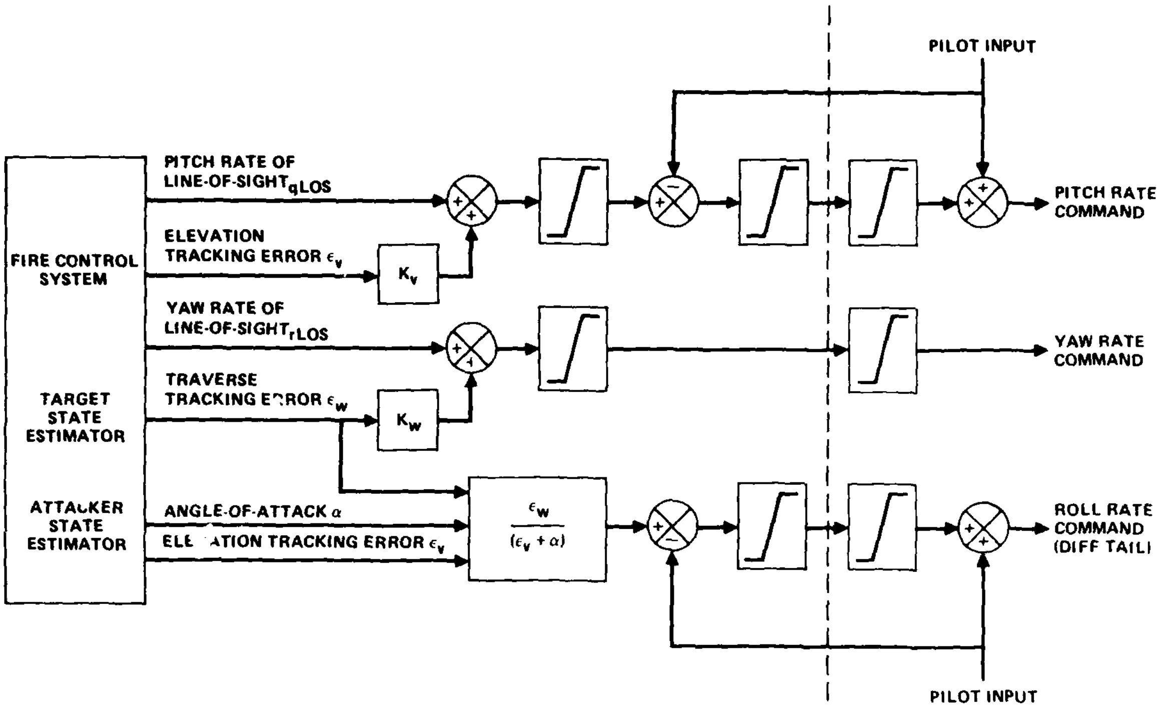


Figure 8. Air-to-Air Gunnery Coupler Control Laws

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Air-to-Ground Gunnery (AGG) - The AGG tracking control has two modes, a "MIN TIME" mode and a "MAX MANEUVER" mode. Each of these modes, in turn, consists of two distinct phases. In the MIN TIME mode, the first phase (convergence steering) is designed to take the aircraft from any attitude (usually wings level) into a banked accelerating turn such that the aircraft's velocity vector becomes aimed at a point which is a gravity drop above the target. At this point, the second phase (terminal steering) takes over using the same tracking loop control laws as AAG.

In the MAX MANEUVER mode, the convergence steering is designed to aim the velocity vector at an offset point from the point which is a gravity drop above the target. This introduces a large traverse error at the end of convergence steering which results in more sideslip and lateral acceleration during terminal steering. The MAX MANEUVER terminal steering uses the same automatic pitch and yaw channel control as AAG, but it allows the pilot to manually control the roll channel. Flight testing will determine whether the MIN TIME mode or the MAX MANEUVER mode results in greater survivability with the required weapon delivery accuracy.

The AGG flight control inner loops have the same modifications as the AAG loops except that: (1) the yaw integrator gain is increased by a factor of ten, and (2) an option exists to introduce "decoupling" feedbacks in the pitch and yaw channels via the CIU. The purpose of these feedbacks is to eliminate inertia-coupling and restoring-moment effects ($M_\alpha\alpha$, $N_\beta\beta$) from the pitch rate and yaw rate responses. This results in more precise control of pitch and yaw tracking loop rates which, in turn, improves pointing control.

Bombing (BMG) - The IFFC BMG mode uses a maneuvering nonwings-level delivery for improved attacker survivability. As in AGG, there is a convergence steering phase and a terminal steering phase. The convergence steering is designed to take the aircraft from any attitude into a banked accelerating turn from which the terminal steering is engaged. The convergence steering may be accomplished either manually or automatically.

The details of the bombing steering concept and its associated equations are explained in Reference (3). This concept results in a parameter triad consisting of normal acceleration, bank angle, and bomb release range associated with the steering commands. Selection of one of these parameters determines the other two, and the parameter selection provides the distinction between convergence and terminal steering as shown in Figure 9. Several combinations of manual and automatic control of the pitch and roll axes for both convergence and terminal steering will be available as options for IFFC flight test.

The IFFC bombing tracking control employs proportional plus integral control in its roll channel to null the delta bank angle commanded by the bombing fire control system. In addition, the desired normal acceleration, received from the bombing system, is commanded as an input to the pitch axis of the flight control system.

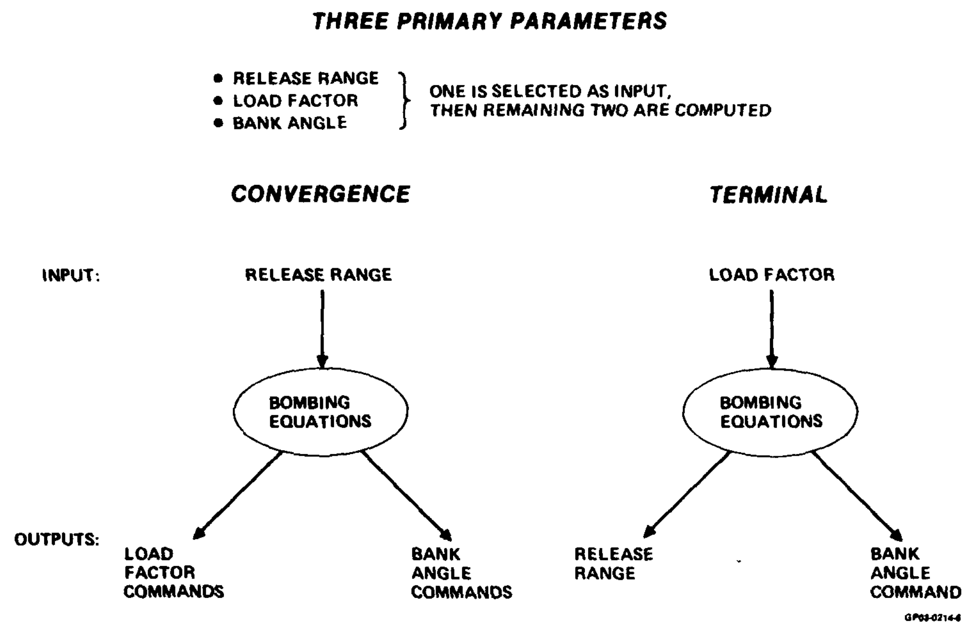

Figure 9. Bombing Steering Parameter Triad

The flight control inner loops in the BMG mode are unchanged from the F-15 baseline CAS in pitch and yaw, thereby retaining the normal acceleration command and coordinated flight characteristics of the baseline F-15 flight control system. In the IFFC CAS roll channel, the forward loop gain is increased just as it is for the gunnery modes in order to achieve a high bandwidth, quick responding roll channel.

4. IFFC System Safety

Safety is a primary consideration for the IFFC program since a simplex digital computer (CIU) is generating large authority command signals for the dual F-15B CAS.

Built-in-Test - The baseline F-15 BIT system is designed to detect 94% of failures in F-15 equipment and to fault isolate 94% of the failures detected. It checks circuitry which cannot be tested during normal system operation. BIT tests are pilot initiated and can be performed only on the ground since they interfere with normal IFFC system operation.

The BIT system has four major components:

- o Baseline F-15 BIT system (including avionics initiated BIT).
- o Central computer initiated BIT (including mux bus checks).
- o CIU initiated BIT.
- o Sensor/tracker initiated BIT.

The central computer initiated BIT test is expanded to include checks on the additional hardware/software which provides the 1553A bus controller capability. The CIU initiated BIT consists of tests which can be categorized into three groups: CIU Self Test, Inflight Integrity Management (IFIM) Valid Tests, and IFFC System Tests. The sensor/tracker initiated BIT determines the status of the ATLIS II with respect to temperature and both video, and laser tracking system operability.

Inflight Integrity Management - Inflight Integrity Management monitors IFFC system operation in flight and prevents IFFC system engagement, or causes disengagement when the IFFC system is already engaged, if a critical system failure is detected. The IFIM system has four major components:

- o CIU IFIM (self checks and IFFC system checks).
- o CAS IFIM (dual circuitry comparison, maximum aircraft rate and acceleration disconnects, and stick force disconnects).
- o CC IFIM (self checks and mux checks)
- o Sensor/tracker IFIM (overtemperature, power fail, computer fail, and laser fail).

IFFC Safety Design Features - The IFFC safety design features include: control command limiters, engage-disengage switching logic, and laser fire command interlocks. The flight control command limiters in both the CIU software and in the dual CAS hardware are attack mode dependent. They reduce coupler authority as pilot input increases. In addition, maximum pitch rate and normal acceleration limits are adjusted for asymmetric maneuvers.

It is important that a high authority system such as IFFC be implemented such that it can be engaged and can remain engaged only as long as specified engage criteria are satisfied. Figure 10 illustrates the IFFC coupler engage-disengage safety switching logic and shows the involvement of the CIU, the CAS, the CC, the throttle engage command, and the IFFC control panel.

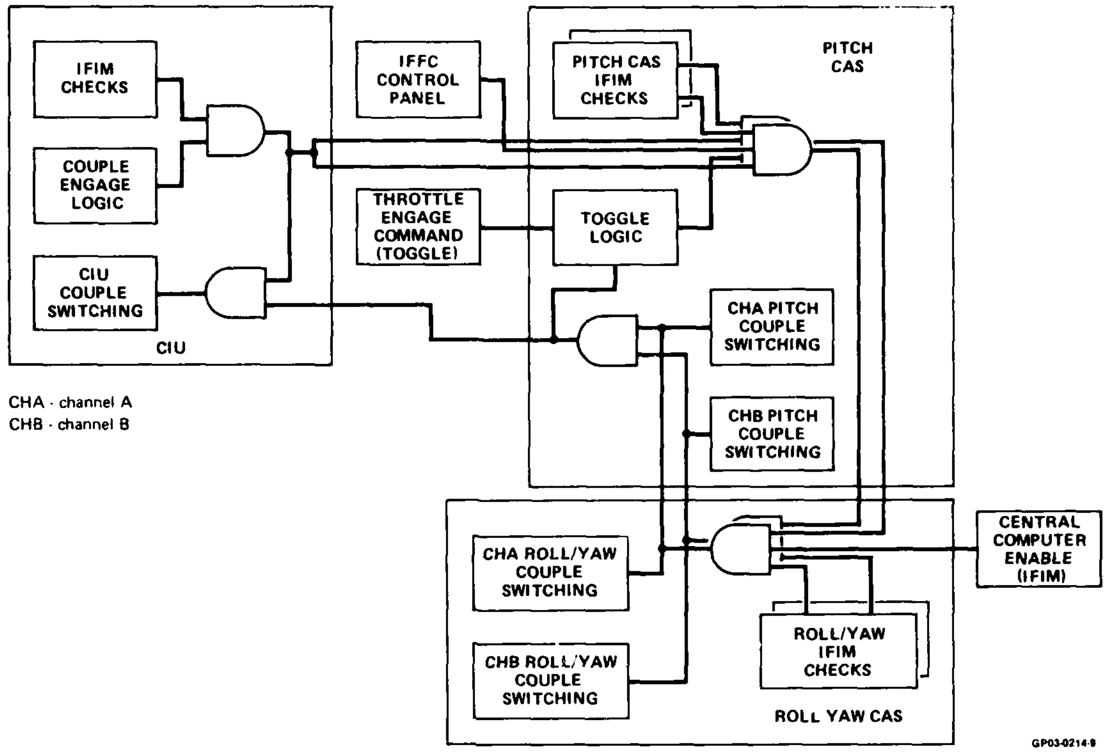

Figure 10. IFFC Coupler Engage/Disengage Safety Switching Logic

The CIU must not only satisfy IFIM checks for failures but it must also satisfy other IFFC system tests to engage the IFFC system or to maintain the IFFC system engaged; for example, the attacking aircraft must be outside of minimum air-to-air range and above minimum air-to-ground altitude. In addition, the system will be automatically disengaged at bomb release, and it will be prevented from engaging if the selected release range will result in a trajectory which violates the minimum air-to-ground altitude.

CAS logic interprets the meaning of the signal coming from the IFFC engage toggle switch on the throttle. For example, unless all CAS IFIM checks are passed, the IFFC system cannot be engaged despite the engage signal from the throttle. Furthermore, loss of any engage parameter will remove the engage command and automatically cycle the toggle switch to OFF. A pitch-command-rate-of-change monitor, which will disengage the IFFC system if its threshold is exceeded, is also implemented in the CAS. The central computer monitors mux bus performance and communication with the CIU. If a failure is detected, the CC enable signal to the CAS is not issued and the CAS will not accept command signals from the CIU.

The ATLIS II laser is inhibited from firing unless certain tests are passed. These tests are implemented by both the CIU and the ATLIS II pod. They test whether sensor/tracker IFIM checks are satisfied, IFFC modes and laser control are selected correctly, tracker lockon is maintained, and the landing-gear-up discrete has been received.

5. System Operational Description

The IFFC system is designed to provide a significant amount of flight test flexibility in order to evaluate alternate mechanizations, to assess the contribution of each system element, and to make changes as flight test results dictate. For example, alternate system parameters can be inserted in flight or between flights. In addition, CIU/CC software is structured to permit modification in areas prone to change, and CAS hardware is implemented to accommodate flight test derived modifications. The NCI can be used in flight to select alternate parameters for the coupler, flight control, sensor tracker, fire control, symbology, and onboard simulation.

Three typical weapon delivery scenarios and the associated HUD displays can be used to illustrate system operation.

Air-to-Air Gunnery - Figure 11 shows a typical air-to-air gunnery scenario and defines six sequential points of the encounter. Figure 12 shows the IFFC HUD symbology corresponding to points 3, 5, and 6. At point 3, the pilot's objective is to steer the aircraft such that the reticle is centered on the target.

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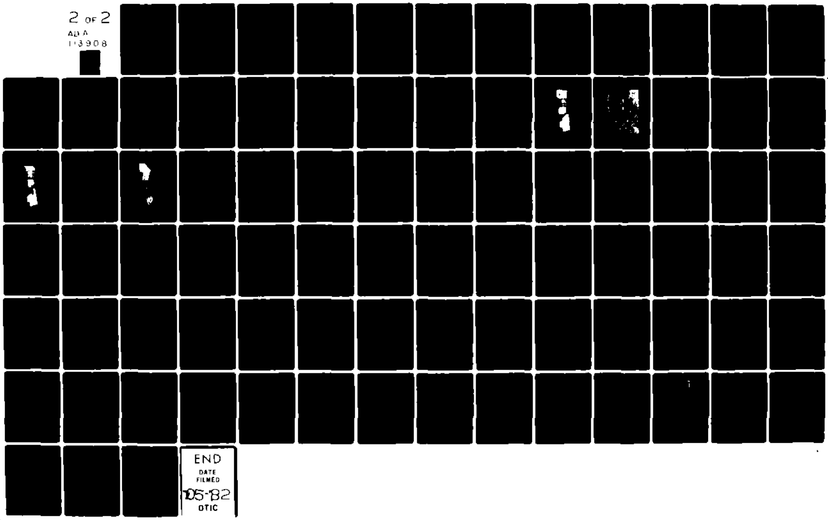

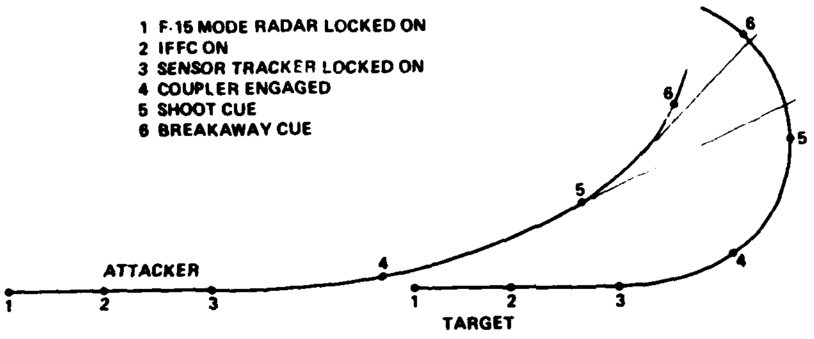

Figure 11. Air-to-Air Gunnery Scenario

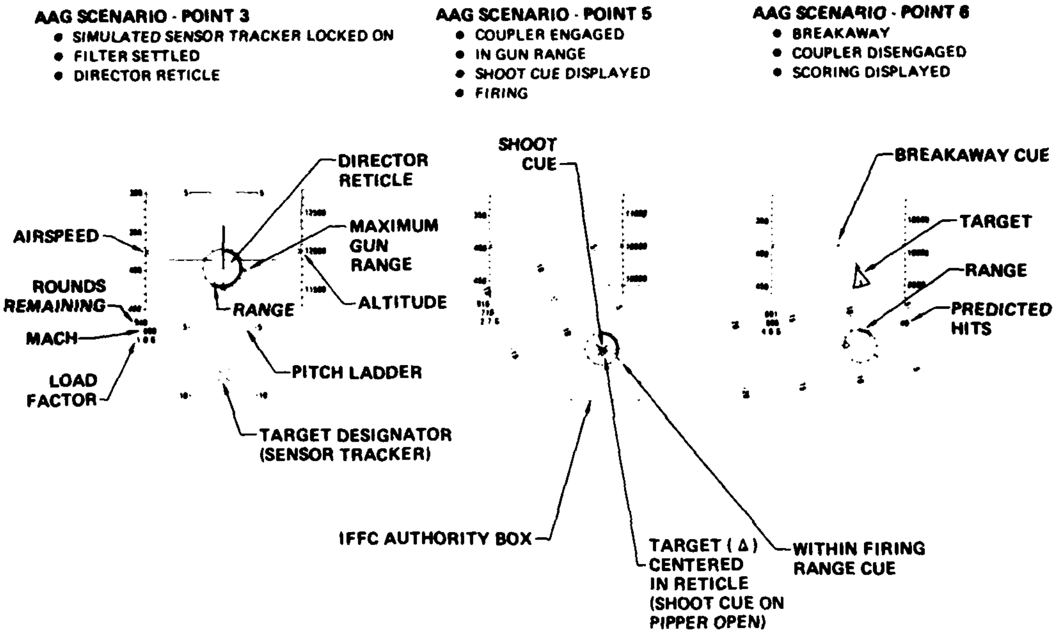

Figure 12. HUD Display During IFFC Air-to-Air Gunnery Scenario

At point 5 the target is within gunnery range and the IFFC system has been engaged and is tracking the target. The authority box indicates the maximum authority of the IFFC system. The pilot's objective is to keep the reticle in the box so that the coupler has the authority to automatically steer the reticle over the target. The shoot cue is on and the pilot is firing.

The HUD display at point 6 shows the breakaway cue which tells the pilot that he is too close or is closing too fast. The IFFC system has disengaged at this point and the pilot should break off the attack. The CIU predicted number of hits, determined by an onboard gunnery scoring system, is also being displayed.

Air-To-Ground Gunnery - Figure 13 illustrates a typical air-to-ground gunnery scenario and defines eight sequential points of the encounter. Figure 14 shows the IFFC HUD symbology corresponding to points 5, 7, and 8. The convergence steering line and the small box on this line are shown at point 5. The objective is to bank the aircraft so that the steering line becomes vertical in the HUD, and to command normal acceleration such that the small box is centered on the reticle. The time-to-go before transitioning to terminal steering is also shown.

The HUD at point 7 (after IFFC engage) shows the authority box. The objective again is to keep the reticle inside the authority box so that the IFFC coupler/flight control system can steer the reticle over the target. The target is at the center of the reticle, and the aircraft is within gun range so that the shoot cue is on and the pilot is firing.

The HUD at point 8 shows the breakaway cue, since the aircraft is still in a dive at a very low altitude.

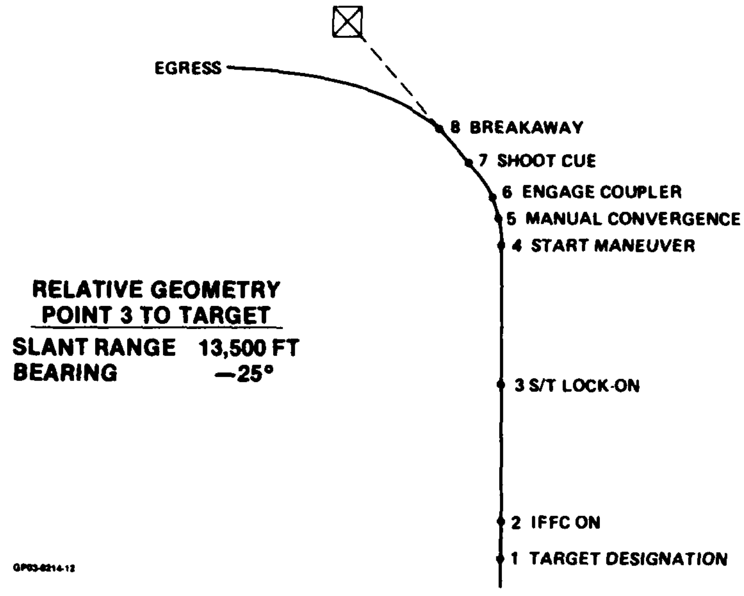

Figure 13. Air-to-Ground Gunnery Scenario

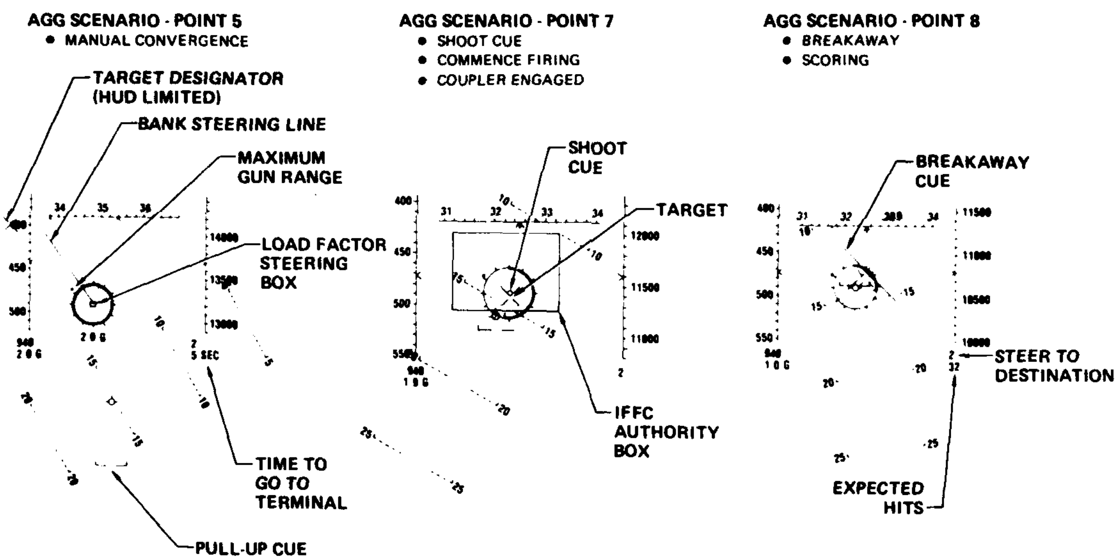

Figure 14. HUD Display During IFFC Air-to-Ground Gunnery Scenario

Bombing - Figure 15 illustrates a typical IFFC maneuvering bombing attack scenario and defines four sequential points of the attack. Figure 16 shows the HUD symbology corresponding to the four points. The HUD at point 1 shows the manual convergence steering display which functions the same as for the AGG mode. The fact that the target is out of the HUD field of view is indicated by flashing the target designator and locating it along the direction to the target. At point 2, the pilot's objective is to keep the reticle inside the authority box which enables the automatic IFFC steering to achieve the desired convergence steering commands.

The HUD at point 3 shows the automatic terminal steering at one second prior to bomb release. Note that the release cue is almost passing through the center of the reticle and that the actual range to target is nearly the desired range to target.

6. Onboard Simulation (OBS)

The OBS mode is designed for: (1) ground and preflight checkout consisting of iron bird tests and weapon delivery closed loop tests, (2) pilot familiarization both on the ground and in flight, and (3) inflight checkout and evaluation of the IFFC system. The onboard simulation capability (Figure 17) will be very useful for checking out the IFFC system on the ground rather than always using expensive flight testing.

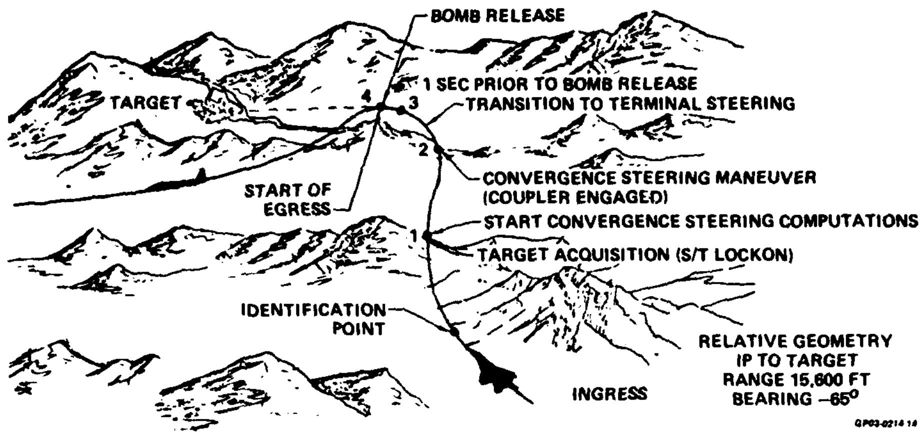

Figure 15. Bombing Attack Profile

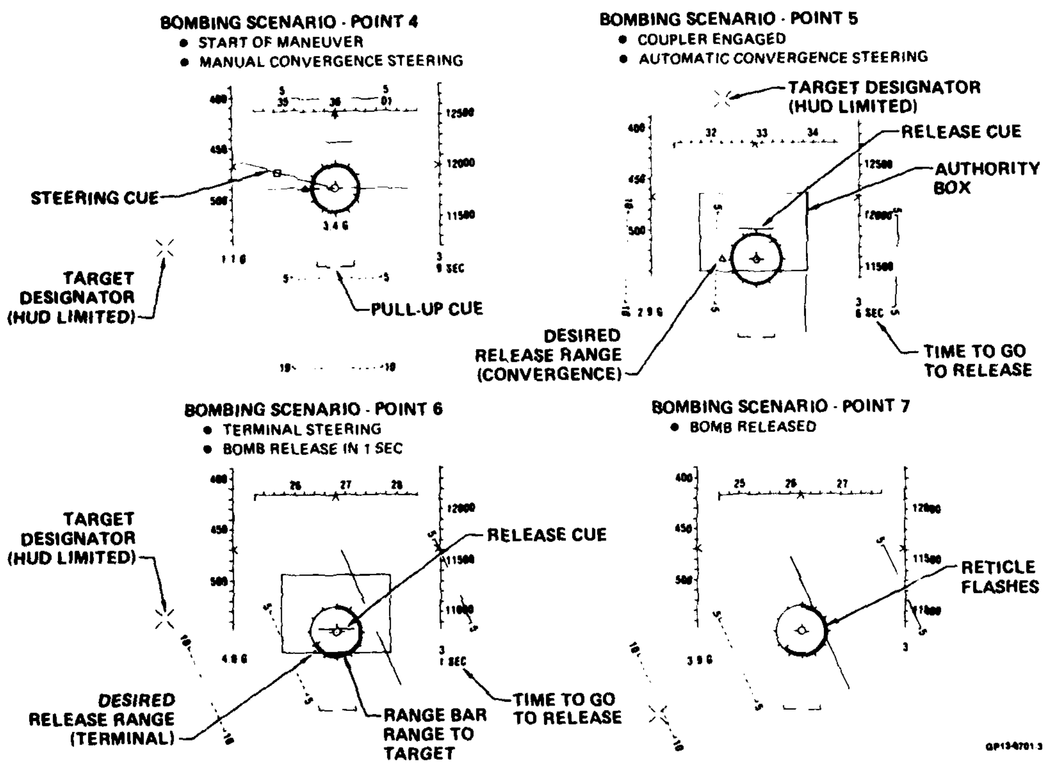

Figure 16. HUD Displays During IFFC Bombing Scenario

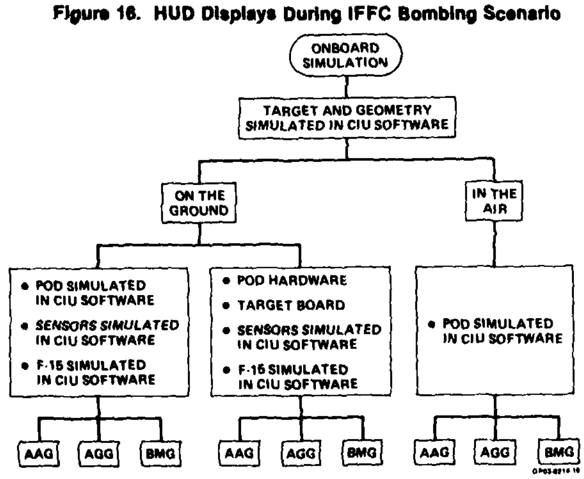

Figure 17. Onboard Simulation Configurations

The OBS was used extensively in ground testing on the aircraft and proved to be a very valuable troubleshooting tool. It was used both in closed loop checkout of the IFFC modified CAS as well as the total IFFC system checkout. OBS was also used extensively in the airworthiness flight testing in checking out the modified CAS as well as the entire IFFC system (minus the ATLIS II). A detailed description of the OBS mode and its implementation is available in Reference 4.

The OBS mode permitted much of the early AAG flight testing to be done without the expense of having airborne targets. The OBS mode is also available for AGG and BMG. Using the OBS, AGG and BMG can be checked out and verified at a safe altitude by using simulated "ground targets" set at an altitude well above ground level. And, if the ATLIS II pod should be unavailable for some period of flight testing, testing of the remainder of the IFFC system can proceed using OBS until the pod is again available.

7. Predicted Weapon Delivery Performance

IFFC simulations were conducted on the MCAIR Manned Air Combat Simulator (MACS) to develop the IFFC system, to trade off mechanization alternatives, and to obtain USAF test pilot opinions.

AAG Simulations - Two configurations were evaluated for the AAG mode; they are termed "Snapshoot" and "Baseline". The Snapshoot configuration uses a larger tracking loop gain (high bandwidth) but can only be engaged for relatively small levels of tracking error (less than 50 mr). The Baseline configuration provides somewhat better tracking time and hits on target; however, since the Snapshoot configuration also performed well, both were retained for flight testing.

AGG Simulations - Three AGG configurations of manual and automatic control were evaluated:

(1) Manual convergence, 2 axis (pitch, yaw) terminal.
(2) Automatic convergence, 2 axis terminal.
(3) Automatic convergence, 3 axis terminal.

Figure 18 shows the relative results for these configurations in terms of expected hits and total tracking time. Configuration (3) outperformed the others; however, all three configurations performed well enough to be carried into IFFC flight testing. All configurations demonstrated excellent attacker survivability against ground targets defended by anti-aircraft artillery, through the use of maneuvering deliveries.

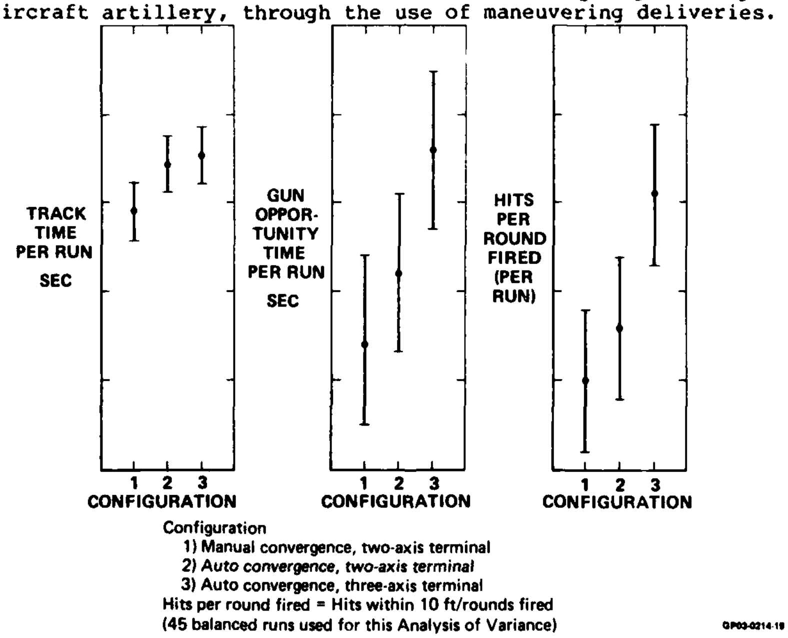

Figure 18. Comparison of Three IFFC Air-to-Ground Gunnery Configurations

Bombing Simulations - Three BMG mode configurations of manual and automatic control were evaluated:

(1) Manual convergence, roll terminal.
(2) Manual convergence, pitch-roll terminal.
(3) Automatic convergence, pitch-roll terminal.

Figure 19 shows the results from the simulation which indicate that configuration (1) outperformed configurations (2) and (3). However, there were some HUD display problems during this BMG mode simulation. For this reason, and since they performed well, all three configurations were retained for IFFC flight test. All configurations demonstrated excellent attacker survivability against ground targets defended by anti-aircraft artillery through the use of maneuvering deliveries and increased release ranges made possible by the accurate IFFC bombing system.

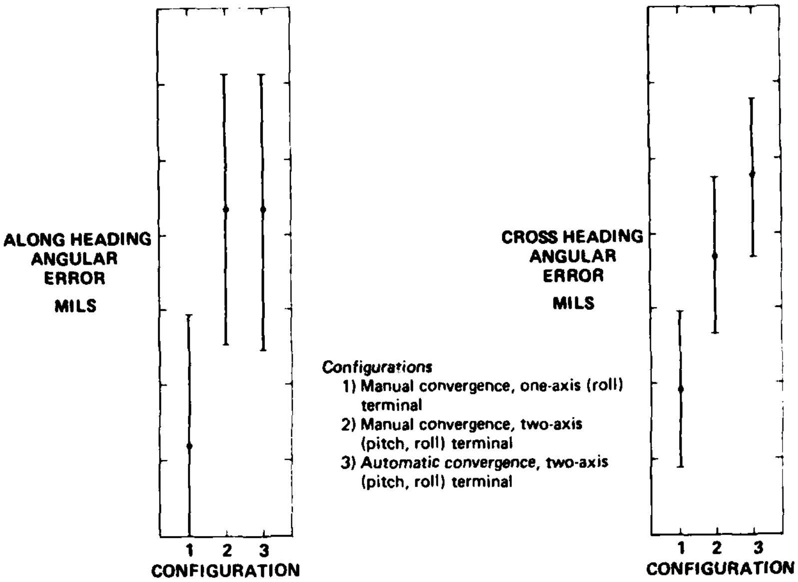

Figure 19. Bomb Delivery Accuracy Comparison of Three Configurations

8. Test and Evaluation

The evaluation of the IFFC system includes the following testing phases:

- o MACS/All-Software Model.
- o Software Test Facility/CAS Lab.
- o MACS/Hardware-in-the-Loop Simulation (HILS).
- o Ground Tests.
- o Airworthiness Tests.
- o Flight Test.

MACS/All-Software Model - In the first phase of IFFC testing, an all-software model of the preliminary IFFC system was evaluated in the MACS. The purpose of this test, which was conducted during the predevelopment phase of the IFFC program, was to select primary and alternate configurations for further development and flight testing. As a result of these tests, best control laws were selected, best controls and displays configurations were chosen, and the adequacy of the safety features was assessed. USAF test pilots participating in this test provided many valuable suggestions that were incorporated into the IFFC system design.

An updated all-software model was also evaluated in the MACS. The purpose of this test, which was conducted during the development phase of the IFFC program, was to: (1) incorporate an updated fire control system and assess its effects on IFFC system performance, (2) provide a first level check on the fire control portion of the CIU Operational Flight Program (OFP) by using the same Fortran as is being used to develop the assembly language OFP, (3) provide a "truth" model for use in the Hardware-in-theLoop Simulation (HILS), (4) study potential problem areas such as sensor/ tracker noise effects and computational delay effects, and (5) evaluate IFFC HUD symbology.

Software Test Facility/CAS Lab - After passing acceptance tests at the suppliers' facilities, the IFFC hardware and software were brought to MCAIR for further testing. This testing used the F-15 Software Test Facility (STF) and the F-15 CAS laboratory.

The STF uses a host computer to simulate aircraft dynamics and F-15 avionics subsystems. It was used to check out: (1) The 1553A bus addition to the F-15 CC, (2) CC interfaces to the HUD, NCI, VSD and CIU, (3) CIU outputs to the CAS, and (4) HUD and VSD functioning. The F-15 CAS laboratory was used to test and support the IFFC modified CAS during the period of Hardware-in-the-Loop simulation testing. A special test bench had been constructed for this purpose.

The STF operated in parallel with the Hardware-in-the-Loop Simulation (HILS) and the ground testing on the aircraft. Problems with the CIU OFP, which were identified in any of the three facilities, were eventually corrected with software patches tested and verified in the STF. During the course of this testing, there were approximately 230 such changes. The STF was also used to generate new OFP versions which included these changes configured as patches in the previous version.

The STF was also used to investigate CIU hardware problems as well as Central Computer (CC) hardware and software problems. This facility will be maintained throughout the flight test program to support the CIU and CC hardware and software.

MACS/Hardware-in-the-Loop Simulation - The next phase of system checkout was the IFFC HILS in which all IFFC airborne hardware and software, except for the sensor/tracker, were assembled and tested in dynamic encounters using the MACS facility and the F-15 Iron Bird. HILS was set up as indicated in Figure 20. USAF and MCAIR test pilots participated in the simulation which: (1) integrated most of the IFFC hardware/software in a dynamic operational environment prior to its incorporation on the F-15B test aircraft, (2) verified F-15B (non-IFFC) system hardware functions, (3) uncovered and corrected IFFC system anomalies, (4) verified on the onboard simulation operation, (5) verified BIT/IFIM module operation, and (6) verified the digital interface between the instrumentation system and the IFFC system.

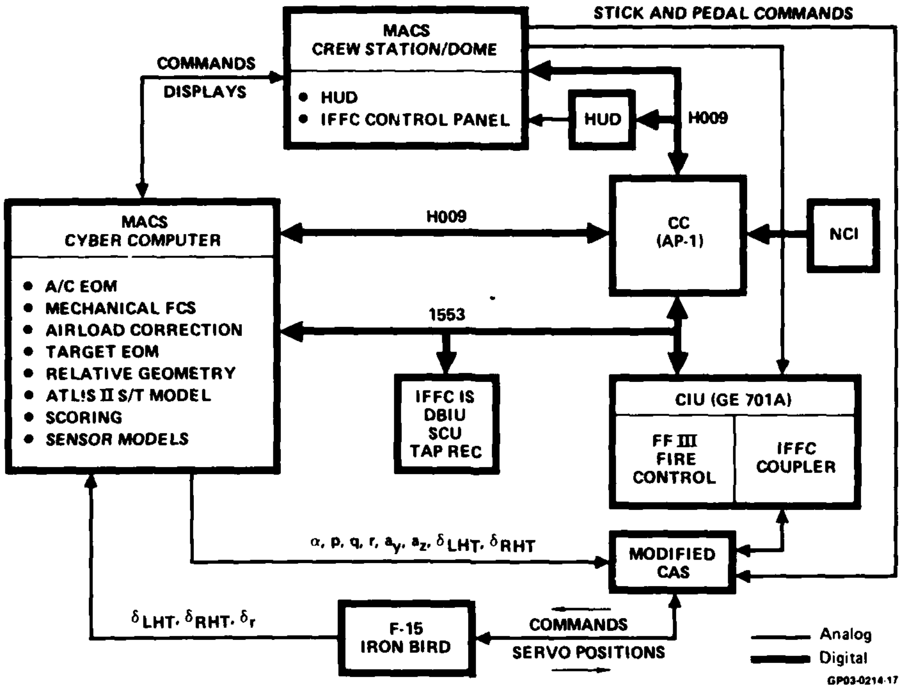

Figure 20. IFFC Hardware-in-the-Loop Simulation Configuration

The HILS operated in parallel with the STF and ground testing on the aircraft. During the course of HILS testing, there were approximately 110 problems identified. These problems can be categorized as CIU OFP errors, CIU and CC hardware problems (e.g., parity errors) and system interface problems. The HILS was used principally for identifying and investigating IFFC system problems but was also used several times for pilot training and familiarization. All three IFFC weapon delivery modes were checked out on the HILS prior to flight testing.

Ground Tests - IFFC ground testing took place following installation of all IFFC I/FF III equipment on the F-15B test aircraft. This testing ran in parallel with the HILS and STF testing previously described. The objectives of ground testing were to: (1) integrate flight control and fire control hardware into the F-15B test aircraft, (2) verify F-15B (non-IFFC) system hardware functions, (3) verify IFFC functional operation in the AAG, AGG and BMG modes, (4) verify onboard simulation operation, (5) verify BIT and IFIM operation, (6) boresight and calibrate system components, (7) verify instrumentation system integration with IFFC system, and (8) assure Electromagnetic Interference/Electromagnetic Compatibility (EMI/EMC) compliance of the IFFC I/FF III system and the other F-15 avionics subsystems.

The IFFCC modified Control Augmentation System was the first piece of IFFC hardware tested on the aircraft. Frequency response, time response, vibration tests and safety feature verifications were run. One structural mode was excited when the roll CAS gain was raised to more than three times its production F-15 level. This oscillation was eliminated with a notch filter in the roll rate feedback.

The total IFFC system was then tested. This included the IFFC CAS, Coupler Interface Unit, Central Computer, Head Up Display, Signal Data Processor, IFFC and FF III control panels, and the ATLIS II sensor/tracker. The OBS-on ground mode was used for this testing. During the course of this testing, 61 problems were identified. These consisted of CIU software problems, CIU hardware problems (e.g., spikes on A/D converters, electrical oscillations in the Load Factor Error Sensor (LOFES) biasing signal, CIU parity errors), as well as system-aircraft interface problems (e.g., scaling, signal polarity).

The AAG mode was thoroughly checked out whereas calendar time limitations permitted only partial checkout of the AGG and BMG modes. Nevertheless, all safety features for the three modes were checked out. Checkout was accomplished first using a simulated sensor/tracker and then using the real ATLIS II sensor/tracker. Vibration testing was done using the real ATLIS II sensor/tracker and its associated target board. No sustained oscillations were observed.

The FF III system tests were run to verify proper ATLIS II functioning and to check the accuracy of the masking algorithm. Several problems were encountered and corrected during this period. The ATLIS II sensor/tracker was also boresighted as part of these tests.

The EMI/EMC tests uncovered three problems. The solutions involved one modification to the CC reset button in the cockpit and two procedural changes for the IFFC flight manual.

Airworthiness Tests - The airworthiness tests ensured that the aircraft with IFFC system installed was safe and ready to enter development flight tests. The aircraft was released for airworthiness flight tests upon satisfactory completion of the ground and hardware-in-the-loop simulation tests. Following satisfactory completion of flight readiness inspections, a functional check flight was flown by the MCAIR project pilot.

The first objective of airworthiness test flights was to confirm that the aircraft functions correctly in the F-15B mode. The second objective was to verify that the IFFC system functions safely and as intended. It was not an objective of airworthiness tests to assess the quality of IFFC system performance.

Aircraft handling in the IFFC mode was evaluated in the middle of the IFFC flight test envelope. Outer loop tracking and inner loop flight control operations were checked. The IFFC I/FF III system was engaged and disengaged under dynamic maneuvering conditions to check for transients. Twelve airworthiness test flights were made during the March to May 81 time period. The system features tested included:

- Safety Disconnects
- Modified CAS Operation
- Onboard Simulation - AAG Mode
- Onboard Simulation - BMG Mode
- ATLIS II Operation - AA Mode
- ATLIS II Operation - AG Mode

These tests were performed "in the middle" of the IFFC flight test envelope; specifically the flight conditions of .7 Mach and .9 Mach at 15,000 feet altitudes. CAS and coupler gains were adjusted and some ATLIS II modifications were made to improve video quality based on results from these flights. The IFFC system performed well and the ATLIS II target tracking capability was demonstrated.

Flight Test - The first nine months IFFC flight test program were planned by MCAIR and are being conducted by Air Force Flight Test Center (AFFTC). The following six months will be conducted by the USAF. The associated contractor's FF III instrumentation and test requirements were incorporated in the flight test plan. Testing is following the usual confidence buildup approach by progressively investigating the IFFC system from zero to full authority automatic control for the three IFFC modes (AAG, AGG and BMG).

It is important in a flight test effort to be able to adjust and refine the flight test plan on a flight-by-flight basis. Quick-look data will be obtained from the onboard telemetry and from video tape replay. It will permit rapid dissemination of quick-look reports.

The IFFC I/FF III flight test envelope is shown in Figure 21. The load factor, flight path angle, and bank angle limitations are noted. The 26000 ft altitude and subsonic Mach number restrictions were imposed to reduce the cost and schedule of the demonstration program. They do not represent technology limitations.

The flight test program began in June 81 at The Air Force Flight Center, Edwards AFB. It will consist of 150 test flights, 50 each for AAG, AGG and BMG. Approximately 100 flights will be allocated to IFFC system development and 50 flights to formal evaluation.

The first flight test phase addressed preliminary functional testing. This phase involved:

- Checkout of the Onboard Instrumentation and Video Systems
- Verification of Correct Gun Operation and Alignment
- A HUD/Windshield parallax assessment
- Determination of Bomb Ejection Velocity and Delays
- Validation of HUD/Gun Alignment
- Investigation for Possible Radar Range and Range Rate Bias
- Air Force IFFC Pilot Training and Familiarization.

	AAG	AGG	BMG
LOAD FACTOR	$-1\,g$ TO $7\,g$	$-1\,g$ TO $5\,g$	$0.5\,g$ TO $5\,g$
FLIGHT PATH ANGLE	ALL	-5° TO -60° (DURING GUNFIRE)	$+60^{\circ}$ TO -60°
BANK ANGLE	ALL	ALL	ALL

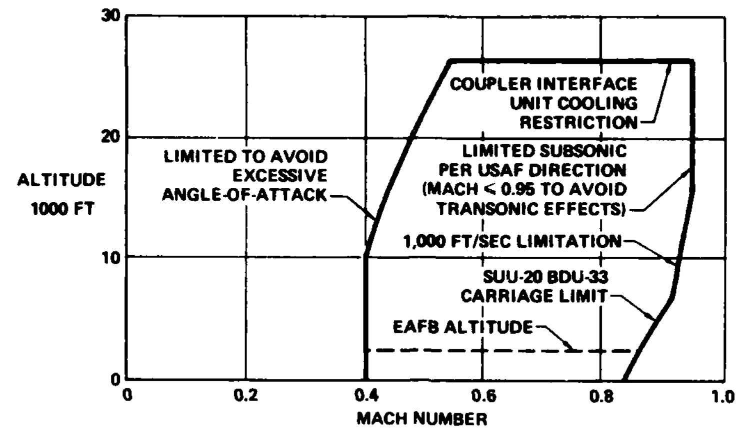

Figure 21. IFFC Flight Test Envelope

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The second phase will involve IFFC system integration. The purpose of this phase, which will consist of approximately 25 flights, is to verify the proper functioning of the CAS, CIU software/hardware, and the ATLIS II sensor/tracker in all three IFFC weapon delivery modes. At the end of this phase, there should be no major implementation errors and the IFFC system should be ready for the development testing phase in which performance will be optimized. The specific objectives of Phase 2 are:

- OBS Checks
- Tracker Target Acquisition and Noise Tests
- Encounter Profile Development
- Tracker Masking/INS Mode Assessment
- Target State Estimator Validation
- Director Fire Control Assessment
- Onboard Scoring Correlation with Ground Based Progams
- Coupler Checkout and Authority Buildup
- Bomb Algorithm Assessment
- ATLIS II Operation
- Integrated System Tests (3 Modes)

The third phase of flight test will be the Development phase where the main objective is to modify IFFC system parameters to improve and optimize system performance.

The fourth phase will be the Air Force Evaluation in which the objective is to generate statistically significant quantities of data to determine the improvement in weapon delivery performance of the optimized IFFC system relative to performance improvement predictions and program goals.

The fifth and last phase will be performed at Air Force Tactical Fighter Weapon Center at Nellis AFB. A realistic anti-aircraft artillery environment will be available to assist in evaluating improvements in survivability during air-to-ground weapon delivery.

Conclusions

An Integrated Flight and Fire Control (IFFC) System has been designed and built, laboratory tested, ground tested on the test aircraft and is currently being demonstrated in flight on an F-15B aircraft. Studies and simulations indicate that the IFFC system can provide excellent weapon delivery accuracy with reduced pilot workload for air-to-air gunnery, air-to-ground gunnery, and bombing. Very good attacker survivability for the interdiction of anti-aircraft artillery defended ground targets through the use of maneuvering deliveries is also indicated. Flight testing of the IFFC system began in June 1981 to verify the improved accuracy and attacker survivability predicted in the studies and simulations. A 15 month flight test program is planned.

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THE INTEGRATION OF MULTIPLE AVIONIC SENSORS AND TECHNOLOGIES FOR FUTURE MILITARY HELICOPTERS

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SUMMARY

The expanding role of the helicopter in the battlefield environment has burdened the pilot with missions of greater complexity and risk with a concomitant increase in pilot workload. Navigation of the helicopter is an essential supportive element to the prime mission and has until recent years been a significant contribution to the workload. Technological advances in navigational electronics such as Doppler navigation radar, computers, integrated avionic control and display systems, etc., now can provide automated navigation with vital benefits in cost, size, weight and power which permit incorporation of these advances into the helicopter. Cost reductions are particularly important since helicopters are used in large quantities in modern military forces. Multi-sensor navigation systems already available and in use in helicopters are discussed followed by a review of the system trade-offs and considerations leading to new systems that use more advanced digital electronic techniques to achieve the goals of reduced pilot workload, improved performance at minimum size, weight, and cost. The beneficial impact of ongoing technological advances in improving the operating capabilities of future avionics systems is indicated.

IMPACT OF HELICOPTER MISSION REQUIREMENTS ON NAVIGATION

The military helicopter has been given a continuously expanding role both on land and at sea. Helicopter missions include rescue, reconnaissance, troop delivery, attack and weapon delivery. These missions are often performed under active battlefield stress and may require the pilot's full attention for the accomplishment of the mission objective. To permit the pilot to cope with the workload imposed by more demanding mission tasks, it has become necessary to relieve him through automation of some of those supportive functions which have been historically performed manually. One of those supportive functions which technology has permitted to be automated is precise navigation.

Under favorable conditions of visibility and unrestricted altitude the pilot workload associated with manual map and compass navigation may be tolerable. However the battlefield environment no longer permits the helicopter to fly at the 300 to 500 foot altitudes which allow the pilot or navigator to navigate using simple visual terrain and map correlation. Experience in recent conflicts has proven the vulnerability of the helicopter when it is exposed at these altitudes. Ultra-low altitude flight (under 50 feet) increases the survivability of these craft from enemy ground fire, but makes VFR flying by use of map and compass difficult. Any tactical incident will almost surely cause the pilot or navigator (if there is one) to lose track of his position and become disoriented even in clear weather. This, coupled with the expectation that battlefields of the future will be active at night and in inclement weather, makes map and compass dead reckoning navigation highly impractical.

Figure 1(a) depicts the functions which in the past had to be performed manually by a pilot in order to navigate. He served as a data gatherer and computer to combine the outputs of several sensors to derive the navigation output. Air speed, attitude, heading, drift angle, and altitude are the major inputs which he must read and combine to plot his position on a map. He eliminated accumulated errors in his dead reckoning computation by periodic updates provided by visual identification of local terrain features, a procedure which is highly ineffective at night, in poor visibility or at low altitudes. Manual dead reckoning is an acceptable technique when missions and helicopters are simple, but with the increasing workload, stress, and confusion during today's tactical situations, it no longer is adequate.

These problems demanded the development of integrated avionics to provide automatic navigation to reduce pilot workload and supply him with position, steering and guidance information and true ground velocity for many purposes besides navigation.

Considerations of survivability and the possibility of providing these outputs using on-board sensors favored a self-contained navigation system as opposed to ground-based aids which could be jammed, spoofed or subject to attack.

DEVELOPMENT OF CURRENT RESPONSE TO BASIC NAVIGATION NEEDS

An Integrated, Automatic Helicopter Navigation System

The system considerations discussed above indicate the need for an automatic navigation system that would, in effect, replace the pilot's manual navigation task with a computer. This allows one to take advantage of the computer's greater capability by adding additional navigation sensors that are more accurate and that further reduce his workload. For example, it now becomes possible to replace the airspeed sensor, which requires the pilot to estimate wind magnitude and direction, with a direct, automatic groundspeed measuring sensor such as a Doppler radar. Figure 1(b) shows the result of applying these new capabilities to the navigational procedures outlined in Figure 1(a).

The navigation and guidance data generated by the computer shown in Figure 1(b) must be displayed in a manner that minimizes the amount of interpretation required by the pilot to translate them into helicopter control actions. The most desirable approach is to enable the pilot to enter the coordinates of destinations or targets prior to or during flight, and to cause the computer to generate a left-right steering signal which, when zeroed, causes the helicopter to fly toward the selected destination. The approach greatly reduces the time spent by the pilot with a hand-held map, and the greater accuracy of such a navigation system reduces the required frequency of position updates and hence further reduces the pilot's map reading tasks.

The U.S. Army, recognizing the need for an automatic navigation system such as that shown in Figure 1(b), chose a Doppler Radar Velocity Sensor (DRVS) for the direct, automatic groundspeed measuring sensor. The selection of this type of a velocity sensor is appropriate for several reasons:

- It is self-contained.
- It can provide accurate data throughout the speed/altitude profile of a helicopter, namely, from negative (backward) flight through hover to positive (forward) flight.
- Accuracy is independent of time and hence, of the length of the mission.
- It provides instantaneous reaction time.
- It is all solid state and has no moving parts.
- A Doppler radar has a lower Life Cycle Cost (LCC) in comparison to other types of velocity sensors, and thus is affordable for large quantities of helicopters.

Kearfott had already begun development of a Doppler system having these characteristics when the Army translated these requirements into a specification. The development was completed under Army sponsorship to produce the AN/ASN-128 Lightweight Doppler Navigation System shown in Figures 2 and 3.

The Receiver-Transmitter-Antenna (RTA) and Signal Data Converter (SDC) form the DRVS. The RTA contains the antenna with its integral radome, R.F. components for transmitting and receiving electromagnetic signals, and electronics to condition the resultant signals. The SDC receives the conditioned signals from the RTA, measures the Doppler shifts, and converts these into the digital form required by the computer. The Computer Display Unit (CDU) contains a custom LSI digital computer and memory that perform all navigation and guidance computations, and displays the results on its front panel. Entry of destination coordinates and other data are made via a keyboard, while control switches are provided for mode and display control selection. The Steering-Hover Indicator Unit (SHIU) displays guidance data to the pilot in "analog" form, that is, as pointers and needles although it also contains a numeric readout of distance-to-go. The SHIU is an optional addition to the system.

The AN/ASN-128 provides accurate navigation over the flight parameters listed below:

Along-heading velocity	-	-50 to +350 knots (-93 to +650 km/h)	
Cross-heading velocity	-	$\pm$100 knots ($\pm$185 km/h)	
Vertical velocity	-	$\pm$5000 ft/min ($\pm$1500 m/min)	
Altitude	-	0 to 10,000 ft (0 to 3000 m) above ground level	
		Land	Water
Attitude - Pitch	-	$\pm 30^\circ$	$\pm 20^\circ$
Roll	-	$\pm 45^\circ$	$\pm 30^\circ$

Accuracy of the AN/ASN-128 can be stated for the DRVS, alone, and for the computed position after Doppler velocity data are combined with heading from the on-board heading reference.

Doppler Radar Velocity Sensor Accuracy (one-sigma)

Along heading	0.25% + 0.1 knot
Cross-heading	0.25% + 0.1 knot
Vertical velocity	0.1% + 0.05 knot

Position accuracy is 1.3% (CEP) of distance traveled when heading reference accuracy is one degree (one-sigma)

Physical characteristics of the AN/ASN-128 and SHIU are listed below.

AN/ASN-128	WEIGHT		VOLUME		POWER	MTBF
	kg	lb	cm^3	in^3	(W)	(h)
DRVS: RTA	4.76	10.5	7,752	473	12.0	12,237
SDC	5.67	12.5	9,296	564	56.6	7,744
DRVS Total:	10.43	23.0	17,048	1,037	68.6	4,742
CDU	3.18	7.0	3,676	224	30.0	3,838
Total:	13.61	30.0	20,724	1,261	98.6	2,121
SHIU	0.91	2.0	1,067	65	15.0	10,000

The AN/ASN-128 provides accurate present position, steering signals and, distance and time-to-go to 10 destinations. It uses a printed grid antenna that is self-compensating for the over-water shift. Navigation data are displayed in both UTM coordinates and latitude/longitude.

Other operational advantages include:

- single transmit-receive antenna makes maximum use of the available aperture, which minimizes random errors - important for accurate fire-control.
- back-up modes should heading, or pitch and roll, or Doppler velocity become unavailable.
- no maintenance adjustments at any support level.
- no special test equipment at the flight line: BITE localizes malfunctions to the LRU and SRU levels.
- CDU provides serial digital outputs of system navigation, guidance and other data for use by external avionics

<u>Description of the AN/ASN-128 and the SHIU</u>

The AN/ASN-128 Lightweight Doppler Navigation system consists of three boxes that weigh only 30 pounds in total and have a predicted MTBF of over 2100 hours. The auxiliary display unit, the SHIU, weighs only 2.0 pounds, and has an MTBF greater than 10,000 hours.

The design of the AN/ASN-128 was centered on making maximum use of the existing heading and attitude sensors and complementing these with a DRVS, a computer, and suitable controls and displays.

A Doppler radar consists of an antenna, RF transmitting and receiving components, and electronics for processing the Doppler frequency shifts and converting these into the desired data format. A body-mounted antenna was selected to avoid gimbals that would otherwise add considerable size, weight and complexity. Four beams of R.F. energy directed downward symmetrically around the nadir are sequentially transmitted by a single antenna toward the ground, and the back-scattered energy is then sequentially received and processed. Although three beams are sufficient to determine the three orthogonal components of helicopter velocity, the AN/ASN-128 uses a fourth redundant beam to enable self-checking of the overall reliability of the velocity measurement.

The antenna must be mounted on the helicopter's underside to enable its beams to illuminate the ground, and thus installation and removal is generally more difficult than that of an electronics box mounted in an avionics bay. The antenna unit was therefore designed to contain the minimum amount of electronics for maximum reliability,

(> 12,000 h) but it does include all transmit and receive R.F. components so that wave-guide runs within the fuselage are avoided. A Kearfott developed printed grid antenna was used to eliminate the weight and cost of waveguide arrays. Installation was further simplified by integrating the radome into the antenna. All electronics required for velocity processing are contained in a separate electronics box, the SDC, located in the avionics bay. All regulated voltages required by the RTA, SDC, CDU and SHIU are provided by a single power supply in the SDC. A single card in the SDC sequentially converts the synchro outputs of the existing heading, attitude and true airspeed sensors into digital form for use by the computer in the CDU.

All computations for the AN/ASN-128 including velocity coordinate transformations, navigation in both UTM and latitude and longitude coordinates, steering signals to ten destinations, and BITE functions, are performed in an LSI digital computer located in the CDU. This approach minimizes the cost of the RTA and SDC when only these two boxes are used as a DRVS.

Two types of BITE test are provided; continuous and manually initiated tests. The CDU and most portions of the SDC and RTA are continuously tested, and the overall SDC and RTA are tested from end-to-end during the manually-initiated test. All test results are displayed on the CDU: system status, the failed box, and the failed module within that box. "GO" is displayed when no failure has occurred. End-to-end test takes only 18 seconds and does not interrupt navigation and guidance computations.

The system has been in production since 1978 and to date over 1000 LDNS systems have been ordered of which 600 have been delivered to the U.S. Army for installation in the AH-1S Cobra and the UH-60A Black Hawk helicopters. The system has been selected by, and is being manufactured for several other countries, including Germany, Spain, Greece and Australia; additional orders have been received from these countries for over 600 systems.

Mission Effectiveness

The above section describes some of the trade-offs leading to the system architecture, performance and physical characteristics of the navigation systems. The system designer together with the operational personnel must also examine the effectiveness of the system in performing the planned mission. To accomplish this a multi-segment mission is examined in the following section to identify the various requirements of the navigation system. Figure 4 shows the selected hypothetical yet typical mission scenario for a utility helicopter. This is followed by a discussion of the capability of the AN/ASN-128 to satisfy these requirements.

The postulated mission commences at the Forward Area Rearming and Refueling Point (FARRP). Coordinates in either UTM or Lat/Long format of the planned destinations or other preselected stored data should be rapidly insertable into the navigation system. A dual coordinate capability is desirable should target handoff be required. Rapid deployment unencumbered by requirements for lengthy alignment times is essential. BITE check should be rapid yet provide high confidence. The helicopter will pass the Forward Edge of the Battle Area (FEBA) enroute to Lz Tiger for a supplies drop. Once past the FEBA minimum radiated emission is desirable to avoid ECM. Enroute the flight crew must be provided with present position, off-track error, distance to go, true ground speed and track angle. On landing at Lz Tiger, it is assumed that power would be turned off requiring that selected destination stored data not be destroyed. Should the battle environment not allow a touchdown landing the system must display hover guidance data. Enroute to Lz Argo, enemy action may cause the helicopter to depart from its planned course. Instantaneous call-up of present position to allow reporting of the encounter, and updated guidance to Lz Argo must be provided. Having arrived over Lz Argo which is a point of known position, a means to quickly update for any accumulated navigation error is desirable. The navigation system must be configured to continue navigation throughout the update to avoid error buildups should higher priority tasks delay completion of the update.

After updating at Lz Argo and while enroute to the next planned destination, Lz Judy, the aircraft is rerouted to an unplanned landing zone - Lz OX. The system must be capable of in-flight alternate destination entry and selection.

Having completed the mission function at Lz OX the system should provide the ability simply to call up the prestored destination Lz Judy and provide all guidance and steering commands required to reach that LZ.

Enroute to Lz Judy a target of opportunity is observed, namely a pontoon bridge over a stream. The ability to over-fly this target and automatically store and display its position is required. Should operational conditions prevent over-flight it is desirable to have an offset target of opportunity storage capability.

The revised flight path to Lz Judy requires flying over a large body of water with an undulating shoreline. A navigation system which automatically provides over-water compensation eliminates the need for the flight crew to manually activate a land-sea switch. This feature is operationally desirable and in fact mandatory for nighttime and all weather flying when passage over water is not detectable.

For night flying it is essential that the navigation system controls and displays be compatible with night vision goggles.

Should the water conditions be extremely smooth, thereby not providing adequate radar return, the system should be capable of storing last "good" velocity, or when provided with True Air Speed (TAS) be able to compute and store most recent wind speed for navigating in these periods of Doppler "Memory."

Departing Lz Judy the helicopter may encounter a rainstorm. It is mandatory that the navigation system be insensitive to echoes from near-in targets like rain which could provide erroneous velocity inputs.

Approaching the FEBA it is desirable to obtain a final position update to ensure that the helicopter will penetrate in a safe area. This updating again must be simple and quickly performed.

After safely passing through the FEBA and prior to reaching the FARRP, the ability to perform an in-flight BITE check would enable rapid ground maintenance action (if required) upon landing thereby permitting rapid redeployment.

Rapid redeployment doctrine requires that those elements of the navigation system having the lowest reliability be located in the most easily accessible location, thus allowing minimum flight line level maintenance times.

The AN/ASN-128 navigation system is highly compatible with the above mission needs.

- Its navigational features include:
  - Either UTM or Lat/Long operation for Target Handoff
  - Storage of up to four "Targets of Opportunity"
  - Ability to enter and store up to ten destinations. Storage is retained with prime power removed.
  - The following guidance signals:
    - CDU
      - Ground speed and track angle
      - Track angle error and distance off course
      - Distance, Bearing and Time to go to destination
      - Wind speed and direction
      - Present position (Lat/Long or UTM)
    - SHIU
      - Track angle error or distance off course
      - Ground speed
      - Distance to go
      - 3D hover guidance
- It provides many unique operational features, some of which are:
  - Automatic over-water compensation
  - Automatic wind computation when provided with TAS
  - Reversion to last remembered velocity when in Memory
  - Immunity from rain effects
  - Capability to erase stored destination data
  - Warm-up time of less than 5 seconds
  - A complete system check using BITE without the need for external GSE. The BITE function may be exercised in flight without degrading navigation and guidance
  - An error code on the CDU display identifying the defective LRU and SRU
  - Backup modes of operation to overcome loss of information from:
    - Vertical gyro
    - Heading reference
    - Doppler radar
  - A display which is compatible with night vision goggles

In addition, because the radiation patterns are directed nearly straight down and the radiated power is very low (20 mW) the system has extremely low ECM detectibility.

IMPACT OF EXPANDED AVIONICS REQUIREMENTS

Although the AN/ASN-128 entered production as recently as 1978, the pace of an ever increasing enemy threat both in lethality and in quantity of air forces and armor has significantly enlarged the mission requirements for future helicopters. The requirement to routinely conduct operations at Nap Of the Earth (NOE) altitudes in marginal weather under reduced visibility has placed ever increasing demands upon the onboard avionics.

More accurate and sophisticated fire control systems coupled to LASER designators, FLIR and highly accurate attitude and velocity reference systems are required. To counter the ECM threat, improved and multi-frequency communication systems have been developed and are being deployed. These demands have significantly increased the number of electronic systems on board helicopters, resulting in the problems of increased crew workload, cockpit space requirements, system cost and weight.

Use of Digital Data Bus Technology

The increasing amount of electronics systems requires the transfer of large amounts of digital data at very high rates. To meet this requirement, digital signal processing techniques and architectures are being applied to interconnect sensors, displays and controls and multiple avionics microprocessors at the subsystem level.

The standard specification developed by the United States military for serial data multiplex transmission is MIL-STD-1553A/B. Military aircraft of the 1980's will rely heavily on multiplex data buses for information distribution in the design of their avionics architecture. In addition, the recently released MIL-STD-1760 specifies that both expendable and non-expendable stores be capable of interfacing with the host aircraft via a -1553B digital multiplex (MUX) bus. The -1553 digital multiplex bus provides a means of transferring data and commands from remote avionics over the same transmission line - in this case, a shielded twisted pair. The use of a digital data bus reduces weight and improves reliability because less wire and fewer connectors are required. Digital transmission techniques provide a higher data capacity, selfcheck on each transmission and reduced susceptibility to electromagnetic interference.

Key to the development and utilization of the digital data bus is the MIL-STD-1553 Bus Controller and Multiplex Remote Terminal (BC/RT) which provides the interface between the aircraft 1553 MUX system and the on-board avionics unit in which it is contained. Singer-Kearfott has developed a multiplex terminal unit of universal applicability for use in avionic LRU's. The BC/RT Module described in the following section utilizes high-density microcircuits and a flexible two-part architecture which is virtually independent of host peculiar requirements, and is therefore usable in a wide range of applications with little or no modification. It is designed to be located in the host LRU since the entire module is packaged on a single card and requires less than 10 watts.

The BC/RT can operate as a controller or as a remote terminal. The mode of operation is selectable by a Master/Slave logic signal or via the -1553 mode command for dynamic bus control transfer.

When designated a remote terminal, the BC/RT is responsive to all -1553A and B command-response requirements. When designated as a bus controller, the BC/RT initiates and supervises all data exchanges over the dual redundant -1553 serial data bus. Data storage and retrieval at the host parallel data bus is via direct memory access.

Features include:

a. Packaged on a single card module

b. Performs as a Bus Controller and/or Remote Terminal capable of executing -1553B Mode and Illegal Command Word Processing

c. Contains separable Word Processing and Message Processing/Microcontroller sections.

d. Word Processing section:

- Transparent interface for -1553A or B compatibility.
- Interfaces with dual redundant MUX buses. Performs all fast response front-end channel functions, including word validation, and command and data word detect and decode.
- Provides all necessary parallel data with the sense and control signals required by the Message Processing/Microcontroller section.

e. Message Processing Microcontroller section:

- Transparent architecture for interfacing with various hosts including those with microprocessors
- Programmable
- 16-bit bidirectional data bus

- 16-bit address bus
- DMA capability to interface with host microprocessor
- Provides memory protection and ability for indirect addressing of host main memory.

Figure 5 is a block diagram showing the two part architecture of the BC/RT and identifies the large number of functions which are packaged on the single card in less than 50 square inches by the optimum utilization of LSI technology.

Communication with the host is via a 16-bit bidirectional data bus. A 16-bit address bus and direct memory access circuits perform the functions of storage and retrieval of data to-and-from host memory. Other significant circuit elements are a 16-bit terminal bus for fast response data manipulation and transfer, and the -1553 Status Word and Last Command Word registers. In addition, an on-board scratchpad register and incrementer are provided.

Some examples of the application of digital data bus technology to advanced avionics are the U.S. Army's AN/ASQ-166 Integrated Avionics Control System (IACS) and the AN/ASN-137 Improved LDNS.

The U.S. Army Avionics Research and Development Activity (AVRADA) developed the AN/ASQ-166 IACS which is characterized by:

1) use of digital data bus to interconnect sensors, controls and displays
2) integration of programmable controls and displays, and
3) widespread application of microprocessors at the subsystem level.

The AN/ASQ-166 IACS provides the pilot with a means of controlling and displaying communications, navigation such as the AN/ASN-137, and identification (CNI) equipments with a single integrated control-indicator. All CNI equipments are remotely located and their information is transferred by a dual-redundant -1553 data bus.

The basic IACS consists of five units - two control-indicators, one status indicator and two interface units. Figure 6 shows a block diagram of a typical IACS installation. The controlled avionics, shown at the right of Figure 6 are remotely located in the avionics bay of the aircraft. The present IACS system can accommodate up to ten radios and associated communications, security equipment and the AN/ASN-137 Doppler Navigation Set.

The Primary Control-Indicator integrates the control and display functions of all ten CNI and associated security equipments and the Doppler Navigation Radar. This integration is accomplished by time-sharing both controls and displays of a large number of previously unrelated systems. Time-sharing of controls and displays conserves panel space by using fewer controls and displays to accomplish the necessary avionics control functions.

The Secondary Control-Indicator provides a minimum capability for emergency situations.

The U.S. Army has embarked on a development program to extend the IACS integration concept from the CNI functions to the remaining cockpit functions. This program has been designated the Army Digital Avionic System (ADAS). IACS is a subset of the ADAS system. Elements of ADAS have been installed in a UH-60A Helicopter Systems Testbed for Avionics Research (STAR) and are undergoing flight test. The AN/ASN-137 is one of the avionic systems included in the ADAS installation in STAR along with the IACS.

Description of the AN/ASN-137

Another example of the use of advanced digital data bus technology that has been developed by the U.S. Army is the Singer-Kearfott AN/ASN-137 Doppler Navigation System.

Overall design of the AN/ASN-137 was based upon two requirements:

- provide a system that has -1553 interface, drives a Projected Map Display System (PMDS) and can operate with an integrated display such as IACS, and
- retain as much of the AN/ASN-128 hardware design as possible to minimize development costs and to have maximum commonality with the AN/ASN-128 LRU's, SRU's, and logistics support equipment.

The changes required for the AN/ASN-137 to meet the above requirements do not affect the Doppler radar portion of the AN/ASN-128. Therefore the Receiver-Transmitter-Antenna (RTA) and a major portion of the Signal Data Converter (SDC) need not be changed. Avoiding RTA changes simplifies retrofit of the AN/ASN-137 into helicopters already containing the AN/ASN-128.

The need to provide navigation, guidance, and velocity data when operating with an integrated display such as IACS instead of the AN/ASN-128 CDU required the computer and memory in that CDU to be transferred to the SDC.

A simplified CDU was developed and is available for those applications of the AN/ASN-137 where IACS is not provided. A major design feature of the simplified CDU is that its operation is the same (except for PMDS related operations) as the AN/ASN-128 CDU to minimize flight crew retraining and chance of confusion when flight crews operate different helicopters.

The -1553 A and B interface was added to the expanded SDC to provide a digital communication capability between the AN/ASN-137 and other avionics in the helicopter. Locating the -1553 interface in the expanded SDC was also desirable since its design would be simplified by having direct access to the computer bus. To handle the increased computational load of the -1553 interface and other avionics (i.e., projected map drive equations, offset targeting, etc.), the AN/ASN-128 program memory was increased from 8,000 to 16,000 16-bit words while still providing about 3000 words of spare memory. The Random Access Memory was increased from 500 to 1000 16-bit words providing about 200 spare words.

To provide for a variety of on-board sensors and displays, a "Program Plug" has been added to the outside of the SDC. Each helicopter designed to accept the AN/ASN-137 will have a captive cable that connects to the Program Plug, and reprograms the interface within the AN/ASN-137 to be compatible with the configuration in that particular helicopter.

A block diagram of the overall AN/ASN-137 system is shown in Figure 7. Some of the key system features are listed below:

- provide -1553 two-way multiplexed bus interface
- can be controlled either by an integrated control/display system such as IACS, or a simplified version of the AN/ASN-128 CDU
- RTA unchanged for ease of retrofit
- Program plug enables operation in different helicopter avionics configuration without changes
- Drives AN/ASN-99 Projected Map Display System (PMDS)

A photograph of the RTA, Simplified CDU and Expanded SDC is shown in Figure 8. The Expanded SDC and Simplified CDU have the same installation requirements as their AN/ASN-128 counterparts, thereby simplifying retrofit. Physical characteristics of the AN/ASN-137 and optional SHIU are listed in the table below.

AN/ASN-137	WEIGHT		VOLUME		POWER	MTBF
	kg	lb	cm^3	in^3	(W)	(h)
DRVS: RTA	4.76	10.5	7,752	473	12.0	12,237
SDC	7.04	15.5	9,896	603	75.0	3,723
DRVS Total:	11.80	26.0	17,648	1,076	87.0	2,855
Simplified CDU	2.82	6.2	3,676	224	30.0	4,584
TOTAL:	14.62	32.2	21,324	1,300	117.0	1,760
SHIU	0.91	2.0	1,067	65	15.0	10,000

The AN/ASN-137 completed its engineering development phase in late 1980, and was integrated by Kearfott with the AN/ASQ-166 IACS, and the AN/ASN-99 Projected Map Display in early 1981. The system started U.S. Army flight test in mid-1981, and will be ready for production start-up in early 1982.

SYSTEM CONSIDERATIONS FOR HIGH ACCURACY APPLICATIONS

Improved Navigation Accuracy

Figure 9 shows the position error of the AN/ASN-128 or AN/ASN-137 Doppler Navigation Systems when a typical helicopter magnetic compass is used to provide heading. An error of one degree, (one-sigma), is characteristic of such compasses, resulting in a CEP of approximately 1.3% of distance traveled. For this system configuration, the position error after 50 kilometers of flight is 650 meters, as shown by the solid line in the figure. The dotted line shows the error build-up when an improved heading reference, with a heading error of 0.5^o (one-sigma) is used. This accuracy can be achieved by careful compensation of magnetic compass deviation error and by data stabilization of helicopter pitch and roll effects on heading.

The increasing performance requirements of advanced attack and scout type helicopters has led to the need for a heading reference with 0.25^o (one-sigma) accuracy and thus requires the use of a new system approach.

The SKH-3700 Attitude Heading And Navigation System (AHANS)

In response to the need for a more accurate heading reference Kearfott has developed the SKH-3700 AHANS. The AHANS is a single-box system that utilizes strapdown inertial sensor technology to provide very accurate heading, attitude, acceleration and angular rate data via a -1553 MUX I/O. The AHANS combines velocity data from the AN/ASN-128 or AN/ASN-137 with its own inertial sensor data in an optimum Kalman filter mechanization. The combined system has a heading accuracy of better than 0.25° (one-sigma).

The AHANS consists of a strapdown inertial sensor block, a powerful computer (SKC-3121) programmed in a High Order Language (HOL), I/O, and power supply. Two two-degree-of-freedom "dry" CONEX* gyros, and three "dry" pendulous accelerometers make up the sensor block. Figure 10 shows a block diagram of the AN/ASN-128 operation with the AHANS. Figure 11 shows the AHANS LRU, typical electronics cards, the sensor block, the gyro and accelerometer.

The Kalman filter in the computer software is a 24-state mechanization. It models both system navigation parameters and combined Doppler/AHANS component error sources. Through the optimum combination of inertial sensor data, Doppler velocity data, and external position observations (when available), the AHANS refines estimates of vehicle position, velocity, and attitude during flight and continually calibrates Doppler/inertial component error sources.

This mechanization provides a rapid reaction "scramble" or in-flight startup capability so important in military operations, without impacting navigation accuracy later in the flight. In addition, improved navigation is available during periods of backup mode operation, such as Doppler in memory.

A prototype version of AHANS is currently in flight test by the U.S. Army, where it is operating with an AN/ASN-128. Navigation results have been excellent, with position errors less than 400 meters after 2 hours of flight. Heading errors have averaged less than 0.25° (one-sigma) during this time.

The AHANS with its powerful digital computational capability when interfaced via a -1553 MUX bus with the Doppler, and controlled by IACS will provide not only navigation but also velocity, heading, attitude, acceleration and angular rates with the accuracies needed for future requirements.

Externally Referenced Systems

Another level of system performance improvement and operational capability can be obtained by combining the AN/ASN-128 or the AN/ASN-137 with externally-referenced systems such as the Global Positioning System (GPS), Position Location and Reporting System (PLRS) or the Joint Tactical Information Distribution System (JTIDS). As was shown in Figure 9, the position error of a Doppler radar navigation system increases with distance traveled. The position error of an externally-referenced system is essentially bounded and predictable. A combined Doppler/position sensing system has several important advantages: The position error is bounded to that of the position sensing system, errors in the Doppler navigation data (velocity and heading) are calibrated by the position sensing system, and finally, accurate navigation and guidance data continue to be available to the flight crew even when operation of the position sensing system is interrupted for any reason including intentional interdiction by the enemy. As future position sensing systems become mature, one can expect increased interest in combining these systems to obtain more optimum navigational accuracy and reliability.

TECHNOLOGY ADVANCES TO MEET NEW REQUIREMENTS

As electronic components decrease in size, weight and power requirements, the demands for increased capability and functions tend to increase to fill the available space created. For example, the Doppler radar echo contains altitude (distance above the ground) information in the phase of its modulation. As microprocessor and VLSI technology develops, it will be possible and desirable to add an altimeter function to the Doppler navigation radar basically without penalty.

CRT displays, when used in combination with microprocessors, are providing new versatility in the amount, format and quality of the information which can be provided to the pilot. The use of multi-color CRT's adds the dimension of color to the data, allowing highlighting of critical data on the CRT face. The advent of flat panel displays will save volume, weight and power over current CRT's.

The increase in memory capacity, which Magnetic Bubble Memories and VLSI offer, will allow digital map storage which can be displayed on a CRT or flat panel together with position, track-made-good, and track-to-go superimposed from the onboard navigation system.

As mentioned above, the progress in circuit miniaturization will produce major changes in the size, weight, power consumption and capabilities of avionic hardware. Microprocessors are being used in almost all signal processing applications, replacing bulky analog devices such as filters and inductors. Discrete microwave circuits using waveguide interconnections are being replaced by printed circuit equivalents (called "Stripline" or "Microstrip"), with integrated active devices embedded in the printed circuitry.

* Trademark of The Singer Company.

Kearfott has been in the forefront in applying these technologies to its product lines, especially on printed microstrip antennas. A printed microstrip Doppler radar antenna is being developed to replace the AN/ASN-128 printed grid antenna. It is only 0.030 inches (0.8 mm) thick and can be applied to conform to an aircraft surface. This new version weighs 2 lb less and requires only 40% of the present volume.

Figure 12 graphically illustrates the benfits of technological breakthroughs to avionic systems. The criterion used here was the weight of Doppler navigation radar systems produced since 1950. The dramatic decrease in weight between 1950 and 1960 is primarily due to replacing vacuum tubes with transistors. Between 1960 and 1970 the weight decreases were accomplished by improvements in antenna technology, resulting in smaller antennas and much simpler antenna stabilizing gimbals. In addition, improvements in RF hardware permitted the replacement of pulsed high power transmitters by more efficient solid state sources using FM-CW modulation techniques. Since 1970, further weight reductions were achieved by replacing mechanical antenna stabilization with digital data stabilization, replacing slotted waveguide antennas with printed antennas, and by the extensive use of digital logic and MSI/LSI components. Each reduction in weight was accompanied by a reduction in cost and size.

The current rapid pace of developments in VLSI, microprocessors, microwave integrated circuits and advanced digital signal processing techniques will afford opportunites for even greater degrees of avionic sensor integration.

By virtue of these cost and size reductions, we will see operational capabilities currently supplied in sophisticated fixed wing aircraft, extended to future helicopters and drones.

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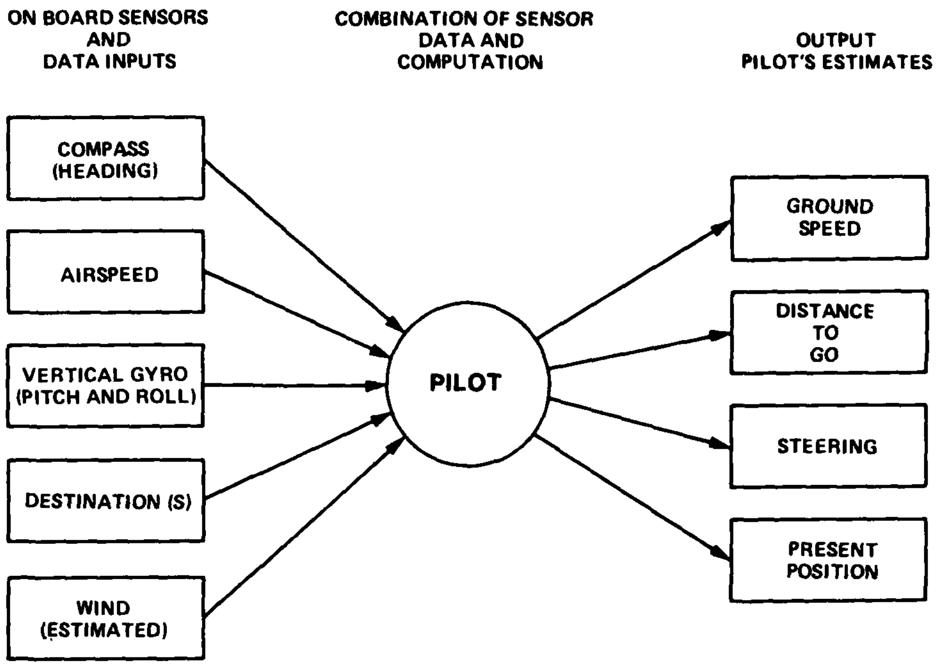
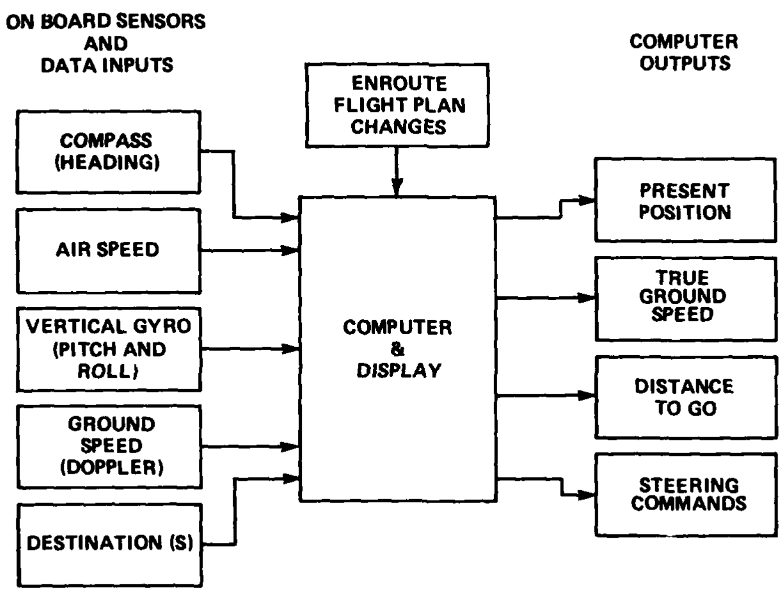
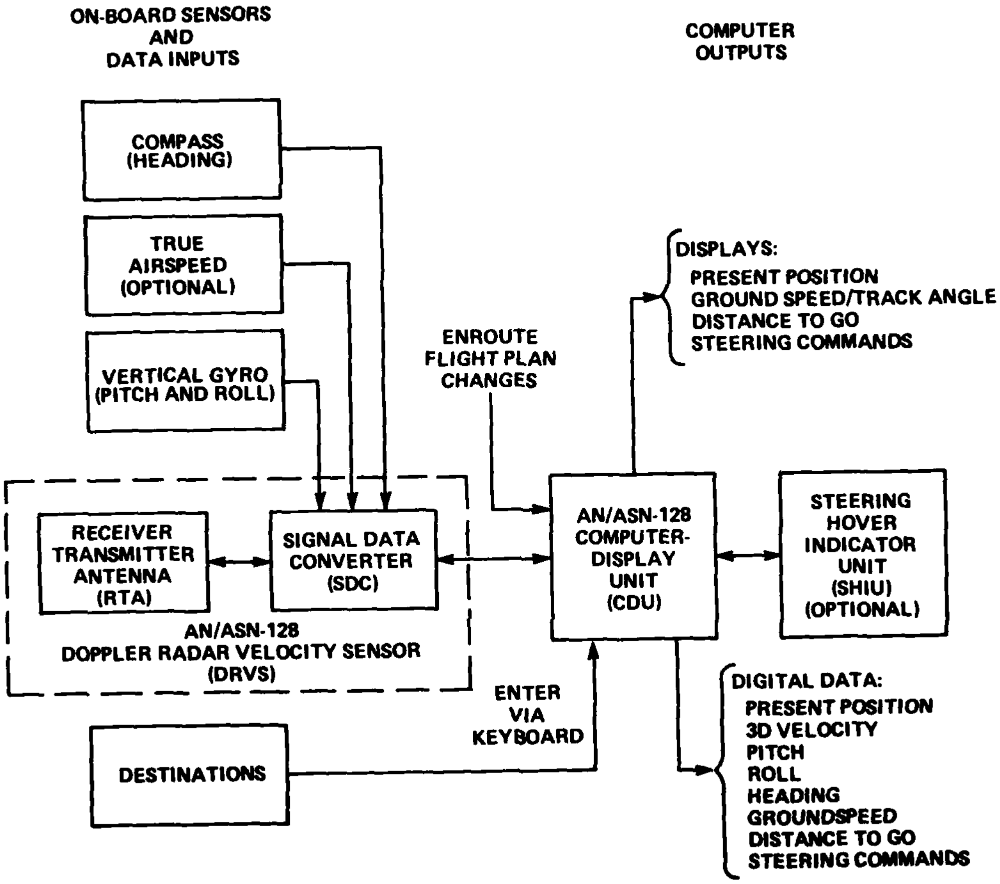
4. Buell, H., "Doppler Radar Systems for Helicopters," published in the Journal of The Institute of Navigation, Vol. 27, No. 2, Summer 1980.



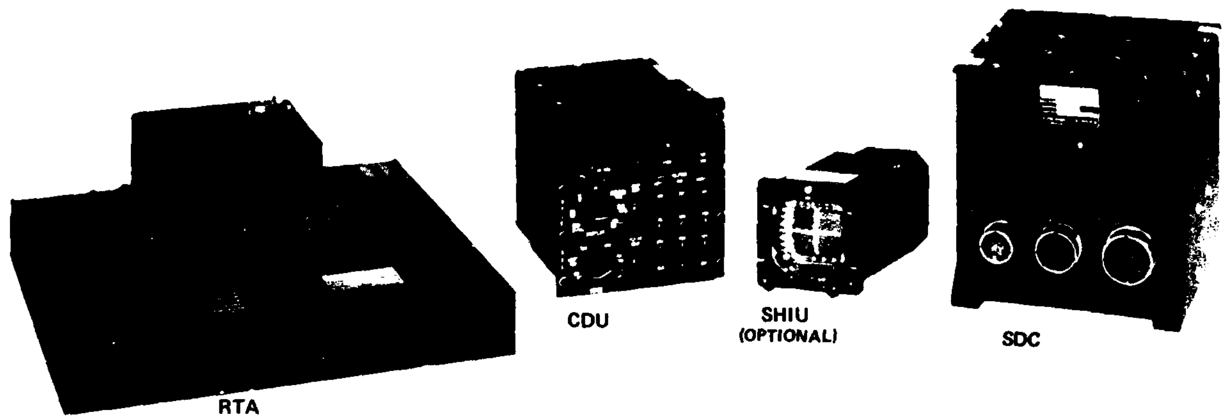
NAVIGATION USING MANUAL INTEGRATION OF INPUTS
FIGURE 1A



NAVIGATION USING COMPUTER INTEGRATION OF INPUTS
FIGURE 1B



AN/ASN-128 BLOCK DIAGRAM
FIGURE 2



AN/ASN-128 LIGHTWEIGHT DOPPLER NAVIGATION SYSTEM
FIGURE 3

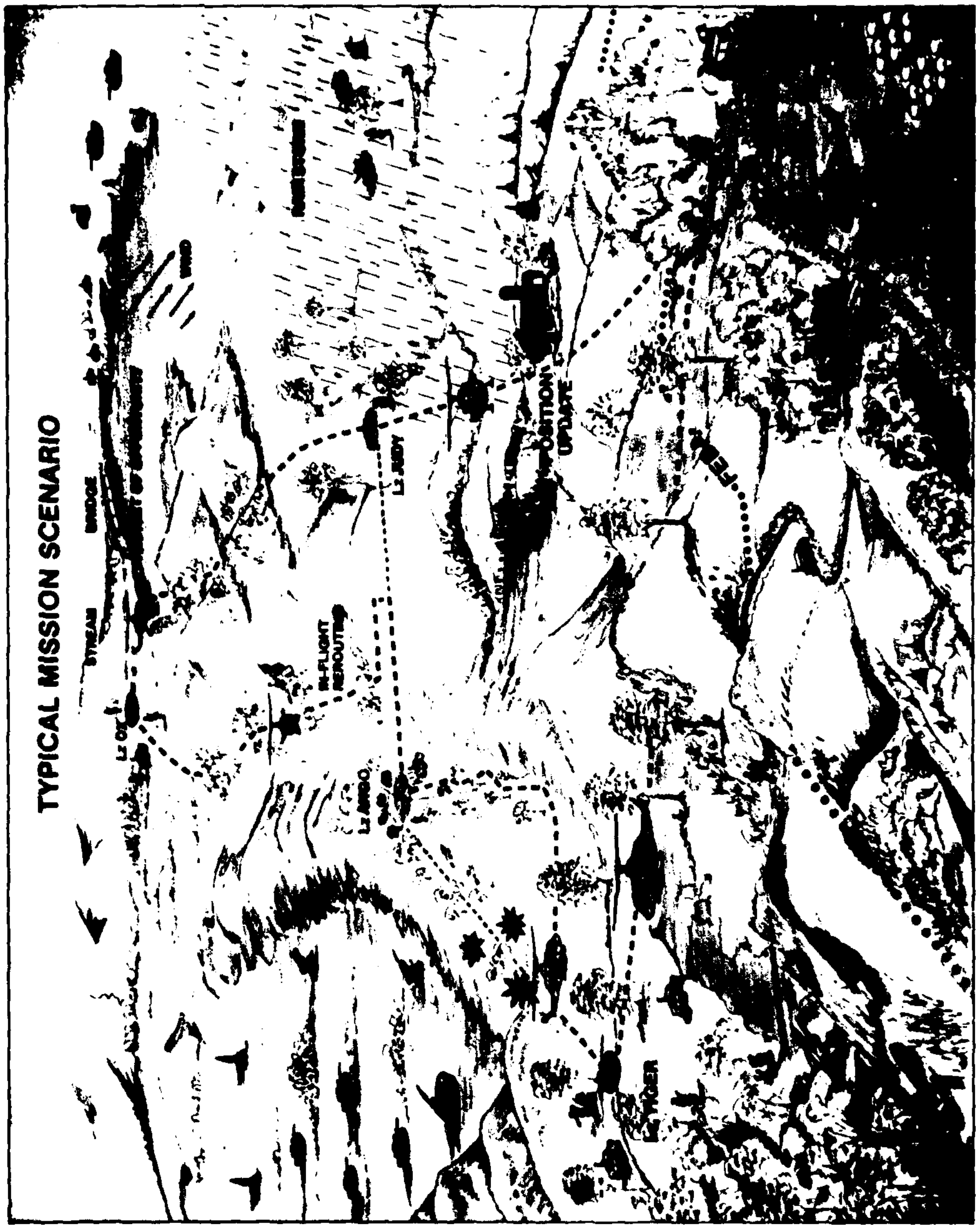
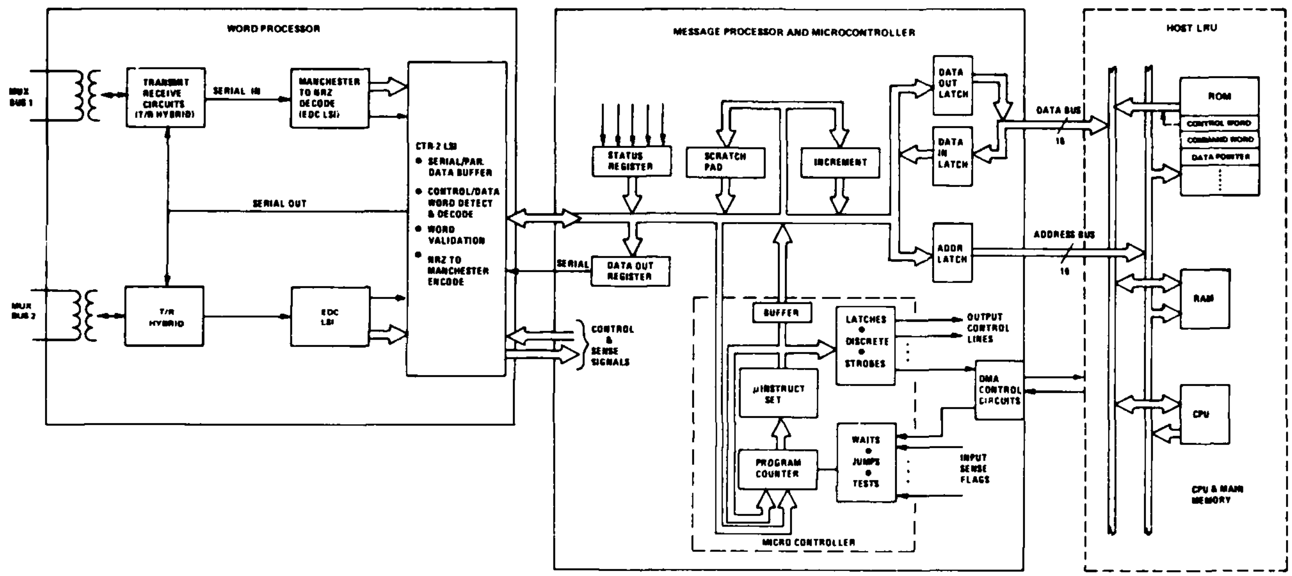
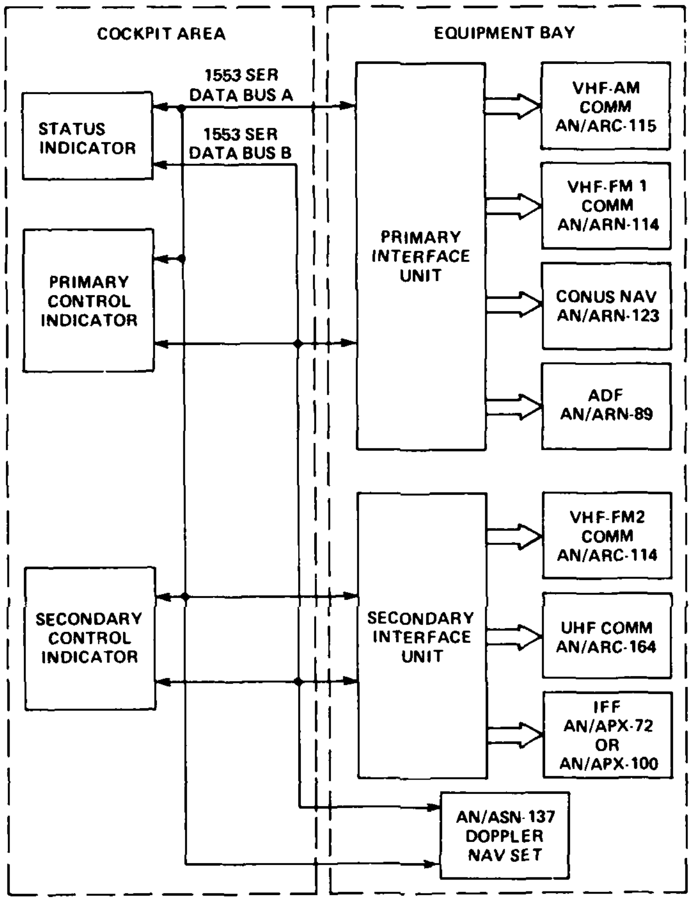

TYPICAL MISSION SCENARIO
STREAM
BRIDGE
IN-FLIGHT REROUTING
POSITION UPDATE

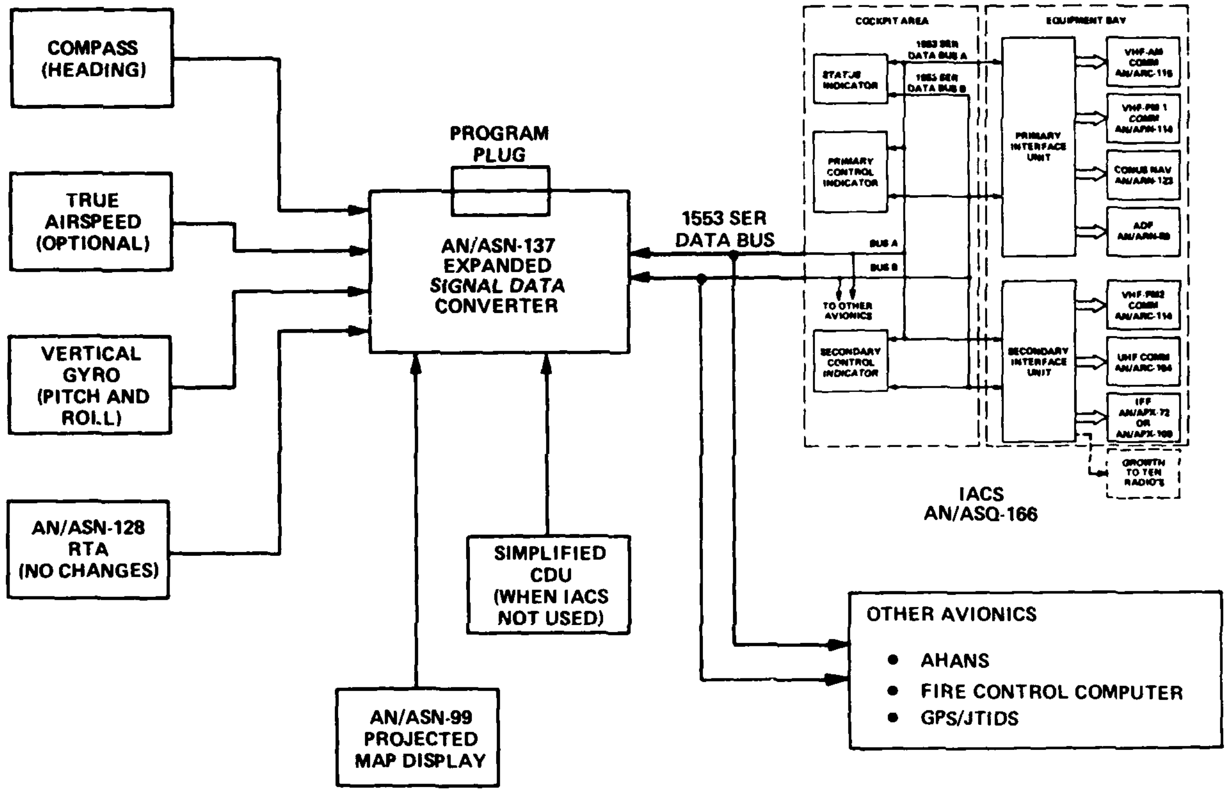
FIGURE 4



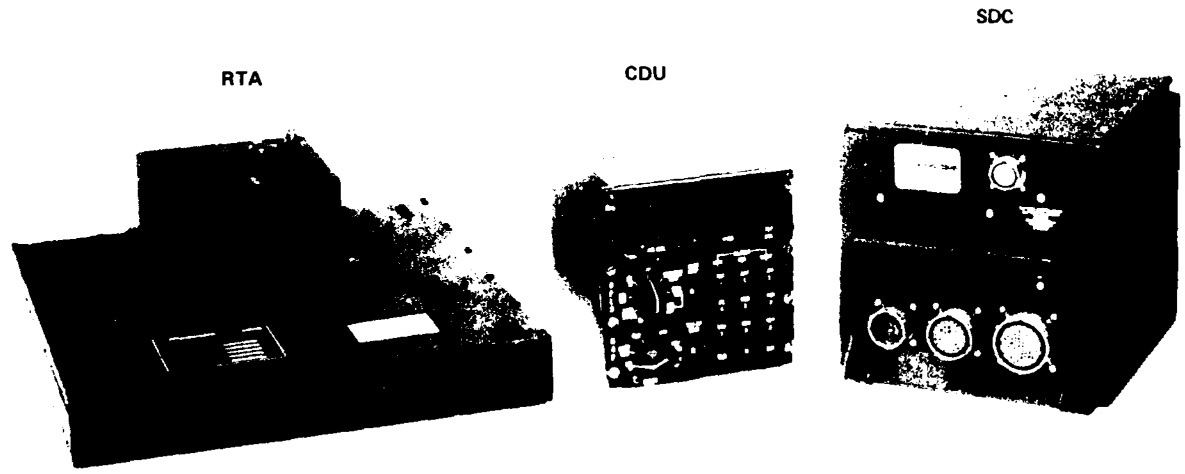
MUX SYSTEM BLOCK DIAGRAM
FIGURE 5



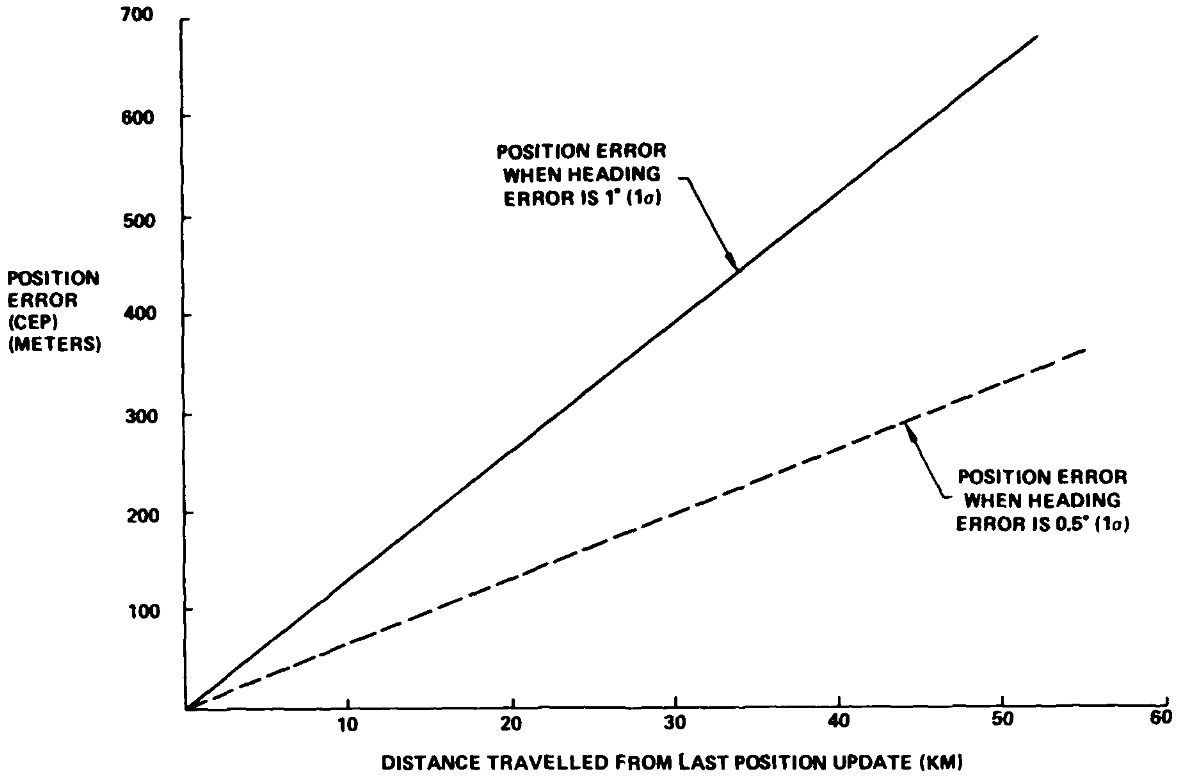
INTEGRATED AVIONICS CONTROL SYSTEM, AN/ASQ-166
FIGURE 6



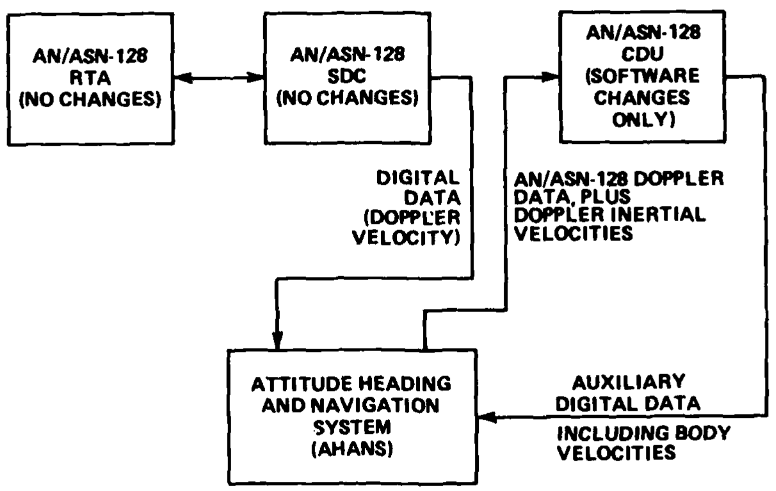
AN/ASN-137 DOPPLER NAVIGATION SYSTEM BLOCK DIAGRAM
FIGURE 7



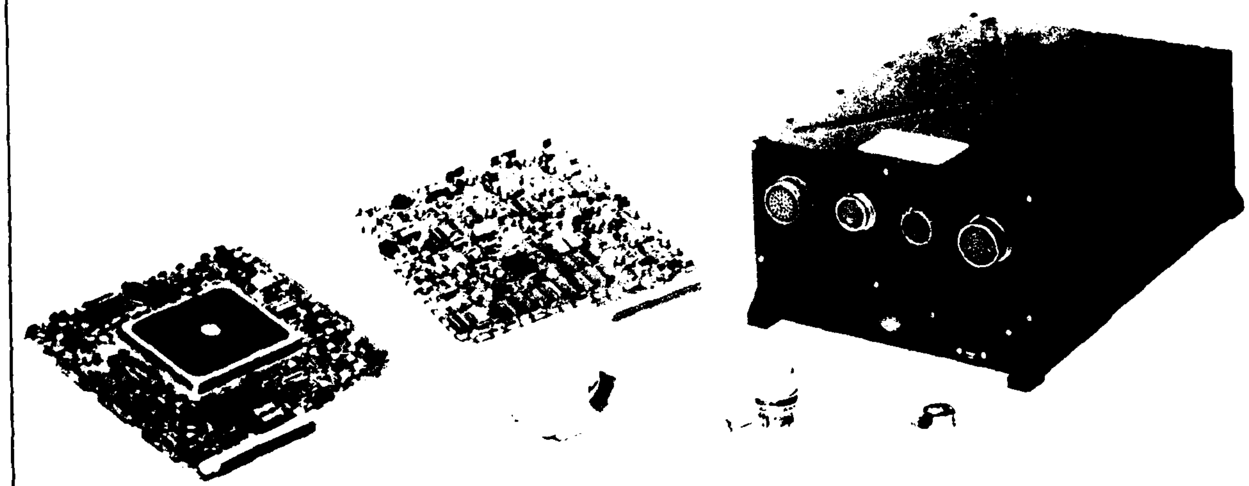
AN/ASN-137 DOPPLER NAVIGATION SYSTEM
FIGURE 8



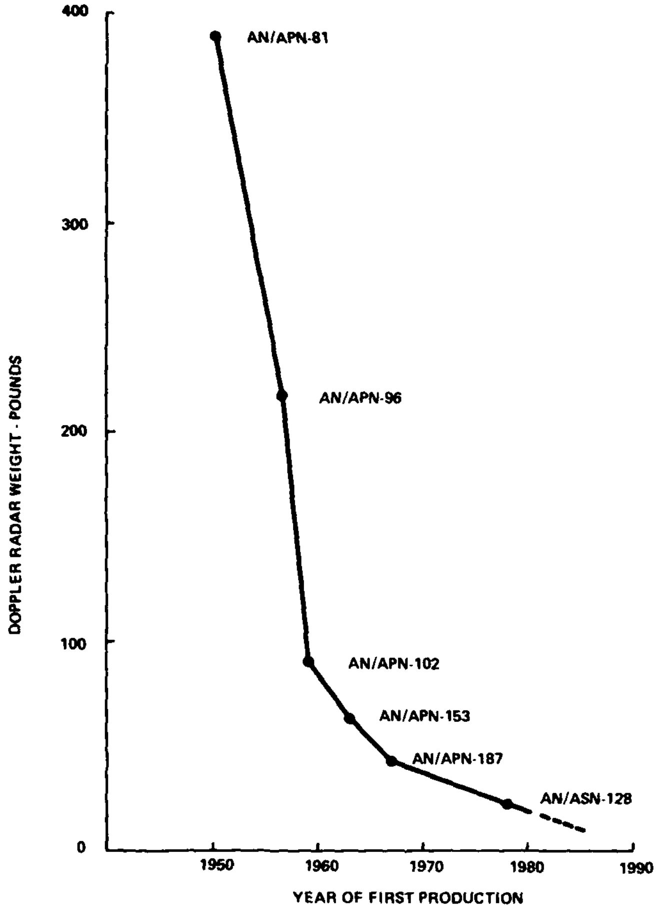
POSITION ACCURACY OF AN/ASN-128 OR AN/ASN-137
FIGURE 9



AN/ASN-128 AND AHANS BLOCK DIAGRAM
FIGURE 10



KEARFOTT ATTITUDE HEADING AND NAVIGATION SYSTEM (AHANS)
FIGURE 11



IMPACT OF TECHNOLOGY ADVANCES ON DOPPLER RADAR WEIGHT
FIGURE 12

DESIGN FOR MAINTAINABILITY AND PLAN FOR MAINTENANCE

by

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SUMMARY

The purpose of this paper is to increase the awareness and emphasize the need to:

(1) Consider maintainability of systems/equipment in the design phase of the hardware.

(2) Do in-depth planning for the actual maintenance effort commencing very early in the weapon system programs.

Maintainability and maintenance planning must get more attention during a system's life cycle (the earlier, the better). There is evidence that often in the past the pressure of other factors such as operational requirements, reliability considerations, budget, and schedule have overshadowed adequate maintainability and maintenance considerations.

This paper presents the reasons for early and adequate attention to maintenance factors and conversely the penalties for not providing them adequate attention.

The U.S. Department of Defense Directive No. 5000.40, Reliability and Maintainability, will be briefly discussed.

Maintenance concepts, specific maintainability design considerations and maintenance planning elements will be outlined and some lessons learned will be presented.

INTRODUCTION

Reliability and Maintainability (R&M) are the two major factors that determine the availability of equipment. When equipment costs were much lower and when there were adequate maintenance resources (trained technicians, parts availability, etc), lack of availability due to low reliability or poor maintainability was simply overcome by buying more hardware.

A recent report by the U.S. congressional watchdog agency, the U.S. General Accounting Office (GAO), stated that "as much as 90 percent of the total cost of a weapon system are ownership (supply and maintenance) costs and only 10 percent are acquisition costs: (Reference A). The U.S. Air Force spends one third of its budget on maintenance. (Reference C, p. IV-1). Maintainability is a result of design; therefore, it just makes sense to stress maintainability in this design process. Vice Admiral E.R. Lemour, Chief of the U.S. Naval Air Systems Command even goes so far as saying that if a system is to have a life cycle of 20 years or more, maintainability should take precedence over reliability as the prime design factor. (Reference D, p. 60).

Reliability and maintainability must be considered together because one can usually be improved but often only at the expense of the other.

Some of the reasons that maintenance/maintainability have not in the past been given their deserved share of attention are:

(1) Early emphasis on maintainability through design requires up-front money. With shortage of defense funds, money has often not been available in the early part of programs. This is an obvious reason that a program should be planned around a life-cycle cost concept. If the life cycle cost concept had been rigidly applied, more funds would have been applied to maintainability early in the programs to save funds in later years.

(2) The most visible sign of accomplishment of a program office is when it gets it's bright new shining airplane on the runway. The next most important evidence of accomplishment is the declaration that the weapon system is mission capable. There has been no event where someone has had to stand up and tell the world that the weapon system is maintainable.

(3) Program Managers and designers are responsible for a program usually for only a few years of its total life cycle. Would they possibly not do things differently if they knew they were going to be responsible for the supportability of the system for their total career or life of the system.

DOD DIRECTIVE 5000.40 "RELIABILITY AND MAINTAINABILITY", JULY 8, 1980

To improve operational effectiveness and to reduce cost of ownership through improved supportability of weapon systems, the U.S. Department of Defense issued this directive to the U.S. Military Departments and the defense agencies. It states these objectives for reliability and R&M maintainability activities:

(1) Increase operational readiness and mission success.

(2) Reduce ownership cost through reduced maintenance and logistic support requirements.

(3) Limit manpower needs for operation and maintenance.

(4) Provide R&M management data.

(5) Ensure cost effectiveness of R&M investment.

The directive requires that R&M be addressed at each major milestone in the system acquisition process. Projected deficiencies of operational readiness, mission success, maintenance manning, and logistic support must be addressed. Another important provision of this directive is that follow-on buys of additional hardware and spares as well as modifications will be to the same specifications as the original equipment.

The reaction to this directive by Gen Bryce Poe, II, the recently retired commander of the U.S. Air Force Logistics Command, was: "The raising up of these (reliability/maintainability) to equal level of emphasis with schedule and performance is long overdue." (Reference 8).

PAYOFF OF EFFECTIVE MAINTENANCE

A $25 million aircraft is not much of a threat to a potential enemy if it is not mission capable. A plane can be rendered not mission capable during the time required for its maintenance or during the time it is awaiting a serviceable subsystem or component to be retained from the repair activity. There is just not enough money in the world to buy "spare" airplanes to compensate for a high non commission rate.

A typical inertial measurement unit can easily cost $100,000 to $500,000. Assume a depot repair requirement of 60 units per month and a total repair pipeline time of 30 days including transportation. Based on an arbitrary average value of $200,000 per unit, a reduction of the need to evacuate 10% of the failed units to the depot could save $1.2 million in pipeline spares costs recurring costs of approximately $36,000 per month in depot repair costs. If the pipeline time of the remaining 54 units per month requiring repair could then be reduced by five days, an additional $2.0 million in pipeline spares costs could be saved.

Let's continue with this example. Assume that a portion of the reduction of the pipeline time was due to easier maintenance at the depot because of more maintainable hardware and to better diagnostic techniques and test procedures. A 10% reduction of depot labor and repair parts could save $600 per unit, or $32,400 per month. Better diagnostic and test procedures could save 10% of depot equipment costs or approximately another million dollars.

Summary of above cost savings based on 20-year program:

	One Time	Recurring
Reduced reparables (60 to 54 units)	$1.20 Mil	$8.64 Mil
Reduced pipeline (30 to 24 days)	2.00	--
Reduced manpower & parts	--	7.77
Reduced support equipment	1.00	--
Sub Totals	$4.20 Mil	$16.41 Mil

TOTAL $20.61 Million

NOTE: This is a potential savings for only one of the many avionics boxes on an aircraft.

MAINTENANCE CONCEPT

The maintenance concept must outline to what depth repair is to be accomplished at which level by the techniques which can be accomplished with the skills expected to be available at each level.

(1) Projected level and locations of maintenance. There are a lot of trade offs which must be made to determine where various maintenance tasks should be done. Traditionally we have had three levels of maintenance (organizational or flight line, intermediate or base level, and depot level). There are some proponents for elimination of the intermediate level shop due to problems with support equipment and retaining skilled technicians. If the intermediate shop is eliminated, we'll have to change our complete concept of spares and parts support.

Generally, depot level support should be planned to be done in military repair depots. This is normally more cost effective as well as better assures response if there is a surge in requirements. Interservice and foreign military sales support should not be overlooked.

Interim contract support using contractor excess production capability in the early phases of a program can be advantageous. It should be planned for early in the program. There is always the danger of contractors being more concerned about his new production rather than his repair commitments.

The assumption that a manufacturer's new build capability can be copied for depot level repair is false. New build and repair are not the same. Depot repair normally requires a more extensive capability than new repair such as more in depth diagnostic capability. Depot requirements can however usually be satisfied by supplementing and adapting the manufacturer's equipment and methods.

(2) Fault Isolation and System Testing Approach:

As avionics become more complex and as it becomes more difficult to train and retain skilled maintenance technicians, Built-in Test (BIT) capability is becomming more desirable. This is particularly advantageous for flight line and intermediate level maintenance.

Automatic Test Equipment (ATE) is normally advantageous for testing of large volume complex avionics.

To compensate for slight test equipment incompatibilities and to assure a useful product is provided to the user, a pyramid of tolerances must be established. This pyramid allows test result tolerances which are less forgiving at intermediate level than at flight line level and less forgiving at depot level than at intermediate level.

(3) Equipment Maintenance Approach:

Equipment is maintained through either repair when broken and/or preventive maintenance. Avionics maintenance usually relies on repair and then only repair on-condition, i.e., repair or replace only the component which exhibited a failure.

The key to cost effectiveness maintenance is to plan the proper maintenance approach at the proper level of maintenance.

(4) Maintenance Demonstration and Verification Tests:

In the past, schedules often did not allow for an adequate maintainability demonstration and if they included only enough time for those which were success oriented, i.e., no time was permitted for corrective action.

To be effective, this demonstration and verification must be done using maintenance technicians of the skill level expected to be available to perform the actual maintenance tasks and if possible in the actual maintenance environment.

PLANNING EFFORT MUST ADDRESS THE FOLLOWING RESOURCES AND CONSIDERATIONS

(1) Support Equipment (SE):

Support equipment usually represents a major expense especially if automated test equipment (ATE) is used. The ATE associated software often costs more than the equipment itself. A software support capability (for changes/updates) must be developed and retained throughout the life of the program. Care must be taken not to over-automate due to these costly factors.

The SE must receive the same level of attention relative to its design and ability to be supported throughout the life of the program as the airborne hardware.

(2) Supply Support:

Good supply support requires a good prediction of the requirements and timely procurement. Availability of parts then is based on timely receipt of parts which meet specifications.

Prediction of parts requirements and future verification and correction have all too often been based upon failure modes experienced during system manufacture which is not necessarily that which can be expected during the operational phase. Logistics and maintenance technician can normally better predict parts requirements based on similar in-service hardware.

Timely procurement can be a problem, particularly with increased lead times. Care must be taken to assure parts being procurred are of the configuration to be included in the final design.

Lead times for parts continue to rise. This is due to causes such as lack of adequate manufacturing capacity (skilled manpower and equipment availability) and limited critical materials availability.

(3) Transportation and Handling:

We'll never know how much damage to inertial navigation/guidance equipment is caused by handling and during transportation. We do know that inertial components are extremely sensitive to physical damage. Logic must tell us that every reasonable action should be taken to minimize this possibility of damage.

This must be taken into consideration during design of the airborne equipment itself and its shipping container. It also requires that all who handle this equipment, both in the shipping container and outside it, are acquainted with its delicacy and are trained in proper handling.

Equipment being transported is not available for use. Spares must be bought to fill this transportation pipeline. It therefore behoves us to plan the transportation cycle between user and the repair sites to take minimal time.

(4) Technical Data:

Technical data is another expensive resource required for maintenance. Care must be taken to assure that the data is written to the expected skill level of the maintenance technicians. It is said that often it is at too high a level because it is written by graduate engineers for lesser educated workers.

The most neglected phase of technical data development is the verification of that data by maintenance personnel in the actual environment in which it is to be used. This must be done in sufficient time to allow corrections and changes to be incorporated in the data prior to its need.

(5) Facilities:

The maintenance concept should be so that there are relatively little peculiar facility requirements at other than at the repair depot level.

Peculiar facility for maintenance requirements have to be identified very early in the program due to the long lead time for design and construction and in some cases physical stabilization.

(6) Personnel and Training:

The complexity of the equipment continues to increase; however, the quality of the manpower required to maintain that equipment is not increasing. It becomes more difficult and lengthy to train and develop skills in personnel. When these skills are developed, it is difficult to retain these people because they become attracted to the non military job market.

Not only must maintenance training be planned for the initial weapon system activation, but follow-on training must be planned to retain an adequate number of trained personnel to compensate for attrition.

Although usually expensive, the use of Contractor Engineering Technical Services (CETS) should be considered to assist the maintenance personnel in becoming proficient in the early part of a program.

(7) Management Data:

Collection and analysis of management data must be a planned activity.

Early in a program, decisions are based upon management data furnished by the contractors and management data collected from similar in-service systems.

Good management data permits us to accurately determine how we are doing against established goals and milestones as well as provide a baseline to measure modifications or program changes.

(8) Safety:

All maintenance tasks must be analyzed relative to safety to both the maintenance technicians and the hardware being maintained.

Solvents specified for cleaning should only be those which are not suspected carcinogens.

Of current concern is the large amount of damage to delicate electronic components from static electricity. This problem has the potential to get bigger as the newer technology electronics come into use.

(9) Security:

Security must be consciously considered in maintenance planning as well as in operational planning. Provisions have to be made to safeguard classified hardware and data throughout the maintenance cycle.

(10) Environmental Considerations:

Governments and the public are becomming more sensitive to the quality of the environment. Recently, environmentalists have expressed concern that freon is destroying the ozone layer of the earth's atmosphere. Freon is used in large quantities for cleaning inertial instruments parts. Some of this freon is not recaptured after use and therefore does escape into the atmosphere.

In maintenance planning, we must be alert to other similar situations whereby the discharge or residue of harmful materials should be controlled or alternate methods or material should be used.

DESIGN CONSIDERATIONS

Design considerations which impact the Maintenance Concept and Maintainability: Design trade-offs must be made to optimize performance, weight-size, cost, schedule, reliability, and maintainability. Remember, the unit must be maintained/repaired many times and for many years after the other design considerations are usually forgotten.

(1) Packaging of the G&C (Guidance and Control), (Single Package vs Multiple Packaging):

Packaging has a great bearing on the maintenance concept and the number and cost of pipeline spares G&C units or sub units which must be procured. Packaging as much as possible in a single housing saves handling of several different units and assures compatibility of subassemblies contained in the single package, i.e., they can be married at manufacture and subsequently at the repair depot. On the other hand, when one subassembly fails, it could mean returning the whole unit to the repair depot, depending on the maintenance concept and spares provisioning. If each subassembly were contained in separate packages(black boxes) then only the failed subassembly need be removed from the aircraft and sent for maintenance.

Weight and size of the airborne equipment are other factors which help dictate packaging. As more females enter the maintenance workforce, we may need to reevaluate some human engineering factors which were developed based on average male physical characteristics.

(2) Accessibility of the G&C System on the Aircraft:

On one of our major aircraft, the pilots seat must be removed to obtain access to the inertial platform. High failure items should be placed where they can easily be removed and replaced. Units should be placed so that indicators or connections used for checkout or serviceing in place are readily accessible.

(3) Electronic Circuit Accessibility:

Fault isolation of electronics provide some special problems to the maintenance technician. Some of the problems which should be considered by the designers are:

(a) Test point accessibility external to the black box.

(b) Ability to isolate to a single faulty electronic module or part.

(c) Circuit tuning should be able to be accomplished by adjustable components rather than through hard wired components.

(4) Accessibility of parts within the system.

Parts subject to high failure should physically be located where they can be replaced without extensive disassembly or removal of numerous other parts.

Frequcntly removed electronic parts or electronic modules should be mated using connectors rather than hardwiring. (These should be permanently keyed to preclude damage caused by an electronic module being able to be installed in the wrong slot).

(5) Standardization:

Standardization should be considered starting with the avionics system itself. It should subsequently be considered through each indenture of hardware down to the last bit and piece. Advantages of standardization are:

(a) Maintenance time reduced due to maintenance technicians experience being applied to reduced span of hardware.

(b) Less range of parts required to be purchased and managed.

(c) Higher probability of part being available.

A recent U.S. GAO report shows that on a (non U.S.A.F.) program, 2000 varieties of parts were procurred that differed very little from parts already in stock. (Reference A)

CONCLUSIONS

Program Managers and designers should take advantage of the vast amount of maintenance expertise that exists in the using commands, in the various logistics support offices, and at the responsible maintenance depots. They should be consulted early in the design phase as well as invited to participate in the various design reviews normally held during an equipment program. This should apply not only to airborne hardware design but also to support equipment design.

Our challenges however, are not only to apply the lessons learned from past experience but to anticipate the effects of ever changing technology and other influences on supportability of a weapon system through its life cycle, such as the increased lead time for parts procurement and the increased inability to retain skilled maintenance technicians. Weapon systems and their subsystems become more complex and due to technology improvements become obsolete sooner however, due to their replacement cost are expected to be used and supported longer.

In short, we must make more complex equipment more maintainable by fewer skilled people, using less costly support equipment, less parts, for a longer period of time. This can be done only through good maintainability design and good maintenance planning.

DEFINITIONS: (From U.S. Dept of Defense MIL-STD-721B)

ACCESSIBILITY: A measure of the relative ease of admission to the various areas of an item.

AVAILABILITY: A measure of the degree to which an item is in the operable and committable state at the start of the mission, when the mission is called for at an unknown (random) point in time.

CAPABILITY: A measure of the ability of an item to achieve mission objectives given the conditions during the mission.

MAINTAINABILITY: A characteristic of design and installation which is expressed as the probability that an item will be retained in or restored to a specific condition within a given period of time, when the maintenance is performed in accordance with prescribed procedures and resources.

MAINTENANCE: All actions necessary for retaining an item in or restoring it to a specified condition.

RELIABILITY: The probability that an item will perform its intended function for a specified interval under stated conditions.

REFERENCES:

The following are referenced in the texts of this paper as noted by reference letter and page in parenthesis; i.e., (C, p1) refers to reference c, page 1.

(a) Coates, James, Agency Report Attacks Tank Repair Features, Columbus Dispatch, Columbus, OH, U.S.A., July 2, 1981, page A-4.

(b) Poe, Bryce II, Gen, U.S.A.F. Letters to the Editor, Avionics Week and Space, Hightstown, N.J., U.S.A., November 10, 1980.

(c) RAM Principles and Application, Study Guide, U.S. Army Management Engineering Training Activity, Rock Island, IL, U.S.A., 1977.

(d) Technical Survey: Reliability/Maintainability, Aviation Week & Space Technology, Hightstown, N.J., U.S.A., October 6, 1980.

Other reference which would be quite helpful for maintainability designers and maintenance planners is:

(e) Blanchard, Benjamin S., Logistics Engineering and Management, Prentice Hall, Inc., Englewood Cliffs, N.J., U.S.A., 1974.

PRODUCTION VERIFICATION TESTING (PVT) OF GUIDANCE AND CONTROL SYSTEMS FOR HIGH RELIABILITY

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SUMMARY

This paper covers the testing of equipment for highly integrated guidance and control systems. Specifically, it looks at a departure to the usual acceptance testing by examining in some detail a technique called production verification testing--or PVT. First, the paper reviews some of the recent literature and experience. Further, the paper through computer simulations, examines various facets of PVT. Particular emphasis is directed toward the implication of 100% testing of all systems prior to delivery through the use of 'N' sequential failure-free cycles. Failure is defined as a relevant or non-relevant occurrence which precludes continuation of testing. Finally, the paper presents some conclusions drawn from computer simulations. For example, it suggests the number of 'N' sequential failure free cycles that may be necessary. It provides insight into interpreting the results. Simulation results suggest a means for determining not only expected operational reliability but also process control problems during manufacturing.

LIST OF SYMBOLS

K	Constant Determined by Circumstances
H	Total Hours
LRU	Line Replaceable Unit
MTBF	Mean Time Between Failure
t	Total Mission Time Per Year
c	Number of Turn-on/Off Cycles
α	Growth Rate
λ_a	Failure Rate Due to Calendar Time (Aging)
λ_o	Predicted Failure Rate
λ_c	Failure Rate Due to Turn-on/Off Cycles
λ_t	Total Failure Rate
λ_Σ	Cummulative Failure Rate
PRVT	Production Reliability Verification Testing
PVT	Production Verification Testing
NM/HR	Nautical Miles Per Hour

PREFACE

The underlying reason for this paper is to present an analysis of production verification testing (PVT). The author's interest in the subject goes back a dozen years. Initial exposure to the technique was through the observation of its implementation to a major program. It was then called production reliability verification testing or PRVT. The results, compared to the usual burn-in tests can best be termed dramatic. Reliability of delivered equipment was significantly improved. The technique was neither widely understood nor used. Somewhat later a program was having difficulty passing reliability testing and delivering either failed or low mean time between failure (MTBF) systems. Remembering the results of PRVT we convinced management to institute a similar procedure. Again, there was dramatic improvement in fielded reliability. Flush with the taste of success, we were able to convince more program managers of guidance and control systems that they should be using this technique as a planned effort in the beginning rather than well into a contracted effort when deliveries were underway. Throughout the time period and up to the present we were perplexed. We knew that when we used this technique, reliability of fielded systems was higher than when PRVT was not used. The problem was we did not know how much testing we needed. Worse yet, the relationship between testing and fielded reliability was not quantifiable. It was for that reason that later efforts in this area were called PVT. The intent was to avoid the impression that PRVT was a replacement for reliability testing. With the wider use of PVT, both in the military and commercial marketplace, it was deemed appropriate to attempt an analysis of the method.

1. INTRODUCTION

Use of guidance and control technology for highly integrated systems--the theme of this conference--is determined in the final analysis by operational requirements. Clearly, one facet of this is performance. If a 0.5 NM/HR inertial navigation set is needed as part of a weapon delivery system, then providing one that performs at 2.0 NM/HR yields limited utility no matter how reliable it is. The converse is equally important. A highly accurate inertial system provides little utility to the user if it does not work. In antonomous integrated systems flying at low level--the price of low reliability is deadly if not expensive. Redundancy is the mathematician's trick for raising the probability of success. The result is a nightmare for the logisticians as well as those trying to balance a budget. The tradeoff between life cycle cost and operational readiness is ever present. If you will--the very topic of this session--affordability and survivability hinges on acquiring reliable equipment with confidence and at minimum expense. It should, however, be pointed out that this paper does not address or suggest changes to Defense Department directive or any service policies and specifications.

2. BACKGROUND

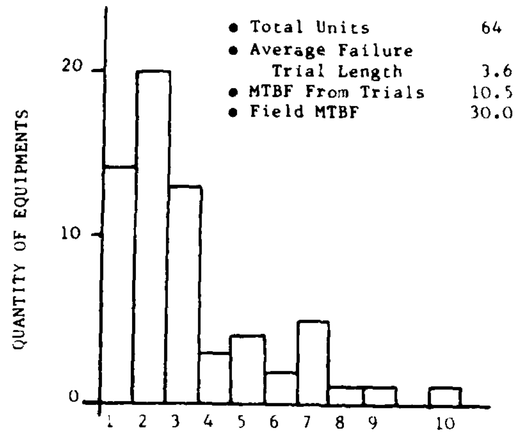
Acquiring reliable equipment is certainly no easy task as evidenced by the literature. Selby and Miller state that a reliability "...credibility gap exists between stated equipment reliability requirements and realized or realizable achievement." (1) They go on to say that this results in programs with unachievable requirements. In fact, they note that achieved versus specified MTBF is typically off about 10:1 and in some cases as much as 20:1. (2) Although these statements were made in 1970, the situation has not changed much. Mr. Gregory's editorial in Aviation Week noted the Defense Department's "...dissatisfaction with the failure rates of systems and subsystems." (3) The same issue of Aviation Week contains a technical survey which delves into reliability and points out specific problems. (4) Clearly, one important aspect of reliability improvement is through appropriate testing. Obviously, you design in reliability, not test it in; but, testing, if judiciously used, can prevent poor systems from entering service where the cost of a failure is magnified. For example, in one case where new systems were delivered with a failure rate of 80% and a corresponding $6000 average repair cost--the average cost increase for delivered systems was about $5000--not an insignificant cost.

In fact, the cost of repair only covers a part of the cost involved. A failed system places an undue strain on numerous resources. For a program in the early stages, it delays delivery of weapon systems. In addition, it causes the demand for additional units which in all likelyhood are not available since the production line is not geared to produce additional quantities. As more systems fail and are returned for repair, they use production facilities normally allocated for production of new equipment. The situation can, if left alone, get so out of hand that deliveries are dramatically reduced. To more clearly understand this situation--suppose that the production facilities are geared to produce 10 systems per month. Further, consider a situation where 5 systems a month are returned for major rework. Since a repair facility is probably as yet not available the failed equipment will displace new equipment or just get stored for later work. In either case, not a happy circumstance. If stored, the weapon systems delivery is potentially reduced by some 50%. If repaired on production facilities and presuming a worst case situation, then delivery of new equipment could be reduced by up to 50%--again leading to a reduction of weapon systems delivery. Problems like this do arise with varying degrees of severity. Delivery schedules are adjusted and means for repair both in plant and on site are incorporated--at additional cost to the buyer.

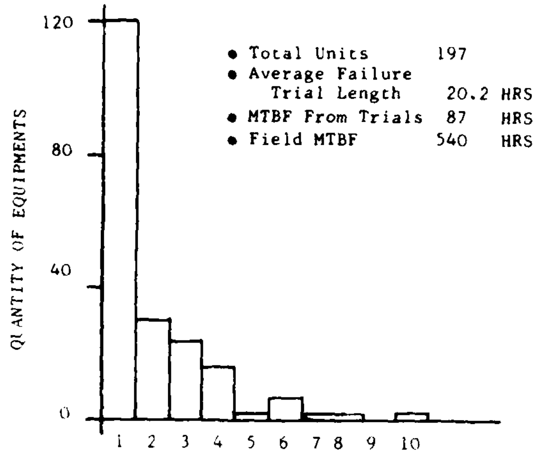
Once the equipment is fielded--the situation takes on a new complexion. High failures demand spares in excess of planned purchases or result in weapon systems that are not mission capable. Doing something about the problem is exasperating to say the least. Additional support services are needed. Along with this, there is a demand for more and more trained technicians in the field. The funds for these efforts are usually in short supply and trained people are harder to acquire and still harder to keep. As newer technology equipment becomes available, the argument is made for solving the reliability problem and acquiring improved performance. And, the cycle begins again. It would be simple but foolhardy to blame the contractor for these problems. Contracting for military avionics is difficult since the complexity is usually enormous; the development and production cycle is long; and the quantities are small. Clearly, what is needed is to devise some mechanism to insure delivery of reliable equipment. Specifying appropriate testing of all delivered equipment is clearly an essential element.

3. FAILURE-FREE TESTING

Hammer states that reliability demonstration testing of military electronics has drawbacks, not the least of which is poor correlation with field reliability results from the small sample of equipment selected for test and the dependence on relevant failure classification which is often inconclusive.(5) From Aviation Week "...Willoughby added that reliability testing only provides a 'snapshot' of the system that is not always a valid indication of what its actual performance will be. And, if you depend on reliability testing, it has to come late in the program when the configuration is stable, which is too late to do anything to fix problems."(6) Hammer suggests failure-free reliability testing based on the testing of all equipment produced--not just a small sample. Further, the test should be based on failure-free trials not relevant failures. The reason is that it provides the producer with a positive economic incentive to implement, in-line, the necessary corrective actions. As far as Hammer is concerned, failure analysis is performed solely for determining the cause of failure and follow-up corrective action--not for supporting non-relevancy rational.(7) Figures 1 and 2 show the result of requiring failure-free trials from his paper.(8) The figures are provided to show that depending on equipment and test conditions the shape of the curve may differ. The reader should note that during early trials there is a sharp rise in the number of failures. Further, the failure frequency drops and settles out. Finally, fielded MTBF is higher than the MTBF from the trials.



FAILURE-FREE TRIALS
FIGURE 1
10-HR FAILURE-FREE TEST



FAILURE-FREE TRIALS
FIGURE 2
50-HR FAILURE-FREE TEST

Anderson's report (9), prepared for the Flight Dynamics Laboratory, surveyed industry practice and opinions concerning environmental burn-in of avionics. 'N' sequential failure free cycles, where 'N' varied from supplier-to-suppler, was embodied in tests of all equipment shipped. According to the report, temperature cycling is often used to precipitate equipment defects while some suppliers also use vibration (53% had experience with random vibration). The report assesses the technical merit of environmental burn-in along two lines. First it surveyed industry and second it evaluated the effectiveness of this type of testing.

(a) Industry Survey Results (10) - The thermal cycle is the primary environmental screen. Cycle lengths of six to eight hours is most common. The number of cycles varies with four and ten cycles being quite prevalent. According to the survey, parts such as defective ICs and semiconductors accounts for almost half the failures. Workmanship, largely soldering, represents about a third of the failures. The use of temperature cycling and random vibration (not necessarily together) was considered the most effective in stimulating failures. As for the failure-free requirement, industry in general felt that:

(1) it should be used as a confirmation of system integrity,
(2) experience shows it is an exceptional tool,
(3) military customers should specify the failure-free period in a temperature/vibration environment and let the contractor select screens.
(4) an investigation of the length of failure-free operation is needed.

In addition, the survey noted that the greatest potential was in accelerating test time by increasing the frequency of cycling and reducing the duration of each cycle. For the near term (3-5 years), the survey revealed that stimulation of defects will continue to be the primary method of testing. For the moment, concern centers on what type (random) or sinusoidal) vibration and level to use and where to place it in the test cycle (i.e. before, and/or in the middle, and/or the end of the failure-free cycle). The far term is less clear. The possibility exists for a shift to combined environments testing which simulates the service or operational environment. (Note: the literature normally refers to this as Combined Environments Reliability Testing or CERT. As with PRVT and PVT, CERT is sometimes referred to as CET since in some cases it is not the intent to so much establish reliability as it is to find and fix design deficiencies.)

(b) Results - "Analysis of environmental burn-in tests shows that the selected LRUs display four common attributes: 1) a decreasing failure rate in the first few cycles (reliability improvement), 2) a relatively constant failure rate(subsequently no reliability improvement), 3) a 'reburn-in' characteristic for failed units (reliability improvement after failure and repair), and 4) a relatively large acceptance test rate at the end of environmental burn-in (performance test in environmental burn-in not as thorough as acceptance testing)". (11) The following figures, related to inertial systems, are extracted from the report to show various aspects of this testing. Figure 3 shows the production flow process; while Figure 4 shows the burn-in characteristics.(12) Note that in some cases the power on in Figure 4 may occur at both low temperature and at high temperature with off cycle before, in between and at the end. This leads to two operations in one complete cycle. Figure 5 shows the random vibration profile. (13) Note that the general procedure now is to use the expected vibration for the specific installation. This means random vibration for jet aircraft and sinusoidal for reciprocating aircraft and helicopters. Finally, Figure 6 shows the failure rate at the 10 cycle burn-in point.(14)

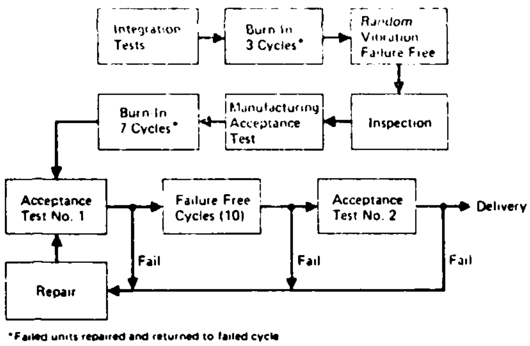


Figure 3 . INS Production Process Flow

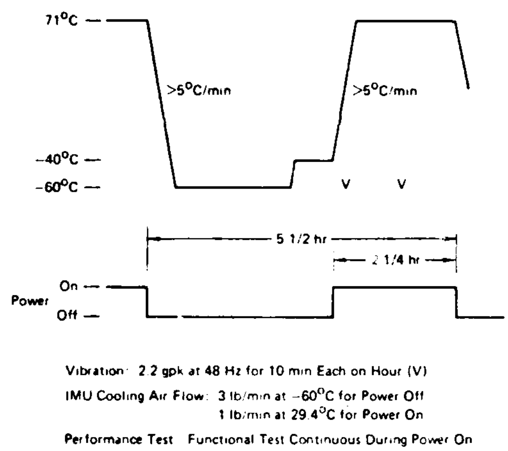


Figure 4 INS LRU and Set Burn-In Characteristics

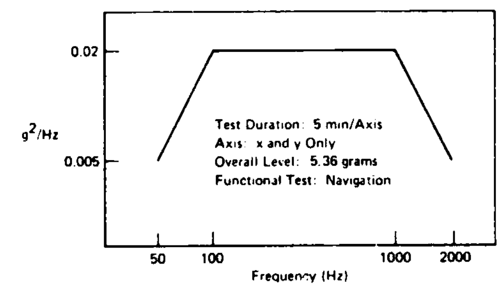


Figure 5 INS LRU Vibration Test

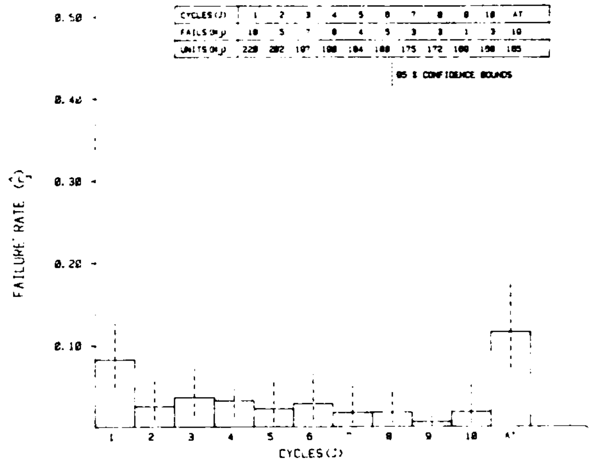


Figure 6 INS IMU Failure Rate for First Failure (10 Cycle Burn-In)

4. MODELLING FAILURE-FREE CYCLES

(a) Model Development - The initial attempt was to develop an equation or set of equations that portrayed the events involved in failure-free cyclic testing. The intent was to provide a simple quick solution that related failures to trials (or cycles) and reliability. The math involved got hopelessly involved. The approach was therefore taken to model the process on a computer using random number generators to represent the failure mechanisms. Figure 7 presents a simple version of the flow diagram used. Initialization parameters included the MTBF of the equipment, the number of cycles, the percentage of equipment expected to be bad-on-arrival (more is said about this later) and the number of equipments to be tested in any particular lot. Results of the process were tracked and stored throughout the test. For example, when a line replaceable unit (LRU) failed-the cycle in which it failed was stored. The number of total cycles needed to complete "n" sequential failure-free cycles was also accumulated and kept for later printout. For the model to have any value, it must replicate actual results with reasonable fidelity. Figure 8 shows a composite portrayal of typical results from a number of contractors on their recent test results. Note that in the early cycles there is a high incidence of failure.

However, as the number of cycles is increased--the frequency of failures drops and then settles to a constant failure rate. Modelled with a single failure rate mechanism, the results were as shown in Figure 9. The curve is relatively flat throughout. Thinking about this in retrospect, its clearly what you should expect given the conditions. The model was then adapted to include a second and very low MTBF operating along with the normal equipment MTBF. The way this was modelled was to have the low MTBF occur with a frequency defined as some percentage of bad on arrival. With a random number generator these equipments which were selected as bad had a low MTBF the rest had a high MTBF. Those with a low MTBF were repeated as "repaired" (in a computational sense) until they passed the sequential failure-free test. The shape resulting from this approach was very nearly in agreement with that of contractor results. What was needed, however, was a reasonable estimate of both MTBFs and how they operated.

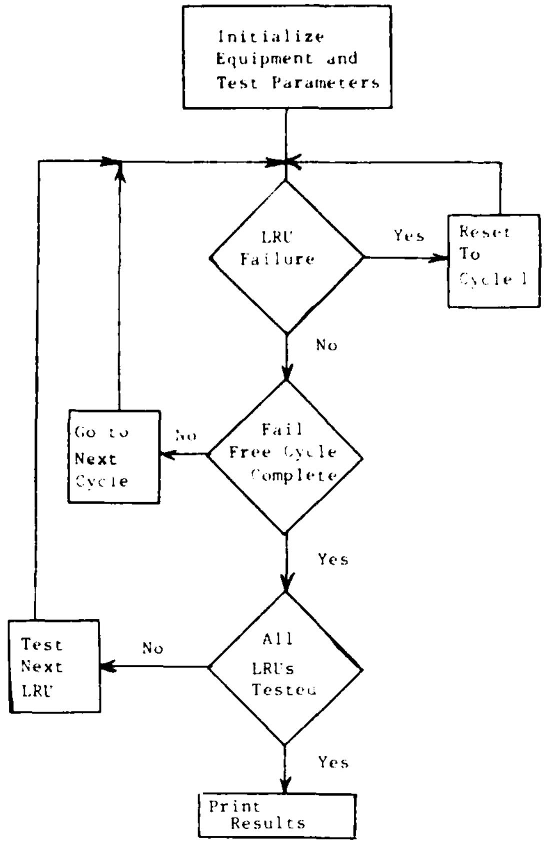


FIGURE 7

COMPUTER FLOW DIAGRAM-PVT

(b) Reliability - Expected equipment reliability was established by using a technique developed by the Charles Stark Draper Laboratory (CSDL)(15) and refined by the National Aeronautics and Space Administration (NASA) (16). The CDSL report, according to NASA, found that failure modes "...are due to mechanical, thermal, and electrical stresses caused by operating time, turn-on/off cycles, calendar time or aging, and environment..." (17) It is within this framework that the report distinguishes between predicted failure rate (λo) and total failure rate (λt) in actual system operational experience and relates one to the other. Eq (1) shows the relationship that exists between the various failure modes and total failure rate.

$$t\lambda_t = t\lambda_o + c\lambda_c + 8760\lambda_a \qquad (1)$$

The symbols were previously defined but the 8760 preceding "λ_a" corresponds to the number of hours in a year. For purposes of this paper, it is assumed for inertial systems that:

$$\lambda_a = \frac{\lambda_o}{10} \qquad (2)$$

$$\lambda_c = \frac{1}{500} \qquad (3)$$

This is based in part upon the data presented in the NASA report and in part by judgement. Users can apply this assumption as a first cut approach to their work--but, for detailed analysis a substantiated value for "λ_a" and "λ_c" is suggested. The resulting equation used for determing operating reliability is therefore:

$$\lambda_t = (t \ \ +876)\cdot\lambda_o + \frac{c}{500}\cdot\frac{1}{t} \qquad (4)$$

Given this equation - what can we infer? For one thing--in a commercial environment given that:

$$\begin{aligned} t &= 5000 \text{ hrs (per year of operation)} \\ \lambda_o &= 1/4000 \text{ (i.e.-4000 predicted MTBF)} \\ c &= 500 \text{ turn ons (in a year)} \end{aligned} \qquad (5)$$

the operating MTBF is about 2000 hours. For a nominal fighter environment given:

$$\begin{aligned} t &= 500 \\ \lambda_o &= 2.\ 1/4000 \\ c &= 500 \end{aligned} \qquad (6)$$

--the resulting operational MTBF is about 300 hours.

Note that the predicted λ_0 is doubled to account for increased temperature effects. This exercise is important because it not only explains the difference that exists between expected military and commercial failure rates but is also an important step in modelling the expected MTBF of a system when undergoing a sequential failure-free test. Two factors dominate. First, temperature doubles failure rate. Second, cycling the systems on and off (and allowing sufficient time between cycles--usually about an hour) stimulates failure modes (recall the previously noted industry survey results). Based on these factors, it seems that inertial systems with a predicted MTBF of about 4000 hours will have about 400 hour MTBF in a PVT chamber when exposed to temperature cycling.

(c) Model Validation - Data was collected from various manufacturers who had used different forms of PVT. The model was adjusted to reflect the number of systems tested and the total number of cycles. Cycle here is defined as one on-off period. This, in keeping with Eq (4), is an important factor since in many cases there are two on-off periods. In the first case, the equipment is subjected to a low temperature turn on and operation. The combination of a high and low temperature operation is considered one cycle for contract test purposes. The reader should therefore note that throughout this test--a cycle represents one turn on and turn off unless otherwise noted. The computer model was first adjusted so that the steady state failure rate was very nearly duplicated. This was done by taking the predicted failure rate and adjusting for temperature, number of turn-on/off cycles, and total operating hours. Since Eq (4) is based on yearly data--all numbers were adjusted to reflect such an operation. Then the bad-on-arrival percentage and the corresponding low MTBF was adjusted to nearly reproduce the actual experienced failures. Figure 10 shows the comparison of actual versus computed PVT results. To facilitate interpretation and preserve the identity of the manufacturers, the failure frequency data is a composite using a five point running average with the failure frequency unquantified. As can be observed, the model reasonably replicates the actual events for the circumstances encountered. The variations in the plot results from discrete failures occuring randomly (or pseudo-randomly in the model)

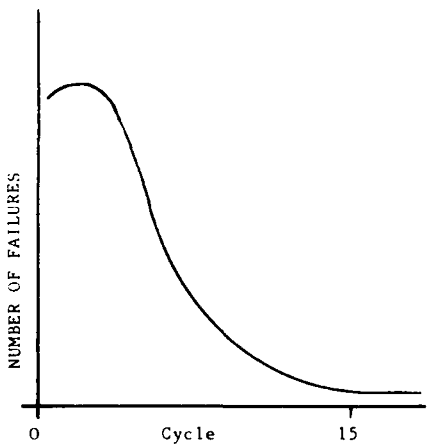


FIGURE 8

TYPICAL FAILURE PROFILE

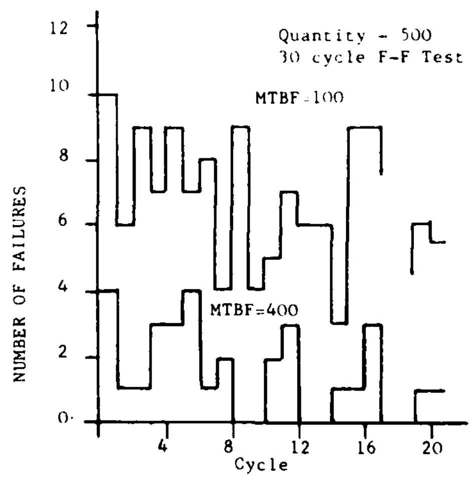


FIGURE 9

SAMPLE FAILURE PROFILE

5. INTERPRETING FAILURE FREE RESULTS

An examination of Figure 10 shows that there is a high failure frequency in the early cycles and that failure frequency settles to a steady state value. For the comparison case, the steady state situation for a 30 cycle test is achieved in about 15 cycles. The steady state failure frequency represents the inherent MTBF of the system under test at accelerated conditions. Remember the equipment is subjected to temperature extremes-turn on and off is frequent and operating time is high. Data gathered in this region can be used to roughly predict operational MTBF. To accomplish this the failure frequency over the steady state region should be accummulated for a substantially larger sample--on the order of one hundred or more systems. Clearly, thi should be over a well defined steady state region. Figure 11 shows a plot of failures versus test MTBF and assumes that the only failure mechanism present is operational failure frequency. That is to say there are no manufacturing defects that lead to bad-on-arrivals, i.e. early failures. If you have been collecting data on some 150 systems based on a 30 sequential failure free cycle test and fc nd that the last half of your plot of failure frequency to cycles produced fifteen failures at nearly steady state--then you could estimate that the systems had a test MTBF of about 200 hours. (Note the failures have to be doubled to translate 15 cycles to thirty and then multiplied by 100/150 to correspond to the curve). Using Eq (4) it is possible to compute λ_0 and then recompute a new total failure rate (λ_t) for expected operational tests.

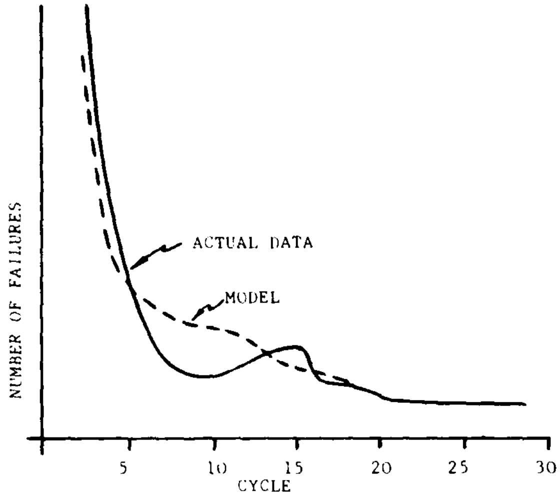


FIGURE 10

COMPARISON OF MODEL TO DATA

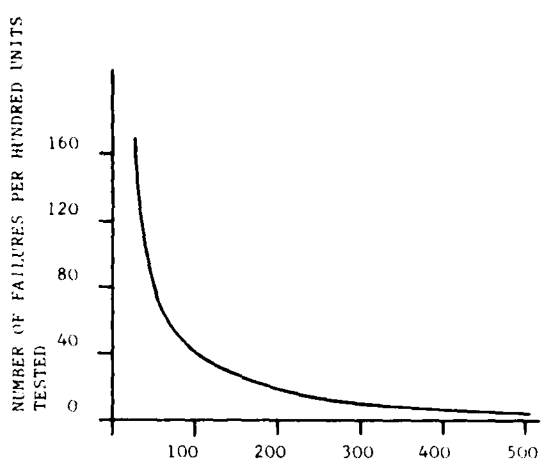


FIGURE 11

EXPECTED FAILURES - f(OPERATIONAL MTBF)

(a) Effects of Varying Early MTBF - Two questions are posed. Suppose you want to determine the effects of both variations in early MTBF and bad-on-arrivals. Figure 12 shows the effects of varying the early (or process control) MTBF. This figure shows the computer simulation of the random process for two different values of early MTBF--2 hours and 5 hours. In this simulation the total number of systems tested were 500. Bad-on-arrival was held constant at 15% and the test MTBF was 400 hours. PVT was based on 30 sequential failure free cycles. Simulation results were plotted by using a five point running average to smooth out the data. For the 2 hour MTBF--128 failures were encountered. Of these about 42 represented steady state failures traceable to the inherent reliability of the system. Two thirds of the failures are related to early or process control failures. Note that steady state was encountered somewhat prior to the 15th cycle. For the systems "tested" it took an average of 31.8 cycles to pass the necessary 30 sequential failure free cycles. In fact, 384 systems completed test without failure--95 systems completed testing between 31-45 cycles--18 systems needed 46-60 cycles and 3 units required between 61 and 90 cycles. This is mentioned in order to put the 31.8 cycle average in perspective and at the same time focus attention to the fact that testing can in some cases be quite lengthy. When early MTBF is increased to 5 hours with other factors held constant to the same previously noted values, the total "failures" encountered was practically the same. The major difference, however, was that the average number of cycles increased to 32.2--but more importantly the point at which steady state was attained shifted somewhat past the 15 cycle point. Comments and data from most contractors indicate that 15 cycles (as defined herein) is about the point at which steady state in reached. Discounting some of the variation in test methods (i.e. equipment reliability, turn on and off cycles and total operating time) and accepting the model as described, the simulation was suggested that early MTBF is probably between 2 and 5 hours.

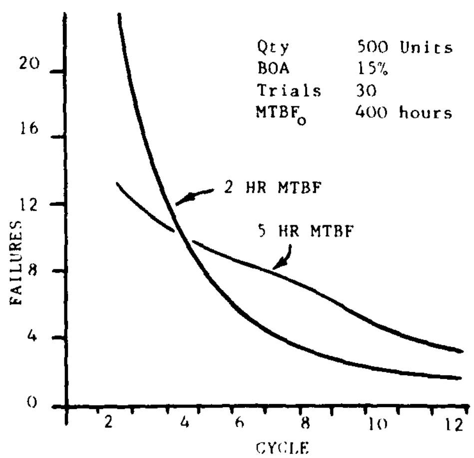


FIGURE 12

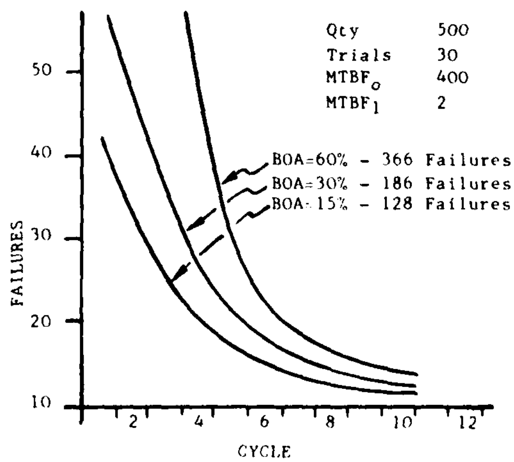


FIGURE 13

(b) Effects of Varying Bad-On-Arrival (BOA) - Figure 13 shows the effects of varying BOA while holding above noted assumption to the same values except that early MTBF is 2 hours. As might be expected, BOA largely affected the frequency of early failures. Table I summarizes the more important data. Again, the average number of cycles to complete 30 sequential failure free cycles was about 32. For 500 systems tested, this means about 1000 cycles more than the 15000 for a perfect system. The important factor is that more than 2/3 of the failures were early failures that were detected largely within 15 cycles.

(c) Actual Experience - Recent experience suggests that the above model reasonably portrays reality. The model must, however, be adjusted to reflect the fact that in some cases early MTBF associated with bad-on-arrival does not sharply increase upon "repair". In fact, the data would suggest a high BOA when in fact it is small coupled with repeat offenders. A more important factor, however, is that in long production runs--the plot of failure frequency shifts downward. That is to say--as fixes are found and incorporated in subsequent units subjected to failure free testing--the total number of failures is decreasing. Figure 14 shows a large production run with failure frequency over succeeding lots. Clearly, there is a shift in the early failure rate indicating that the manufacturer took steps to preclude failure of succeeding equipment subjected to PVT. Uncertainty concerning company liability is now certainly better understood. Early on, many assumed that testing could easily cost considerably in time and money. This stemmed from the fact that the total amount of PVT cycles to satisfy "n" sequential failure cycles was considered indeterminant. Therefore, the venture was assumed quite risky. What most found was that except for early systems--delivery was not affected. It should be pointed out that early system delivery while a problem--shifts the problem away from the field and focuses immediate attention to fixing the problem. This is as it should be since field problems are quite expensive and incur pipelines "costs" as well. With a better understanding of PVT there has come more approval by the manufacturers. Later deliveries are no problem. In fact, since PVT filters out the bad actors--returns of failed equipments is dramatically reduced. This has two positive impacts. First, systems delivered will tend to show a reliability more in line with expected total failure rate for the particular use (recall the earlier application of the NASA equation considering temperature, operating time and turn-on/off cycles). Secondly, because the bad actors are filtered out, the pipeline back to the repair facility (probably the manufacturer) is reduced and the workload and subsequent deliveries are maintained.

TABLE I

	Bad-On-Arrival 15%	30%	60%
Total Failures	128	186	366
Average Cycles	31.8	32.2	33.
Units Passed In:			
30 Cycles	384	325	166
30-45 Cycles	95	153	312
46-60 Cycles	18	19	17
61-90 Cycles	3	3	5

NOTE: MTBF = 400 Hours Early MTBF = 2 Hrs
Trials = 30 Cycles Units Under Test = 500

If there is any problem, it lies in two areas. First, the length of testing, i.e. the number of cycles and secondly, the reporting associated with failures in the PVT cycle. Depending on the manufacturer and his contractual obligations--there is a feeling by some that the number of PVT cycles should be reduced to the point at which it settles out--rather than going to 30 trials (or 15 cycles in cases where there is a cold and hot turn-on and off within a cycle, i.e. two trials within one cycle). As for reporting, most contractors agree with Hammer (as previously noted) that failure analysis is performed solely for determining the cause of failure and follow-up corrective action. The problem is that some contracts call for detailed failure analyses. This is appropriate for formal reliability tests where the sample size is small (say three or four systems). In the case of PVT where every system manufactured is tested to cull out darly failure rate prone systems--the value of such reports is questionable. Certainly, no such report is required had the system failed in the field. One approach to reducing the number of reports is to conduct an out of PVT cycle test if a unit has failed prior to an "n" sequential failure free test. This rightly assumes that a failure, if it is going to occur, will most likely occur in the first cycle. Statistically, it reduces the number of failure reports but it adds one more cycle to the overall test program. Such an approach--reduces the number of reports but increases cost and schedule in subtle ways. Because of the expected number of failures--a possible approach is a simple tabular report. It could for example report LRU number, shop replaceable unit number, assembly number, part type and/or number, time, cycle number, types of cycle (e.g. high or low temperature turn-on, or vibration), etc.

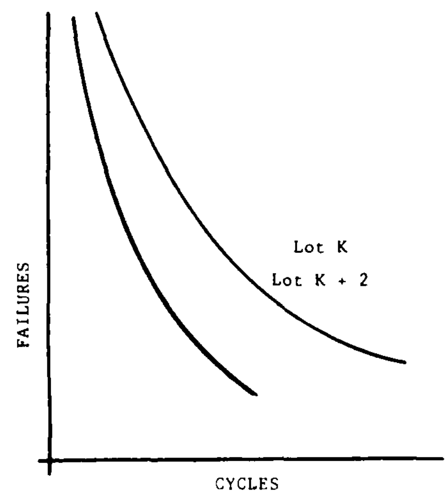


FIGURE 14

6. FINDINGS

First, and foremost, it is clearly evident that PVT can be modelled on a computer. Using random number processes--the model is comprised of two MTBFs. The larger value represents the expected MTBF of the equipment for the test conditions, i.e. frequent turn-on/off, extremes in environment, etc. The smaller value, usually on the order of 2 hours MTBF but coupled to a bad-on-arrival (for example 15% and higher) reasonably correlates with actual test results. A two hour MTBF coupled with the bad-on-arrival rate used for modelling purposes tends to portray the defective parts and workmanship that have been stimulated to fail. The persistance of the low MTBF, given a repair has been effected, appears to depend upon the degree of parts screening and workmanship. If anything--this low MTBF does not appear to grow i.e. reflect MTBF maturation. What does happen is that equipments tested in a PVT environment screen out the defective parts and workmanship problems through repeated turn-on and off in the presence of temperature vibration. It is for this reason that there is such a dramatic improvement in fielded reliability. This certainly reaises the question of effective parts screening at incoming inspection. It would appear that this could be improved, but it is doubtful that it can screen out all marginal parts. PVT is important because it focuses attention on failed parts at system level operation. Most importantly, it does that at the manufacturers' facility where configuration control is possible. Therefore, the manufacturer can effect necessary changes in parts and workmanship since he has the audit trail readily available.

Second, the analysis of the PVT model and examination of contractor data suggests that roughly 15 sequential failure free cycles are necessary to screen out defective parts and workmanship problems. Recall here that cycle refers to one turn-on/off event. For some contractors who currently use one high temperature and one low temperature turn-on (i.e. two distinct events) this would correspond to 7 or 8 cycles. This assumes that the lower value MTBF ($MTBF_1$ about 2 hours) is a reasonable characterization of the problems and that it does not get any better. In fact, the more likely situation is that the turn-on/off cycle coupled with environmental conditions probably shocks the system and are not entirely time dependent. The question then is, are more cycles necessary and for what reasons? The answer is yea. For the moment, we are not absolutely assured that the lower value MTBF is exactly 2 hours. Clearly, it is necessary to conduct sufficient "n" sequential failure free cycles in order to reach a steady state failure frequency. This is necessary in order to screen out the non-random failures. A second and equally important reason is to have data over a sufficient number of cycles to establish a constant failure frequency. This facilitates the estimation of system reliability under test conditions which can be translated to the operational MTBF using the NASA equation. While it is not the intent to use PVT to establish reliability--it is certainly important to have a reasonable estimate of system MTBF.

Third, as noted above, turn-on/off cycling is a key element as is operating time. Therefore, it is important to stimulate these non-random failures as expeditiously as possible. The amount of on-time for any particular cycle depends on the data required. For inertial systems--one hour is usually sufficient to give a reasonable indication of system performance. The amount of off-time depends on the test parameters. Thermal extremes and the time to approach steady state values dictate off-time between cycles. A combined environments chamber incorporating the PVT cycles is an ideal approach. This is not essential. In fact, recent experience with random vibration incorporated at the beginning and/or at the endo of "n" sequential failure free tests where temperature cycling is used has been fairly effective. An effective cycling approach is one hour on--one hour off. This type of cycling given the environment and Eq (4) accelerates random failures frequency and stimulates failure of non-random problems. This, however, does not overstress equipment or reduce its life. If you recall, the model results (which also reasonably protrayed the actual experience)--it took on average 32 cycles to pass a required 30 sequential failure-free test. Even if it took a few hours per cycle, the total accumulated hours would not exceed 100. The environment usually reflects expected vibration and expected temperature extremes.

Fourth, PVT significantly improves the probability that delivered systems meet their design reliability. It does this because all delivered systems are subjected to "n" sequential failure free tests. If it does nothing else, it keeps the undivided attention of manufacturer from beginning to end. Because of this, any failure--particularly with initial deliveries--will be scrutinized very carefully. For example, in a large production of 1000 units a repeat failure for 999 times not only cost cycles but also repairs. This explains the improvement shown in Figure 14.

Fifth, while PVT significantly improves the probability of higher reliability, it is not a substitute for good design practice. Clearly, you cannot test reliability into a poorly designed product. What was particularly fascinating about the analysis and modelling of PVT was that evidently the manufacturers had, in fact, done an excellent job of design. The steady state failure frequency when translated through the NASA equation (i.e. Eq (1)) fairly well corresponded with the design and operational MTBF. The fact that they experienced problems of low MTBFs is largely traceable to defective parts and workmanship (i.e. process control). When these were attacked through PVT--the resulting MTBF of delivered equipment approached expected operational MTBF. In fact, in dealing with reliability, it should be based on the steady state failure frequency as noted earlier. Operational reliability predictions based on all PVT failures is misleading because it includes non-random data. Figure 2 shows that failure free testing resulted in an 87 hours MTBF while actual field MTBF was 540 hours. Similar experience was developed by manufacturers who applied PVT or versions of it prior to shipping equipment.

Sixth, the early failure phenomenon is clearly worth particular attention. The model results as compared to recent manufacturers tests imples an MTBF of a few hours. Anderson (18) suggests that failures that occur in initial cycles represent defects rather than chance failures. Bezat et.al. (19) are of the opinion "...that infanct mortality is in all probability a misnomer...in fact, most solid-state electronics piece parts have such a long life that every true failure within a given population of (actively stressed) parts improves the reliability of the remainder of that population..." This explain why the early low MTBF had to be mechanized as a non-linear function with a bad-on-arrival rate statistical "switch". In fact, it means that attaining design reliability is highly improbable unless screens like PVT are incorporated to cull out defects so that only chance failures are left. Given that we are dealing with defects and not chance events means that we should take another look at reliability estimates and projections based on the Duane (20) reliability growth model

$$\lambda_{\Sigma} = KH^{-\alpha} \qquad (7)$$

The equation imples that reliability is improving as hours are accumulated and that the improvement is a function of the value of alpha. Selby (21) indicates that an alpha of "...of 0.1 can be expected...where no specific consideration is given for reliability...(and) 0.5 for a hard-hitting aggressive reliability program..." The question is--what are appropriate values and what is the basis for them when you consider that many failures are not random in nature and not purely infant mortality?

7. CONCLUSIONS

Production Verification Testing (PVT) can be modelled on a computer. Simulation and comparisons with actual data indicate that for 'N' sequential failure free testing, there is an early and high incidence of failures. This is followed by a decline in failure frequency which approaches a constant failure rate. To replicate this, the model is best represented by two MTBFs. The first is the MTBF expected for the environment. This figure can be estimated using the NASA equation (i.e. Eq (4)). The second MTBF should be on the order of a few hours coupled with bad-on-arrival percentage. This latter approach reasonably models the high incidence of failures early on in failure free testing and reflects the influence of defective parts and poor workmanship. This second and low MTBF can persist, depending on the extent of defects and workmanship problems and give the appearance of high bad-on-arrivals. In any event, from a modelling viewpoint, the MTBF does not improve or mature. About fifteen cycles is needed to establish steady state failure rates. More cycles--on the order of thirty--are suggested to insure that steady values are, in fact, achieved and also to roughly establish system reliability. Modelling of this type is important since it provides interested parties with insight into their results. For those just getting started, it provides a tool for establishing a rough quantitative measure of expected failure, the number of cycles necessary to complete testing and the average number of cycles to complete tests. With early data it is possible to estimate and control problems.

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HARDWARE-SOFTWARE TRADE-OFFS FOR DIGITAL FLIGHT CONTROL OF GUIDED MISSILES

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SUMMARY

This paper compares hardware and software requirements for guided missile flight control systems. The utility and application of three main analog flight control systems are discussed - the open loop system, the rate gyro controlled system, and the accelerometer controlled system. For each of these analog systems, digital requirements are derived and comparisons of the requirements and performance of each system are made. The discussions point out hardware and software trade-off issues that occur in the design process.

PREFACE

Modern homing missiles require flight control systems tailored to accomplish intercept over a prescribed flight envelope at minimum cost. This paper examines flight control system alternatives in order of increasing complexity and discusses the hardware-software trade-offs implied by each alternative. Miss distance is the measure of homing performance, and miss distance is affected by the three main guidance parameters - effective navigation ratio, relative stability, and response time. The flight control system instrumentation configuration is the significant factor in determining how these guidance system parameters vary with knowledge of aerodynamic flight condition - altitude, Mach number, angle of attack and wind angle. The software requirements, which vary for each flight control system, are significant factors in digital computer architecture, sizing, and cost.

The flight control system utilizes an autopilot that converts guidance system commands into commands to the control surface actuators (Ref. 1). The actuators in turn produce deflections of the aerodynamic control surfaces that cause the missile to maneuver. The resultant dynamics may be measured by rate gyros and accelerometers to form a feedback control system that stabilizes the airframe and produces the fast flight control transient response required for accurate homing.

Three types of analog control systems are presented and analyzed for their performance properties when used in guided missiles. The general requirements for digitizing these systems are derived and some pertinent trade-offs discussed to point the way to developing a minimum cost system to meet a given target threat.

1. INTRODUCTION

The incorporation of tracking seekers into anti-aircraft missiles created a whole new set of flight control problems. Creative designs solving these problems emerged from all of the engineering disciplines. Aerodynamicists tailored the flight vehicle and its control surface size and locations to meet controllability requirements. Servomechanism designers developed control surface actuators to meet new performance requirements. Parasitic vehicle control problems that interfered with homing system performance emerged. The small miss distances required magnified the importance of guidance component imperfections and dictated relatively stringent control of the flight control system dynamics - acceleration gain, time constant, and relative stability.

Homing missiles use a seeker to track the target, a noise filter to separate the relative target motion from the noise, a guidance law to generate the appropriate missile acceleration commands to intercept the target, and a flight control system to achieve the desired acceleration response to the guidance commands. (Figure 1) (Ref. 1, 2, 3). This paper concentrates on the flight control system and its interaction with the missile's homing performance. The flight control system utilizes an autopilot that converts guidance system commands into commands to the control surface actuators (Ref. 4). The actuators in turn produce deflections of the aerodynamic control surfaces that cause the missile to maneuver. The resultant dynamics may be measured by rate gyros and accelerometers to form a feedback control sytsem that stabilizes the airframe and produces the fast flight control transient response required for accurate homing.

Historically, at Raytheon, radar guided missile development started with the design of a seeker that was tested in an available missile airframe, called the Lark. (Ref. 5) Its dramatic success led to the development of another seeker for the Navy's Sparrow missile which in turn, led to a completely new flight control system. The relatively undamped airframe required a rate gyro for damping, and the proportional navigation guidance law required lateral acceleration control which was achieved with missile body mounted accelerometers that developed an error command between the desired and achieved acceleration. A low pass filter was needed on the acceleration error to attenuate high frequency effects sensed by the accelerometer, such as structural vibrations. As technology improved, the time constant of this filter became longer until it approached an ideal integrator. Such a flight control system performed well during the glide phase of flight when the missile airframe was aerodynamically stable. However, to reduce minimum

*The work was performed partially under the Office of Naval Research Contract N00014-75-C-0549 ONR Task 233-052.

range, it was necessary to home during the boost phase of flight, where the missile airframe may be aerodynamically unstable. This was simply achieved by integrating the rate gyro to generate a signal proportional to angle-of-attack and using this signal to electronically add stability, called synthetic stability, to the missile. It is a tribute to the genius of its designers that this three loop type of flight control system is used in all operational radar guided homing missiles today, even though the targets have changed dramatically and the required zone of performance has increased enormously.

Until recent times these autopilot/flight control systems have been implemented using analog circuitry. However, with the advances in digital computer electronics (Ref. 6-10), modern missiles are becoming digitally controlled. Digital control has many advantages. The first is low cost because many electronic functions are time shared and because the modern mass production manufacturing technology for digital computers is well developed. A second advantage is adaptability because the computer can be programmed to make logical decisions and because these programs can be changed if desired. A third advantage is the capability for built in self test without adding substantially to the amount of electronics on board the missile. A fourth advantage is the large dynamic range possible because dynamic range can be increased by adding bits to the word length. A fifth advantage is long term stability - digital circuitry does not have the drifts of analog circuitry. Finally digital circuits are more reliable than analog circuits.

Of course, the advantages of digital control bring certain problems (Ref. 11). Special analog-digital and digital-analog interfaces must be prepared. Sampling data rates must be high enough to avoid foldover of high frequency phenomena into the low frequency control bandwidth, and computer word lengths must be compatible with these data rates. Last, but not least, software must be organized, programmed, and debugged. All of these problems have been solved for the designs currently developed at Raytheon.

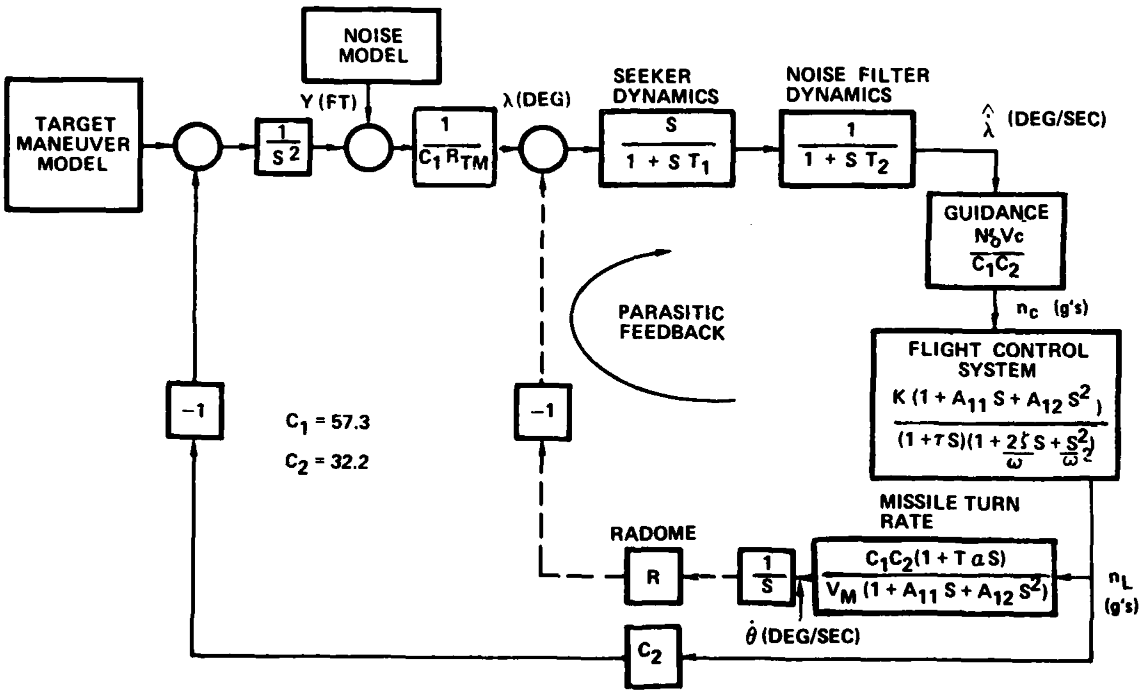

Figure 1. Linear Miss Distance Dynamics

2. FLIGHT CONTROL SYSTEM - GUIDANCE SYSTEM INTERACTIONS

The guidance system must get the missile close enough to kill the target when the warhead explodes. Therefore an appropriate measure of guidance performance is miss distance, which is defined as the minimum distance between the missile and target. Two important causes of miss distance in a radar guided proportional navigation system are random target maneuver and target glint noise. The missile-target engagement model of Figure 1 is used to obtain quantitative comparisons of miss distance as a function of the significant guidance system parameters. In turn, this information is used to evaluate the effects of flight control system parameters on missile performance.

The flight control system is represented by the following transfer function

$$\frac{n_L}{n_C} = \frac{K(1+A_{11}S+A_{12}S^2)}{(1+\tau S)(1+\frac{2\zeta S}{\omega}+\frac{S^2}{\omega^2})} \tag{1}$$

The values for the zeroes are determined by the aerodynamic configuration. The values for the control system gain (K), the time constant (τ), the damping ratio (ζ), and the natural frequency (ω) are determined by the flight control system gains and parameters. Neglecting parasitic effects, the guidance system transfer function from line-of-sight rate to missile acceleration ($n_L/\dot{\lambda}$) (Ref. 4, 12, 13) is

$$\left.\frac{n_L}{\dot{\lambda}}\right|_{\substack{\text{PERFECT}\\ \text{RADOME}}} = \frac{KN'_oV_c(1+A_{11}S+A_{12}S^2)}{(1+T_1S)(1+T_2S)(1+\tau S)(1+\frac{2\zeta}{\omega}S+\frac{S^2}{\omega^2})} \tag{2}$$

where N' is the desired effective navigation ratio ($N' = KN'_o$) V_c is the closing velocity and T_1 and T_2 are the seeker and noise filter time constants.

The actual effective navigation ratio is one major determinant of homing performance. Typical performance results displayed in Figure 2 show that if the effective navigation ratio is too high, miss distance increases due to noise, and if it is too low, miss distance increases due to target maneuver. Therefore the effective navigation ratio has an optimal value and in no case should it exceed certain bounds. A variation of the flight control system gain K from unity is equivalent to an effective navigation ratio change and optimal homing performance cannot be attained.

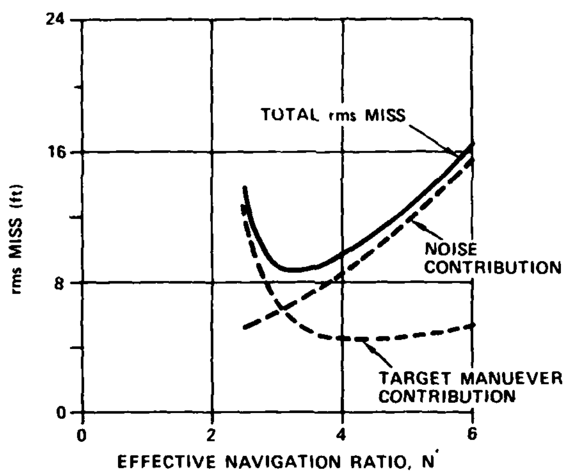

Figure 2. Effective Navigation Ratio Influences System Performance

Another major determinant of homing performance is the guidance system time constant. With parasitic effects neglected, the guidance system time constant T_Σ, can be approximated as the coefficient of the first order term in the denominator of Eq. (2) or

$$T_\Sigma \approx T_1 + T_2 + \tau + \frac{2\zeta}{\omega} \tag{3}$$

Therefore increasing the flight control system time constant, τ, increases T_Σ while decreasing that time constant decreases the guidance time constant. Figure 3 shows that decreasing the flight control system time constant increases the miss distance contribution due to noise and decreases the miss distance due to target maneuver, while increasing the time constant does just the opposite. For the set of inputs considered it appears from Figure 3 that the flight control system time constant can be made arbitrarily small, but this cannot be done because of radome refraction effects. When the parasitic radome loop is considered the guidance transfer function becomes

$$\left.\frac{n_L}{\dot{\lambda}}\right|_{\substack{\text{IMPERFECT}\\ \text{RADOME}}} = \frac{K N'_oV_c(1+A_{11}S+A_{12}S^2)}{(1+ST_1)(1+ST_2)(1+S\tau)(1+\frac{2\zeta}{\omega}S+\frac{S^2}{\omega^2})+\frac{KN'_oV_cR}{V_M}(1+T_\alpha S)} \tag{4}$$

where R is the radome slope and T_α is the missile turning rate time constant. At high altitudes, where T_α is large, the guidance transfer function can become unstable due to either excessive positive or negative radome slopes. Thus a small flight control system time constant yields a large guidance system sensitivity to radome slopes as shown in Figure 4. Therefore the optimal value of the flight control system time constant is bounded on the high side by target maneuver considerations and on the low side by radome refraction effects.

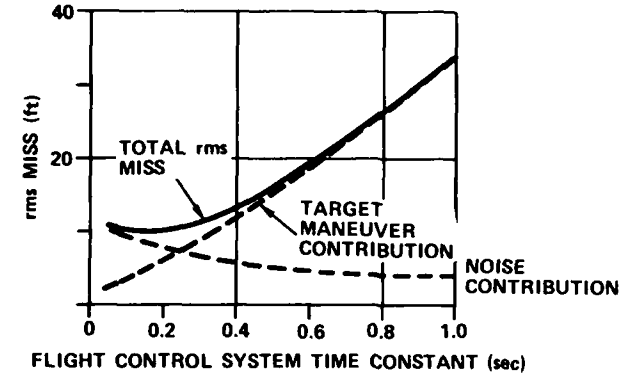

Figure 3. Flight Control System Time Constant Influences System Performance

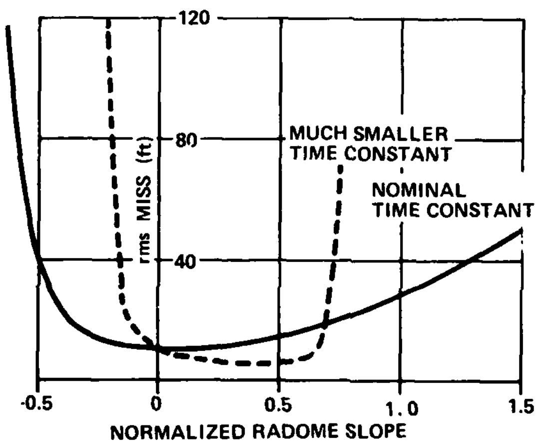

Figure 4. Decreasing Flight Control System Time Constant Makes System Performance More Sensitive to Radome Slope

Flight control system damping becomes more important when radome effects are considered. Figure 5 shows typical high altitude performance for a flight control system with high and low damping. The system with low damping is far more sensitive to radome effects than the system with higher damping.

The flight control system design influences homing performance primarily through its low frequency gain, its damping, and its time constant. The low frequency gain variations modify the effective navigation ratio which is important in all homing applications. The damping affects the homing stability, which is primarily important in radar homing applications. The time constant effects the homing dynamic response, which is important in all homing applications.

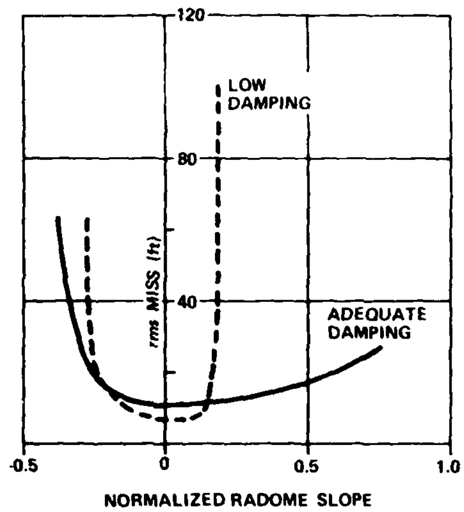

Figure 5. Decreasing Flight Control System Damping Makes System Performance More Sensitive to Radome Slope

3. ANALOG FLIGHT CONTROL SYSTEMS

Analog flight control systems require standard analog specifications of gain, bandwidth, dynamic range and noise susceptibility. The following discussion develops the requirements and properties of analog flight controls for open loop, rate gyro controlled and accelerometer controlled systems for use in homing guidance.

Open Loop Flight Control

The open loop flight control system, shown in Figure 6 is simple and requires no instrumentation. This system modifies the guidance command, n_C, by a gain, K_{OL}, to achieve unity acceleration gain (n_L/n_C) of the flight control system. With the airframe transfer function given by

$$\frac{n_L}{\delta} = \frac{K_1(1+A_{11}S+A_{12}S^2)}{1+B_{11}S+B_{12}S^2} \tag{5}$$

and actuator dynamics neglected, the control system transfer function can be written as

$$\frac{n_L}{n_C} = \frac{+K_{OL}K_1(1+A_{11}S+A_{12}S^2)}{1+B_{11}S+B_{12}S^2} \tag{6}$$

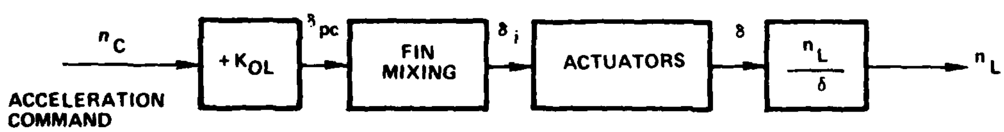

Figure 6. Open Loop Flight Control System

Except for the electronic gain K_{OL}, the flight control system transfer function is that of the bare airframe. Since a typical homing missile has little aerodynamic damping, the system transfer function will be very underdamped. If an open loop flight control system is used in a radar homing system, the low damping will cause instability through the parasitic feedback caused by radome refraction slope (see previous section and Ref. 13, 14). However, the open loop system could be used in a system that does not have appreciable radome refraction slopes such as an infrared system, provided that the actuator bandwidth is low enough to attenuate the mechanical structural vibrations of the airframe.

Since the system transfer function is that of the airframe, the airframe must be stable to obtain adequate homing performance. Therefore the airframe requirement for this type of flight control system is that the center of gravity never moves aft of the center of pressure.

Unity acceleration gain can be achieved by making the flight control system gain the inverse of the aerodynamic gain K_1, i.e.:

$$K_{OL} = \frac{+1}{K_1} \tag{7}$$

Since the airframe gain, K_1, varies with flight condition, so will the control system gain as shown in Figure 7. The airframe gain variations can be compensated to the accuracy that the aerodynamic data are known. Inaccurate compensation will degrade homing performance because the proper effective navigation ratio, N', will not be obtained. Therefore missiles which use this simple control system require accurately defined aerodynamic characteristics. An actual history is described in Ref. 15 where extensive full scale wind tunnel testing was needed to characterize the aerodynamic gain accurately enough for satisfactory navigation ratio control.

Rate Gyro Flight Control

The rate gyro flight control system uses a rate gyro connected in a rate command system as shown in Figure 8. With pitch rate transfer function dynamics given by

$$\frac{\dot{\theta}}{\delta} = \frac{K_3(1+T_\alpha S)}{1+B_{11}S+B_{12}S^2} \tag{8}$$

and actuator and rate gyro dynamics neglected, the control system transfer function can be expressed as

$$\frac{n_L}{n_C} = \frac{+KK_RK_1}{1-K_RK_3}\left[\frac{1+A_{11}S+A_{12}S^2}{1+S\left(\frac{B_{11}-K_RK_3T_\alpha}{1-K_RK_3}\right)+\frac{B_{12}S^2}{(1-K_RK_3)}}\right] \tag{9}$$

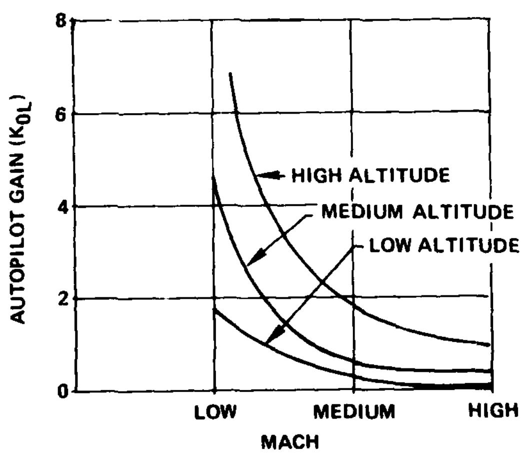

Figure 7. Open Loop Autopilot Gain Varies Widely with Altitude and Mach Number Changes

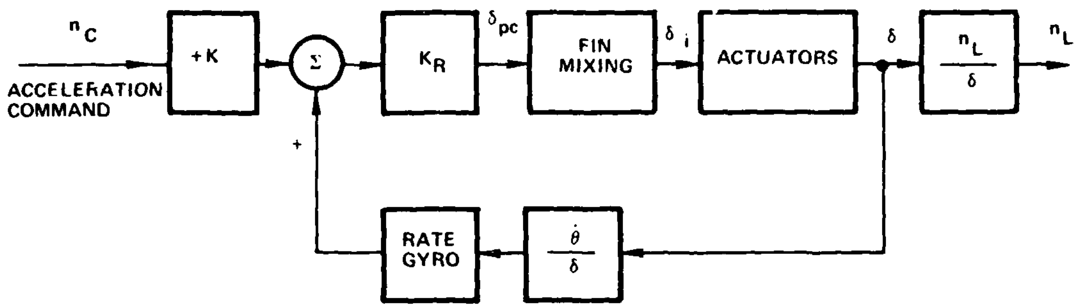

Figure 8. Rate Gyro Flight Control System

The flight control system gain, K, provides unity transmission when set to

$$K = \frac{+(1-K_R K_3)}{K_R K_1} \qquad (10)$$

For the usual case where the loop gain $|K_R K_3|$ is less than unity, this flight control gain (K) has the same variation as the open loop gain but it is magnified by the factor $1/K_R$. Since K_R is usually less than one, this system is especially sensitive to altitude and Mach variations. In addition, any noise on the command is magnified by the high gain. This makes seeker instrument noise requirements more stringent to keep the command noise low. A large dynamic range in the actuator electronics is required in order to avoid noise saturation. Figure 9 shows a typical variation in autopilot gain versus Mach number and altitude. Note the large values of gain required.

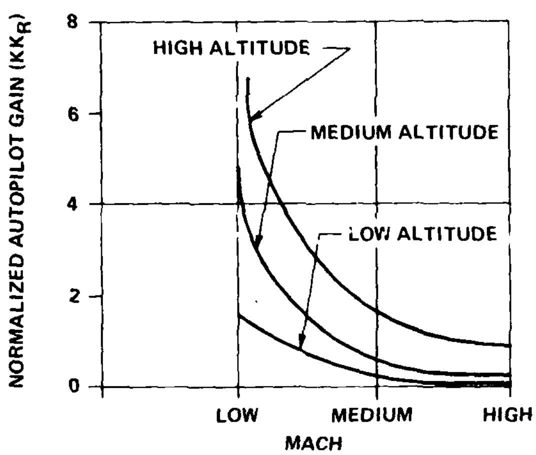

Figure 9. Rate Gyro Autopilot Gain Varies Widely with Altitude and Mach Number Variations

The rate loop gain, K_R, is adjusted to increase the low damping of the airframe, as can be seen from Eq. (9) to a more acceptable value for radar homing. The dynamic response of this system is essentially that of a quadratic transfer function with good damping and a resonant frequency slightly higher than the natural frequency of the airframe. Typically this frequency is high at low altitude, high Mach and decreases as altitude and/or Mach decreases. Therefore, the response time is short, but variable with flight condition.

The open loop transfer function (HG) can be obtained from Figure 8 and Eq. (8) and is

$$HG = \frac{-K_R K_3(1+T_\alpha S)}{(1+B_{11}S+B_{12}S^2)} \qquad (11)$$

For cross-over frequencies greater than the airframe roots $\frac{1}{T_\alpha}$ and $\sqrt{\frac{1}{B_{12}}}$ an approximate expression for the crossover frequency of the asymptotic Bode diagram, ω_{CR}, is

$$\omega_{CR} \approx \frac{-K_R K_3 T_\alpha}{B_{12}} \tag{12}$$

Thus, the gain, K_R, not only determines the system damping but also the system cross-over frequency. For a design with adequate stability margins the cross-over frequency should be much lower than the lowest bandwidth of the actuator or rate gyro bandwidths.

The rate gyro flight control system has good damping, but its acceleration gain is more dependent upon both altitude and velocity than the open loop system. Its time constant is short, but dependent on aerodynamic parameters of altitude and Mach number.

Long range homing takes place after the missile fuel has been expended and the missile center of gravity is forward of its center of aerodynamic pressure. This is a stable configuration and is evidenced by a positive value for B_{12} in the aerodynamic transfer functions or similarly by a negative moment due to angle-of-attack. However, for short range intercepts before the missile fuel has been expended, the missile center of gravity may be aft of the center of pressure. This is an unstable configuration and is evidenced by a negative value for B_{12} or a positive moment due to angle-of-attack change. Stability can be regained by integrating the rate gyro to obtain a signal proportional to angle-of-attack over short intervals. This signal drives the rate loop and supplies an electronic, or "synthetic," stability to the inherently unstable airframe. The block diagram is shown in Figure 10. The closed loop transfer function, neglecting actuator and rate gyro dynamics can be expressed as the following cubic.

$$\frac{n_L}{n_C} = \frac{K_S K_1}{K_3}\left[\frac{1+A_{11}S+A_{12}S^2}{1+S\left(\frac{1-K_R K_3-K_R\omega_I K_3 T_\alpha}{-K_R\omega_I K_3}\right)+S^2\left(\frac{B_{11}-K_R K_3 T_\alpha}{-K_R\omega_I K_3}\right)+\frac{B_{12}S^3}{-K_R\omega_I K_3}}\right] \tag{13}$$

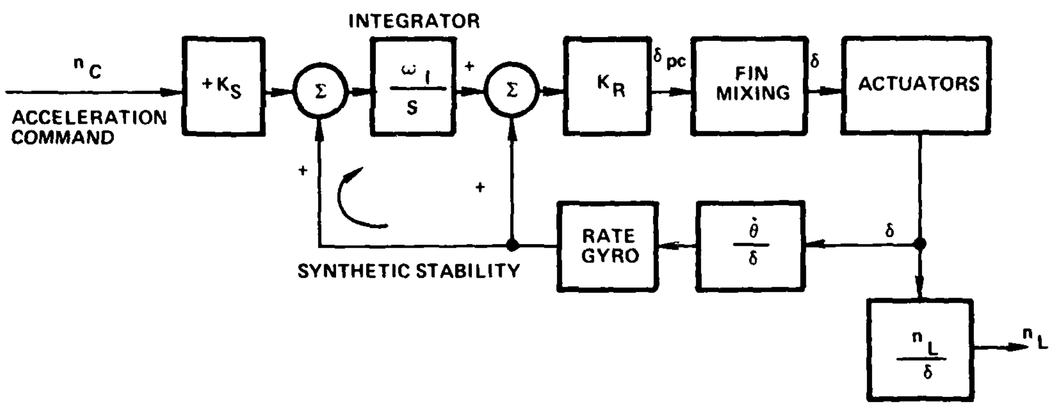

Figure 10. Integrated Rate Gyro Flight Control System

For unity transmission, the autopilot gain, K_S, is proportional to the ratio of the airframe rate and acceleration gains or

$$K_S = \frac{K_3}{K_1} \tag{14}$$

which is essentially independent of altitude and inversely proportional to velocity as shown in Figure 11. Therefore the effective navigation ratio can be maintained over a wide altitude band even though the aerodynamic data are not well known. More detailed discussion of the properties of this system are given in Ref. 18.

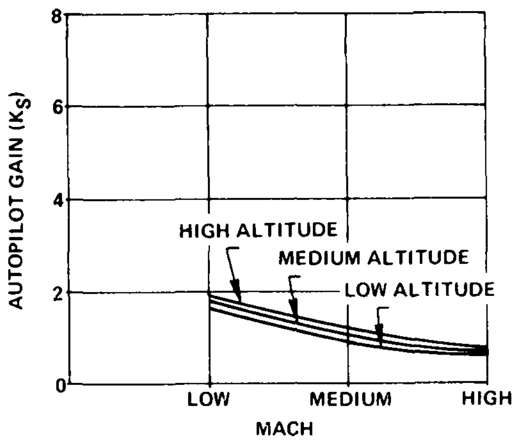

Figure 11. Integrated Rate Gyro Gain is Independent of Altitude

Accelerometer Flight Control

When an accelerometer is added to the missile and connected to control the system by the error between the acceleration command and the actual acceleration, the three loop flight control system shown in Figure 12 results. This system achieves dc gain control independent of altitude and Mach, and fast response time for both stable and unstable missiles. The transfer function from fin position to acceleration at the accelerometer location is given by

$$\frac{n_A}{\delta} = \frac{K_2(1+A_{21}S+A_{22}S^2)}{1+B_{11}S+B_{12}S^2} \tag{15}$$

Note that the zeroes of n_A are slightly different than those of n_L because the accelerometer is not at the missile center of gravity. Neglecting the rate gyro, actuator and accelerometer dynamics, the control system transfer function can be written as

$$\frac{n_L}{n_C} = \frac{K_0 K_A K_1}{K_3+K_A K_2}\left[\frac{1+A_{11}S+A_{12}S^2}{1+D_1 S+D_2 S^2+D_3 S^3}\right] \tag{16}$$

where

$$D_1 = \frac{1-K_R K_3-K_R K_3 \omega_I T_a-K_R \omega_I K_A K_2 A_{21}}{-K_R K_3 \omega_I-K_R \omega_I K_A K_2} \tag{17}$$

$$D_2 = \frac{B_{11}-K_R K_3 T_a-K_R \omega_I K_A K_2 A_{22}}{-K_R K_3 \omega_I-K_R \omega_I K_A K_2} \tag{18}$$

$$D_3 = \frac{B_{12}}{-K_R K_3 \omega_I-K_R \omega_I K_A K_2} \tag{19}$$

The control system gain, K_0, provides unity transmission when set to

$$K_0 = \left(\frac{K_3+K_A K_2}{K_A K_1}\right) = \left(1+\frac{1845}{K_A V_M}\right) \tag{20}$$

Generally $\frac{1845}{K_A V_M}$ is small compared to unity and so the autopilot gain, K_0, is independent of both altitude and Mach number as shown in Figure 13. In other words, the gain of this system is quite robust to changes in altitude and Mach number, or to error in knowledge of the aerodynamic coefficients of the missile.

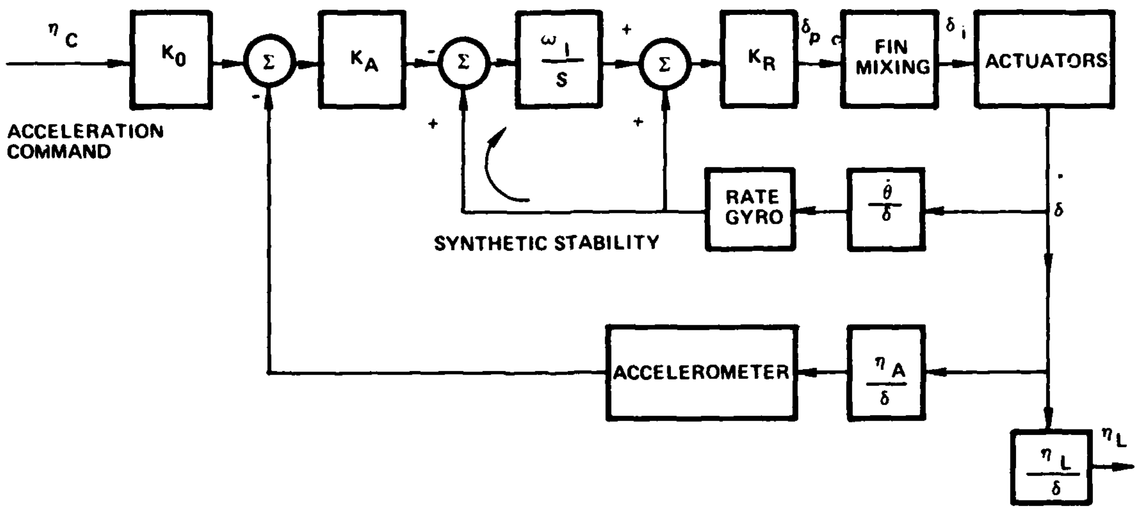

Figure 12. Acceleration Flight Control System

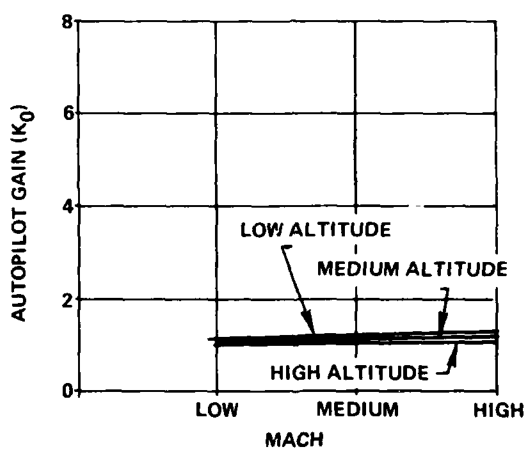

Figure 13. Accelerometer Autopilot Gain is Independent of Altitude and Mach Number Changes

As can be seen from Eq. (16) the dynamic response of the accelerometer flight control system is a cubic, like the integrated rate gyro system. However, the additional feedback loop makes another gain available so that specified values of time constant, damping, and cross-over frequency can be obtained for both stable and unstable airframes by a proper combination of the three gains. The time constant of this system is not limited to values greater than the missile's turning rate time constant. Therefore, to first order, the gain K_R determines the cross-over frequency, ω_I determines the damping and K_A determines the time constant. Therefore, the time response of the missile can be reduced to the values needed to intercept high performance aircraft that may maneuver drastically in an effort to escape interception.

Summary of Analog Control Systems

The open loop flight control system damping ratio and time constant are that of the airframe and are therefore independent of the autopilot gain, K_{OL}. The low damping of this flight control system makes it unacceptable for radar homing applications due to parasitic attitude loop instabilities.

Some rate gyro flight control system characteristics are determined by the autopilot gain. Increasing the rate gyro gain, K_R, increases both the damping and crossover frequency of that system. The maximum damping achievable is determined by the maximum allowable crossover frequency. The time constant of this system is that of the airframe and is therefore independent of autopilot gain.

The accelerometer flight control system characteristics are controlled more completely by the autopilot gains. The rate loop gain, K_R, determines the crossover frequency, the synthetic stability gain, ω_I, determines the damping while the accelerometer gain, K_A, determines the time constant. This system gives the designer most flexibility in setting the above guidance parameters independentl· to obtain optimal system performance.

4. DIGITAL FLIGHT CONTROL

General

Digital autopilots require specifications on a) the system data rate, b) the number of words, c) the least significant bit, d) the word length, e) the gain coefficient data rate, f) the transport lag, g) the throughput, and h) the storage. (Ref. 16, 17) These requirements are needed to specify the computer architecture as well as its dynamic properties, such as speed and memory capability.

This section of the paper will develop the general expressions needed to obtain these digital specifications from the analog and digital properties desired. Then appropriate numerical specifications will be given for the open loop, rate gyro controlled and accelerometer controlled systems. Comparative figures for performance requirements are generated by dividing the altitude and mach number performance envelope into three nested zones of progressively increasing size. Zone 1 encompasses a low altitude and low mach number zone. Zone 2 includes zone 1 and extends the altitude and mach numbers to higher, but medium ranges. Zone 3 includes zone 2 and extends the altitude and mach numbers to higher values. The software specifications for each zone are developed and tabulated in the remainder of the paper.

a) System Data Rate

The system data rate is the sampling rate in the signal dynamic path. Signal sampling must be fast enough to minimize aliasing or foldover of the frequency response in the region of interest. This is the Nyquist rate which is found by doubling the frequency where the amplitude is 20 dB down or more. For a first order system with a break frequency of f_o, the sampling frequency is then 20 f_o.

b) Number of Words

The number of words required to represent a zone is determined by the dynamic range of the gains required over that zone and the allowable change in gain from one zone to another as given by Eq. 21. (Ref. 11)

$$N_W = 1 + \frac{\log_2 R_d}{\log_2 r} \qquad (21)$$

where

N_W is the number of words required

R_d is the dynamic range of the gains, i.e. $R_d = K_{max}/K_{min}$

r is the allowable ratio between adjacent gains = $K_i/K_{i-1} > 1$

If a 10 percent change in gain is allowed by system performance, r = 1.1/1 = 1.1.

c) Least Significant Bit

The least significant bit determines the accuracy of the control gain. It is set from the smallest gain needed. In missile control, the product of the control gain, K_o, and the aerodynamic gain, K_1, is the desired result and guidance places an accuracy on this product, P, rather than on either gain individually. Therefore, more control gain error is allowed if the aerodynamic gain is well known. For instance,

$$K_o K_1 = P \qquad (22)$$

Therefore

$$\frac{\Delta K_o}{K_o} + \frac{\Delta K_1}{K_1} = \frac{\Delta P}{P} \qquad (23)$$

where

$\Delta P/P$ = the total acceleration control uncertainty

$\Delta K_o/K_o$ = the control gain uncertainty

$\Delta K_1/K_1$ = the aerodynamic gain uncertainty

For purposes of the paper, allow 10 percent control gain and 10 percent aerodynamic gain uncertainty to yield 20 percent uncertainty in total acceleration control. Then the least significant bit (LSB) is the minimum value of ΔK_o. It is found from the allowable percentage error at $K_{o_{min}}$.

$$L_{SB} = p\ K_{o\ min} \tag{24}$$

where

$$p = (\Delta K_o/K_o)_{allowed}$$

d) Word Length

The required word length (Ref. 11) is determined from the maximum control gain and the least significant bit as

$$W_L = \log_2 \frac{K_{max}}{LSB} \text{ bits} + 1 \text{ sign bit} \tag{25}$$

where

W_L is the word length in bits

e) Gain Coefficient Data Rate

Data rate involves two loops, a signal loop and a gain control loop, as shown in Figure 14.

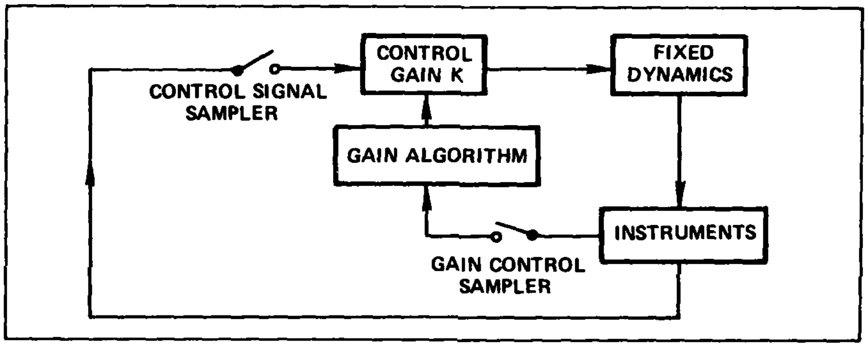

Figure 14. Signal and Gain Control Samplers

The sampling rate of the signal loop was previously discussed. The sampling rate of the gain control loop must be fast enough to faithfully represent the required gain changes as the environment changes with time.

The factors that determine the gain path sampling rate are derived in this section. A control gain versus time plot is shown in Figure 15.

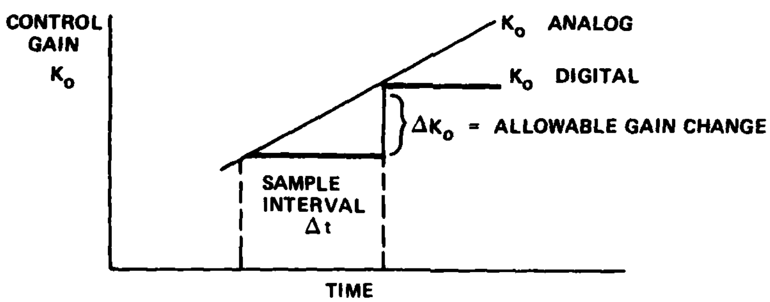

Figure 15. Time Variation of Control Gain

If the sample interval Δt is short enough, the control gain increment is

$$\Delta K_o = \dot{K}_o\ \Delta t \tag{26}$$

Therefore Eq. (26) cont.

$$\Delta t = \frac{K_o}{\dot{K}_o}$$

If a fractional change (p) in K_o is allowed,

$$\Delta K_o = p\, K_o \tag{27}$$

then

$$\Delta t = \frac{pK_o}{\dot{K}_o} \tag{28}$$

For K_o only a function of altitude and speed, the rate of change of the control gain is

$$\dot{K}_o = \frac{\partial K_o}{\partial V}\dot{V} + \frac{\partial K_o}{\partial h}\dot{h} \tag{29}$$

where

V is the missile velocity

h is the altitude

Under worst-worst case assumptions $\dot{V}$ becomes the maximum acceleration of the missile and $\dot{h}$ becomes the maximum velocity of the missile.

Then

$$\left(\dot{K}_o\right)_{max} = \left(\frac{\partial K}{\partial V}\right)_{max} a_{max} + \left(\frac{\partial K}{\partial h}\right)_{max} V_{max} \tag{30}$$

Thus, the minimum sampling interval for any zone is

$$(\Delta t)_{min} = p\, \frac{(K_o)_{min}}{(\dot{K}_o)_{max}} \tag{31}$$

The sampling interval obtained in this way is the smallest necessary because the analysis uses worst-worst case assumptions. When the design reaches the simulation stage, an increase of this sampling interval may by shown to yield adequate system performance under more realistic conditions than the worst-worst case used in the derivation. In addition, the percentage change, p, allowed in K_o, is a direct factor in the sampling, i.e., a 40 percent change will allow twice the sampling interval that a 20 percent change will allow. Clearly, the gain change sampling interval for each gain in the system will require careful examination by the design engineer to arrive at a consistent set of rates for all the system gains.

f) Transport Lag

Transport lag is the dead time between the time when control measurements are taken and the time when control commands exit the computer. For low bandwidth control loops, transport lag may not be a factor. However as performance requirements become more stringent careful attention must be paid to transport lag to insure control stability. Computations must be divided between a critical path consisting of those computations that can be made only after the measurements are taken and a noncritical path consisting of computations that can be performed based on past data. For a crossover frequency of 80 rad/sec a time delay of 1 msec will give a phase lag, θ, of

$$\theta = \omega_{CR}\tau_D = \frac{80}{1000} = 80 \text{ mrad or } 4.6° \tag{32}$$

Clearly the transport lag is a significant factor in the system phase margin and stability.

g) Throughput

The throughput required is determined by the number of operations (adds, multiplies, data transfers, etc.) required per second. An estimate of the throughput required can be obtained prior to actually writing the computer program from count of arithmetic operations. Many times this count is reduced to an equivalent number of adds per second. After programming, it is only necessary to count the operations and average over the appropriate time.

h) Storage Required

The storage required for the control gains depends upon whether all the gains are stored and retrieved or whether a few are stored and the exact gain obtained from interpolation. Interpolation routines trade throughput for storage and the proper choice of any system depends upon the hardware that can be made available for that system. For the paper, assume all the gains are stored. Then the amount of storage required is

$$S_R = W_L \times N_W \times N_G \tag{33}$$

where

N_G = number of gains stored.

W_L = word length in bits.

N_W = number of words.

S_R = amount of storage required in bits.

Open Loop Control

The set of gains, K_{OL}, used for the open loop calculations at the zone boundaries are listed in Table 1. (Ref. 18).

Table 1

Open Loop Control Gains

Altitude \ Speed	Low A (Zone 1)	Low B (Zone 1)	Med C (Zone 2)	High D (Zone 3)
Low A	1.77	0.198	0.117	0.0775
Low B	2.57	0.287	0.170	0.112
Med C	4.70	0.521	0.307	0.204
High D	15.24	1.68	1.00	0.662

The dotted lines show the division into zones that are used for comparisons of specifications as the range of flight conditions expands.

a) The System Data Rate

The allowable system data rate depends fundamentally on the frequency of aerodynamic short period mode (f_d) and on the structural resonance frequency. For an open loop control application, the structural resonance is usually attenuated enough by the actuator so only the aerodynamic frequency is significant. A sampling frequency of 20 f_d is sufficiently larger than the Nyquist frequency to achieve a control bandwidth consistent with the missile aerodynamics. For each zone, the aerodynamic frequency, the sampling frequency, and the time between samples is shown in Table 2.

Table 2

Open Loop Sampling Requirements

Zone \ Sampling Requirement	Highest Aerodynamic Frequency f_d(Hz)	Sampling Frequency f_s(Hz)	Sampling Interval T_s (millisec)
Zone 1	2.81	56.3	17.75
Zone 2	4.09	81.8	12.2
Zone 3	5.34	106.9	9.35

Note that the sampling interval decreases as the zone coverage gets to higher speeds.

b) Number of Words

From Eq. (21) and the open loop gains of Table 1

Zone 1 $R_D = \dfrac{K_{OL\ max}}{K_{OL\ min}} = 2.57/0.198 = 12.98$

$r = 1.10$ for a 10 percent level change

$$N_W = 1 + \frac{\log_2 R_d}{\log_2 r} = 1 + \frac{\log_2 12.98}{\log_2 1.10} = 1 + 26.9 = 27.9,$$

$N_W = 28$ words

Zone 2 $R_d = 4.70/0.117 = 40.1$, $N_W \rightarrow 40$ words

Zone 3 $R_d = 15.2/0.0775 = 196$, $N_W \rightarrow 57$ words

Note that the number of words needed to represent the gains increases dramatically as the zone coverage expands.

c) Least Significant Bit

The open loop control gain only affects the effective navigation ratio. Miss distance data previously presented shows that N' can vary by ±20 percent. If we assume 10 percent uncertainty in the fixed aerodynamic gain, then the control gain can have a 10 percent error. Then from Eq. (24) LSB = $p\ K_{OLmin} = 0.1\ K_{OL\ min}$ for each zone

	Zone 1	Zone 2	Zone 3
LSB	0.0198	0.0117	0.00775

d) Word Length

For each zone the word length can be found from Eq. (25). The open loop word length requirements are calculated using the information from this formula and tables and LSB.

Zone 1 $W_L = \log_2 (2.57/0.0198) + 1 = 8$ bits

Zone 2 $W_L = \log_2 (2.69/0.0117) + 1 = 10$ bits

Zone 3 $W_L = \log_2 (15.2/0.00775) + 1 = 12$ bits

As the zone increases in coverage the word length must be increased.

e) Gain Coefficient Data Rate

For an allowable control gain change of 10 percent, the minimum sampling interval is independent of zone for the cases considered here.

$(\Delta t)_{min} = 19.2$ msec for all zones

f) Transport Lag

The homing loop is the only closed loop in the open loop control system. Bandwidth and crossover frequencies are not a meaningful measure of the homing loop because that loop dynamics varies with range from the missile to target. A rule of thumb in this case is that the transport lag should be less than one third of the sampling interval.

	Zone 1	Zone 2	Zone 3
Transport Lag (msec)	5.92	4.07	3.12

g) Throughput

The open loop control is a simple multiply in each axis. For comparative purposes, we only consider a single axis and neglect instructions like fetch from store. Then we get one multiply required per sample interval. If we say a multiply is the equivalent of 10 additions and we define an addition as an operation we get

$T_p = 10/T_S$ operations/sec per axis

and the throughput requirements can be calculated using the information from Table 2.

Zone 1 T_p = 10/0.01775 = 563 ops/sec/axis

Zone 2 T_p = 10/0.0122 = 820 ops/sec/axis

Zone 3 T_p = 10/0.00935 = 1070 ops/sec/axis

Note the expansion of throughput required as the aerodynamic zone expands.

h) Storage

For a system where all the gains are stored and there is 1 gain per axis ($N_G = 1$), $S_R = N_W$ in words or $S_R = N_W \times W_L$ in bits Eq. (33). Then for each zone, the storage required is obtained from Sections b) and d) and is

Zone 1 S_R = 28 words x 8 bits/word = 280 bits

Zone 2 S_R = 40 words x 10 bits/word = 400 bits

Zone 3 S_R = 57 words x 12 bits/word = 570 bits

The number of bits required can be calculated using the word length specifications as above or a fixed number of bits per word if the machine is already chosen.

Summary of Open Loop Digital Control Specifications

The open loop digital control specifications are summarized in Table 3.

Table 3

Summary of Open Loop Digital Control Specifications

Parameter	Zone 1	Zone 2	Zone 3
a) System Data Interval (msec)	17.75	12.2	9.35
b) Number of Words (words/axis)	28	40	57
c) *Least Significant Bit*	0.0198	0.0117	0.00775
d) Word Length (Bits)	8	10	12
e) Gain Coefficient Data Interval (msec)	19.2	19.2	19.2
f) Transport Lag (msec)	5.9	4.1	3.1
g) Throughput (ops/sec/axis)	563	820	1070
h) Storage (bits/axis)	280	400	570

Rate Gyro Control

The basic rate gyro control system has two gains, the rate loop gain and the control gain. The rate loop gain K_R sets the damping and the control gain, K, sets the effective navigation ratio. Therefore specifications must be set on each of these gains. The set of gains used for these calculations at the zone boundaries are listed in Table 4 and Table 5.

Table 4

Rate Loop Gains for Rate Gyro Control System

Rate Loop Gain, K_R		Zone 1	Zone 2	Zone 3
Speed / Altitude	Low A	Low B	Med C	High D
Low A	0.0934	0.0465	0.0413	0.0344
Low B	0.115	0.0574	0.0508	0.0421
Med C	0.157	0.0793	0.0701	0.0581
High D	0.237	0.149	0.1311	0.108

Table 5

Control Gain for Rate Gyro Control System

Control Gain, K	Zone 1		Zone 2	Zone 3
Speed / Altitude	Low A	Low B	Med C	High D
Low A	20.6	5.1	3.4	2.7
Low B	24.1	5.9	3.9	3.1
Med C	31.7	7.5	5.0	4.0
High D	66.3	12.2	8.3	6.6

With two gain and two signal paths, the rate gyro system requires twice the number of specifications as the open loop system. Specifications on each are presented in the same order as in the open loop discussion.

a) System Data Rate

The rate loop contains both the aerodynamic short period frequency and the structural resonance frequency as seen by the gyro. The rate gyro loop sampling rate must be at least twenty times the crossover frequency loop. If the structural frequency is low enough and its damping is small enough, it will govern the sampling rate. For our purposes, we let the crossover frequency govern, then the sampling frequencies are those in Table 6.

Table 6

Rate Gyro Sampling Requirements

Sampling Rqmt / Zone	Highest Crossover Frequency(Hz)	Sampling Frequency (Hz)	Sampling Interval (msec)
Zone 1	3.4	68	14.7
Zone 2	4.1	102	9.8
Zone 3	6.7	134	7.5

The control signal path is very similar since its bandwidth is the closed rate gyro loop which is approximately equal to the crossover frequency. In this example the sampling rate requirements are similar for both the open loop and rate gyro systems because of the similarity of the highest crossover frequencies.

b) Number of Words

The number of words for each gain table and a 10 percent gain ratio can be calculated from Eq. (21) and Tables 4 and 5. The word requirements for each gain are tabulated below.

Table 7

Word Requirements for Rate Gyro Autopilot

	Rate Gain		Control Gain	
	R_D	N_W	R_D	N_W
Zone 1	2.47	11	4.72	18
Zone 2	3.80	15	9.32	25
Zone 3	6.89	22	24.6	35

c) Least Significant Bit

The least significant bit is 10 percent of the smallest control gain as previously described. However the rate gain affects the damping which can be allowed to vary 50 percent. Since the damping is proportional to the product of the rate gain and an aerodynamic gain which can be uncertain by 10 percent, the rate gain can be uncertain by 40 percent. Therefore the rate loop gain LSB is 40 percent of the smallest rate loop gain. Using Eq. (24) and the information from Tables 4 and 5 we can arrive at the requirement.

Table 8

Least Significant Bit Requirements for Rate Gyro Autopilot

Gain / Zone	Rate Gain	Control Gain
Zone 1	0.0186	0.51
Zone 2	0.0165	0.34
Zone 3	0.0138	0.27

d) Word Length

For each zone and each gain the word length is obtained from Eq. (25) and is tabulated below.

Table 9

Word Length Requirements for Rate Gyro Autopilot

	Rate Gain (bits)	Control Gain (bits)
Zone 1	$3.63 \rightarrow 4$	$6.56 \rightarrow 7$
Zone 2	$4.25 \rightarrow 5$	$7.54 \rightarrow 8$
Zone 3	$5.09 \rightarrow 6$	$8.94 \rightarrow 9$

Note that these word lengths are much smaller than those of the open loop control gains because of the smaller gain variations. The difference in word length requirements between the gains of this autopilot are due to differences in allowable uncertainties in the gain themselves.

e) Gain Coefficient Data Rate

For an allowable control gain change of 10 percent and rate loop gain change of 40 percent, the gain coefficient sample intervals are calculated from Eq. (30) and (31) and Tables 4 and 5. The data rate requirement is shown below.

Table 10

Gain Coefficient Data Rate Requirements for Rate Gyro Autopilot

	$(\Delta t)_{min}$ (msec)	
	Rate Gain	Control Gain
Zone 1	575	49.1
Zone 2	575	47.4
Zone 3	575	34.8

Note that both of these sample intervals are substantially less than those for the open loop system. The differences in requirements for both systems is due to gain variations and allowable uncertainties in the gains themselves.

f) Transport Lag

The transport lag mainly affects the stability of the rate loop. If we allow 3 deg (0.0524 rad) phase lag at the highest crossover frequency in each zone, we get from Eq. (32)

$$\tau_d = \frac{\theta}{\omega_{CR}} = \frac{0.0523}{\omega_{CR}} \text{ sec} \tag{37}$$

for the rate gain calculation. Note that ω_{CR} is determined by the rate gain. The control gain calculation is similar. Table 11 can be generated by utilizing the above formula with the information from Table 6.

Table 11

Transport Lag Requirements for Rate Gyro System

Transport Lag (msec)

Zone \ Gain	Rate Gain
Zone 1	2.45
Zone 2	1.65
Zone 3	1.24

g) Throughput

The rate gyro system requires two multiplies and an addition to compute the control commands. This is 21 operations per sampling interval per axis. Therefore the operations rate is $21/T_S$ per axis as listed below for each zone.

Throughput

Zone 1	1428	ops/sec/axis
Zone 2	2142	ops/sec/axis
Zone 3	2800	ops/sec/axis

h) Storage

The storage is calculated from the sum of the rate words and the control words. Since each of these words requires a different number of bits, this calculation is weighted to get bits/axis. Combining the information from Tables 7 and 9, we get

Zone 1 $S_R = 11 \times 4 + 18 \times 7 = 170$ bits/axis

Zone 2 $S_R = 15 \times 5 + 25 \times 8 = 275$ bits/axis

Zone 3 $S_R = 22 \times 6 + 35 \times 9 = 447$ bits/axis

Of course, if the words cannot be packed, the storage required will increase as

Zone 1 $S_R = 7 \times (11 + 18) = 203$ bits/axis

Zone 2 $S_R = 8 \times (15 + 25) = 320$ bits/axis

Zone 3 $S_R = 9 \times (22 + 35) = 513$ bits/axis

Thus if the data can be packed the storage requirements for the rate gyro system is somewhat less than for the open loop system. If the data cannot be packed the storage requirements are about the same. The smaller number of bits and required number of words of the rate gyro-system compensate for the extra gain.

Summary of Rate Loop Digital Control Specifications

The rate loop digital control specifications are summarized in Table 12.

Table 12

Summary-Rate Gyro Digital Control Specifications

	Zone 1		Zone 2		Zone 3	
	Rate Gain	Control Gain	Rate Gain	Control Gain	Rate Gain	Control Gain
a) System Data Interval (msec)	14.7		9.8		7.5	
b) Number of Words (words/axis)	11	18	15	25	22	35
c) Least Significant Bit	0.0186	0.51	0.0165	0.34	0.0138	0.27
d) Word Lengths (Bits)	4	7	5	8	6	9
e) Gain Coefficient Data Interval (msec)	575	49	575	47	575	35
f) Transport Lag (msec)	2.45		1.65		1.24	
g) Throughput, (op/sec/axis)	1428		2142		2800	
h) Storage (bit/axis)	170		275		447	
- packed						
- unpacked	203		320		513	

Accelerometer Control

A representative three loop accelerometer control system having a time constant of 0.3 sec, a cross-over frequency of 50 rad/sec, and a damping ratio of 0.7 is used to derive digital requirements. This system has four gains - the rate loop gain (K_R), the synthetic stability gain (ω_I), the acceleration gain (K_A), and the control gain (K_o). The gains used at the zone boundaries are tabulated in Table 13.

Table 13

Accelerometer Control System Gains

Accelerometer Gain

K_A

Altitude \ Speed	Zone 1: Low A	Zone 1: Low B	Zone 2: Med C	Zone 3: High D
Low A	4.344	0.3879	0.2829	0.7431
Low B	6.484	0.7316	0.3828	0.4597
Med C	12.34	1.706	0.7795	0.5788
High D	41.84	6.768	3.128	1.969

Synthetic Stability Gain

ω_I

Altitude \ Speed	Zone 1: Low A	Zone 1: Low B	Zone 2: Med C	Zone 3: High D
Low A	21.28	22.13	16.23	7.617
Low B	20.77	22.89	18.93	13.14
Med C	18.86	23.28	21.63	18.53
High D	12.36	21.30	22.55	22.27

Rate Loop Gain

K_R

Altitude \ Speed	Zone 1: Low A	Zone 1: Low B	Zone 2: Med C	Zone 3: High D
Low A	0.2923	0.1096	0.06520	0.0408
Low B	0.4467	0.1609	0.09492	0.0594
Med C	0.9181	0.2999	0.1758	0.1093
High D	4.927	1.108	0.6098	0.3698

Control Gain

K_o

Altitude \ Speed	Zone 1: Low A	Zone 1: Low B	Zone 2: Med C	Zone 3: High D
Low A	1.381	3.131	2.948	1.556
Low B	1.264	2.171	2.492	1.932
Med C	1.147	1.532	1.777	1.784
High D	1.045	1.140	1.202	1.241

In this control system the rate loop gain (K_R) sets the crossover frequency, the synthetic stability gain ω_I) sets the damping, the acceleration gain (K_A) influences the time constant and the control gain (K_o) sets the effective navigation ratio. As with the other flight control systems, specifications must be set on each of the four gains. Specifications on each of the gains are discussed in the same order as with the other flight control systems. It should be noted that a different analog time constant, damping, or crossover frequency gives a different set of gains which, in turn, would give a different set of digital specifications. Therefore an analog-digital trade off exists which can be used to meet digital hardware or software constraints.

a) System Data Rate

As with the other flight control systems the crossover frequency of each of the loops governs the sampling frequency. Using the rule of thumb that the loop sampling rate must be at least twenty times the crossover frequency we derive the following sampling requirements for the accelerometer flight control system in all 3 zones.

Table 14

Accelerometer System Data Rate Requirement in All Zones

Gain	Highest Crossover Frequency (Hz)	Sampling Frequency (Hz)	Sampling Interval (msec)
K_R	7.96	159.2	6.28
ω_I	1.91	38.2	26.2
K_A	0.478	9.56	105
K_o	0.478	9.56	105

b) Number of Words

Using the gains in Table 13, the number of words required in each zone are shown in Table 15.

Table 15

Number of Words for Accelerometer Control

	K_A		ω_I		K_R		K_o	
	R_D	N_W	R_D	N_W	R_D	N_W	R_D	N_W
Zone 1	16.7	31	1.1	2	4.07	16	2.48	11
Zone 2	43.6	41	1.4	5	14.1	29	2.73	12
Zone 3	148	54	1.9	8	120	52	3.00	13

c) Least Significant Bit

The control gain and rate gain can only be allowed to vary 10 percent because the navigation ratio and crossover frequency can only be allowed to vary 10 percent. Since the damping can be allowed to vary 40 percent (as with the rate gyro system) the synthetic stability gain can also vary by that amount. For capability against high performance targets the time constant and therefore the acceleration gain can not be allowed to vary by more than 50 percent. Using Eq. (24) and the data from Table 13 give rise to the following table.

Table 16

Least Significant Bit for Accelerometer Gains

	K_A ($\tau \rightarrow 50\%$)	ω_I ($\zeta \rightarrow 40\%$)	K_R ($\omega_{CR} \rightarrow 10\%$)	K_o ($\dot{N}' \rightarrow 10\%$)
Zone 1	0.194	8.31	0.0110	0.126
Zone 2	0.141	6.49	0.00652	0.115
Zone 3	0.141	3.05	0.000409	0.105

d) Word Length

Using the LSB data from Table 16, the information from Table 13 and Eq. (25) yields the following word length requirements in Table 17.

Table 17

Word Length Requirements-Accelerometer Control

	K_A	ω_I	K_R	K_o
Zone 1	7	3	7	6
Zone 2	8	3	9	6
Zone 3	10	4	15	6

The least stringent requirements in word length are for the synthetic stability gain. The requirement is relaxed because this quantity does not vary widely with mach and altitude and we can allow a 40 percent variation in this gain. The control gain requirement is more stringent mainly because we are allowing only a 10 percent variation on the gain. The fact that this gain varies only slightly over the entire mach-altitude regime prevents the word length requirement on the gain from being more stringent. The accelerometer gain requirements are stringent because of the wide dynamic range of this gain. The most stringent requirements are on the rate gain because of its wide dynamic range and the 10 percent constraint on its variability.

e) Gain Coefficient Data Interval

The accelerometer gains in Table 13 were used with Eq. 31 to develop the minimum data interval at which the gains need to be changed.

Table 18

Gain Coefficient Data Interval-Accelerometer Control

$(\Delta t)_{min}$ (msec)	K_A (p = 50%)	ω_I (p = 40%)	K_R (p = 10%)	K_o (p = 10%)
Zone 1	74	5860	83	122
Zone 2	74	1408	72	122
Zone 3	74	392	44	122

f) Transport Lag

The transport lag that gives a phase lag of 3 deg in its signal path is found as follows.

Rate Path $\tau_d = \dfrac{\theta}{\omega_{CR}} = 0.0524/50 = 0.00105$ sec

Synthetic Stability $\tau_d = 0.0524/12 = 0.00437$ sec

Accelerometer and Control Gain $\tau_d = 0.0524/3 = 0.0175$ sec

Note that the rate path is the most stringent and it's still about a millisecond, the same as for the rate gyro system.

g) Throughput

The throughput is calculated from the operations required to compute the control command. These are four multiplies, one for each gain, and three addition, one after each multiply except the last. The equations are:

CO = acceleration command

$A = K_o \times CO$ + Accel at 159.2 Hz

$B = K_A \times A$ + Gyro at 38.2 Hz

$C = \omega_I \times B$ + Gyro at 9.56 Hz

$D = K_R \times C$ at 9.56 Hz

As before, we weight a multiply as 10 adds, then each of the four equations is 11 adds or operation. Weighting each by its frequency of occurrence gives an operations throughput requirement of 2381 operations/second.

h) Storage

The storage required increases with zone. If the words are packed, then Eq. (33) is applied to each gain and the results are summed to get the storage for each zone, as follows

Zone 1, $S_R = 31x7 + 2x3 + 16x7 + 11x6 = 401$ bits

Zone 2, $S_R = 41x8 + 5x3 + 29x9 + 12x6 = 676$ bits

Zone 3, $S_R = 54x10 + 8x4 + 52x15 + 13x6 = 1430$ bits

If the word lengths are held constant at the maximum word length, i.e., packing is not possible, then the calculations change as follows,

Zone 1, $S_R = 7x(31+2+16+11) = 420$ bits

Zone 2, $S_R = 9x(41+5+29+12) = 783$ bits

Zone 3, $S_R = 15x(54+8+52+13) = 1905$ bits

In either case the numbers are not much different.

Software Comparisons

A comparison of software specifications for the open loop, rate gyro controlled, and accelerometer controlled system is shown in Table 19. The open loop system is not usable for radar guidance so it really can't be compared with the others. However, both the rate gyro and the accelerometer system can be used for radar guidance and the table reveals many interesting comparisons. First note that the transport lag and throughput of the rate gyro system is very sensitive to the zone size, but the transport lag and throughput of the accelerometer system are independent of zone size. Both systems require more storage as zone size increases with the accelerometer system requiring more storage. A close look at the throughput shows that for Zone 1, the low altitude, low speed zone, the rate gyro system requires less throughput than the accelerometer system. However, for Zone 2, the medium altitude, medium speed zone, the throughputs are about the same, and for Zone 3, the zone that covers all altitudes and speeds, the accelerometer system requires less throughput than the rate gyro. Clearly as the performance envelope grows, the additional hardware of the accelerometer system is reducing throughput requirements. Nevertheless, the accelerometer system always requires more storage. If this poses a problem, algorithms for interpolation and extrapolation of the gains can be used to trade-off storage for throughput.

It must be remembered that the sample calculations shown in this paper do not cover all of the trade-offs possible. The rate gyro gains were based on achieving a given damping ratio within a prescribed accuracy. Change these numbers and the requirements change. Likewise the accelerometer gains are based on achieving a certain time constant, damping ratio, and crossover frequency within a prescribed tolerance for each of these parameters. Change them, the gains change and so do the digital requirements. A complete set of trade-offs can be run using computer programs that do all the calculations required over the region of application of the guided missile system to be designed.

Table 19

Comparison of Digital Control Specifications

	Open Loop	Rate Gyro		Accelerometer			
	K_{OL}	K	K_R	K_o	ω_I	K_R	K_A
a) System Data Interval (ms)							
Zone #1	17.75	14.7	14.7	105	26.2	6.28	105
#2	12.2	19.8	9.8	105	26.2	6.28	105
#3	9.35	7.5	7.5	105	26.2	6.28	105
b) Number of Words (words/axis)							
#1	28	18	11	11	2	16	31
#2	40	25	15	12	5	29	41
#3	57	35	22	13	8	52	54
c) Least Significant Bit							
#1	0.0198	0.51	0.0186	0.26	8.31	0.011	0.194
#2	0.0117	0.34	0.0165	0.115	6.49	0.00652	0.141
#3	0.00775	0.27	0.0138	0.105	3.05	0.000409	0.141
d) Word Length (Bits)							
#1	8	7	4	6	3	7	7
#2	10	8	5	6	3	9	8
#3	12	9	6	6	4	15	9
e) Gain Coefficient Data Interval (ms)							
#1	19.2	49	575	122	5860	83	74
#2	19.2	47	575	122	1408	72	74
#3	19.2	35	575	122	392	44	74
f) Transport Lag (ms)							
#1	5.9	2.45		17.5	4.37	1.05	17.5
#2	4.1	1.65		17.5	4.37	1.05	17.5
#3	3.1	1.24		17.5	4.37	1.05	17.5
g) Throughput (op/sec/axis)							
#1	563	1428		2381			
#2	820	2142		2381			
#3	1070	2800		2381			
h) Storage							
#1	280	170		401			
packed #2	400	275		676			
#3	570	447		1430			
unpacked #1	280	252		420			
#2	400	400		783			
#3	570	630		1905			

CONCLUSIONS

This paper makes comparisons of the hardware-software considerations both in terms of performance and requirements for major flight control systems. The instrumentation chosen influences guidance parameters - effective navigation ratio, damping and response time. The instrumentation chosen also influences on-board computer sizing requirements. A proper choice of both hardware and software specifications is needed to minimize cost.

It is shown that the open loop system yields an effective navigation ratio that depends upon both altitude and speed. Its response is that of the airframe and its damping is so low that the system is not usable for radar homing against aircraft targets. Although the computer throughput and storage requirements for this system are minimal, large word lengths are required to make the system work over a large altitude - speed space.

The rate gyro system improves the low damping ratio of the airframe but cannot always stabilize an unstable airframe. The computer throughput and storage requirements are about the same as the open loop system, but the word length requirements are less stringent.

The accelerometer flight control system removes the speed - altitude dependence of the effective navigation ratio, has good damping, short response time and can stabilize an unstable airframe. The software penalty for the improved performance is significantly larger computer throughput and storage requirements and word lengths slightly greater than the open loop system for Zone 1, the small zone. However, if it is necessary to cover a large altitude-speed space, the throughput requirements of the accelerometer system are smaller than those of the simpler gyro system.

The techniques presented in this paper show some of the important steps which must be taken to determine the hardware and software required for a particular guided missile flight control application. Both the hardware and software requirements have profound influence on cost and performance, and both must be properly specified to obtain optimum performance at minimum cost.

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ANALYSIS OF COMPUTING SYSTEM CONFIGURATIONS FOR HIGHLY INTEGRATED GUIDANCE AND CONTROL SYSTEMS

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SUMMARY

Highly integrated guidance and control systems promise improvements in mission performance and fault tolerance, but there is an attendant requirement for centralizing at least portions of the computing system resource. The resulting combined centralized/distributed or hierarchical architectures present system designers with challenges, such as assuring that adequate margins are allowed for processing nodes and communications links throughout the system. Few tools exist for analyzing and validating these complex computing system architectures in their early development stages. This paper highlights the importance of early and sustained validation of architectures for highly integrated systems, and focuses on two early validation tools developed by Systems Control in its project work. A description of the two tools, 1) Generalized Reliability and Maintainability Program (GRAMP), and 2) Functional Emulation, is presented, along with a discussion of their utility in the development of highly integrated guidance and control systems.

INTRODUCTION

Increasingly sophisticated military operations require correspondingly more capable and reliable airborne systems for avionics, weapons management, and control of flight. The need for coordination among airborne systems and the potential for performance improvements by addressing combined performance objectives have spurred the exploration of functional integration (e.g., integrated fire/flight control, integrated flight/ trajectory control, integrated flight/propulsion control). The importance of such highly integrated systems to mission success and survivability has elevated the reliability requirements of traditional "outer-loop" functions (navigation, guidance, C^3) to a level where they rival those of flight control for tolerance to faults.

Expected Benefits of Integration

Figure 1 illustrates some of the expected benefits of the moment toward highly integrated guidance and control systems. Among these are:

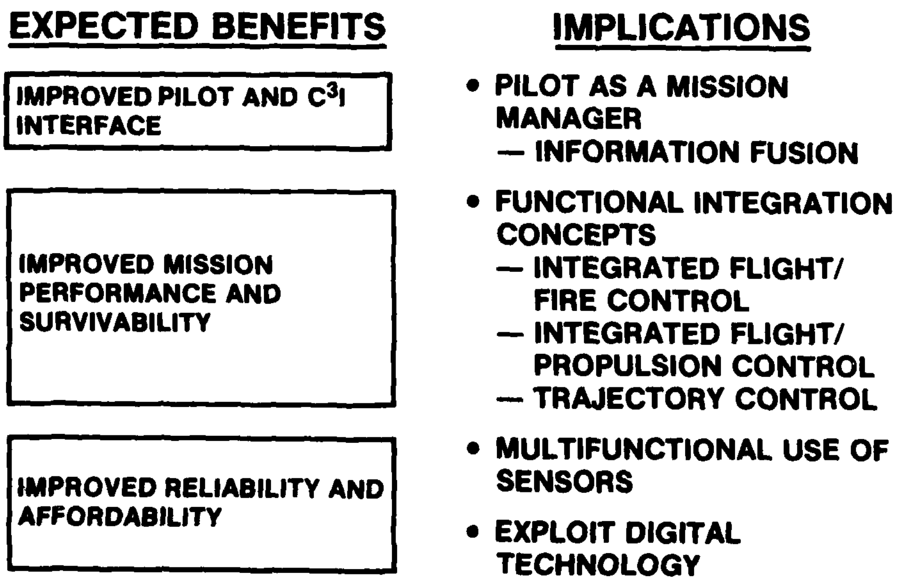


Figure 1 Highly Integrated Systems

- Improved pilot and C^3I interface. Such improvements provide a single, coherent interface for the pilot as he manages his sophisticated aircraft, either autonomously or in cooperation with other vehicles.

- Improved mission performance and survivability. Seeking such improvements has spurred the development of functional integration concepts, such as those listed. Coupling these previously autonomous systems often increases data transfer requirements and total processing requirements. It also may elevate the fault tolerance requirements of avionics functions which were previously not flight critical.

- Improved reliability and affordability. These improvements can be realized by, for example, multi-functional use of sensors and by exploiting digital technology. Here the self-test capability of digital system scan be used to provide redundancy.

Impact on Computer System Configurations

With recent advances in computer architecture and data-bus technology, the capability now exists to develop computing system configurations which satisfy the performance and reliability requirements of functional integration while optimizing affordability. Such configurations benefit, for example, by the cost advantages of multi-functional use of sensors. While considerable work has been done on the development of fault-tolerant configurations for avionics and flight control systems separately, the design of appropriate configurations for highly integrated systems is an immature technological area. Other authors [1] have postulated a trend towards hierarchical architectures, which attempt to derive the benefits of both centralized and distributed processing.

Hierarchical, distributed, fault-tolerant architectures present system designers with a number of unique design challenges, such as hardware/software partitioning, mode/task/data synchronization, and redundancy management. Design trade-offs associated with meeting these challenges can have far-reaching impacts on the performance and reliability, and ultimately the affordability, of the developed system. Clearly, automated tools need to be developed and applied to the quantitative analysis of computing system architectures in their earliest possible development stages.

This paper presents two techniques for early quantitative analysis of alternative computing system architectures for highly integrated systems. These techniques are being developed by Systems Control in its on-going work in fault-tolerant digital flight control, integrated guidance and control, and the digital control of jet engines. Example applications of the techniques are also presented, and an assessment is offered of their utility in reducing the overall development costs of complex systems.

NEED FOR EARLY AND SUSTAINED SYSTEM VALIDATION

Previous digital systems development programs have been marred by many problems which tend to escalate development costs. Most of these problems can be traced to:

1) inadequate initial allocation for duty-cycle and memory margins in system nodes and links;

2) inadequate, inaccurate, or inconsistent documentation of requirements and design specifications;

3) belated discovery of design errors,

or some combination of these. Inadequate margins force the system implementers to compromise many desirable features of the design; for example, if duty-cycle timing is a problem, an otherwise straightforward and easily analyzed design may be obscured by reducing the iteration rate of selected functions or resorting to coding "tricks". Often this problem is compounded because its extent is discovered late in the development process, causing extensive redesign to implement the shortcuts. Inadequate documentation can lead to confused communications between analysts, designers, implementers, and testers. In several development programs, such confusion has been responsible for more than half of the total system errors found [2]. Belated discovery of design errors necessitates iterations over prior activities, thereby increasing development costs and "slipping" development schedules.

Figure 2 illustrates the relative cost of correcting errors during various development phases, as found in a number of large-scale software development programs [2].

Front-End Emphasis

The nature of these historical system development problems indicates that high pay-off can be achieved by placing much more emphasis on the early stages of system development. For illustration, an idealized structure of the modern system development process [3] is shown in Figure 3. In the structured approach, the system development is divided into some number of distinct stages (in this case, eight stages), each having identifiable outputs and each undergoing a specific verification or validation step. Phased development as illustrated in Figure 3 is reflected in recent United States Department of Defense Directives for system acquisition (DOD Directive 5000.1, 5000.2), related military standards (MIL-STD-483, MIL-STD-490), and more recently in Federal Aviation Administration and EUROCAE recommendations for the certification of airborne digital systems.

Considerable work has been done in the development of automated tools and techniques for the realm of "software engineering", which covers roughly Stages 3 through 5 of Figure 3. Given a complete and coherent system design specification (Stage 2), the software design can be accomplished reliably using disciplined documentation control and such proven techniques as design walk-throughs. Often a "program design language" is used to improve communication between engineers and programmers, and to provide a basis for semi-automated verification of design constructs. Once the coding process (Stage 4) has

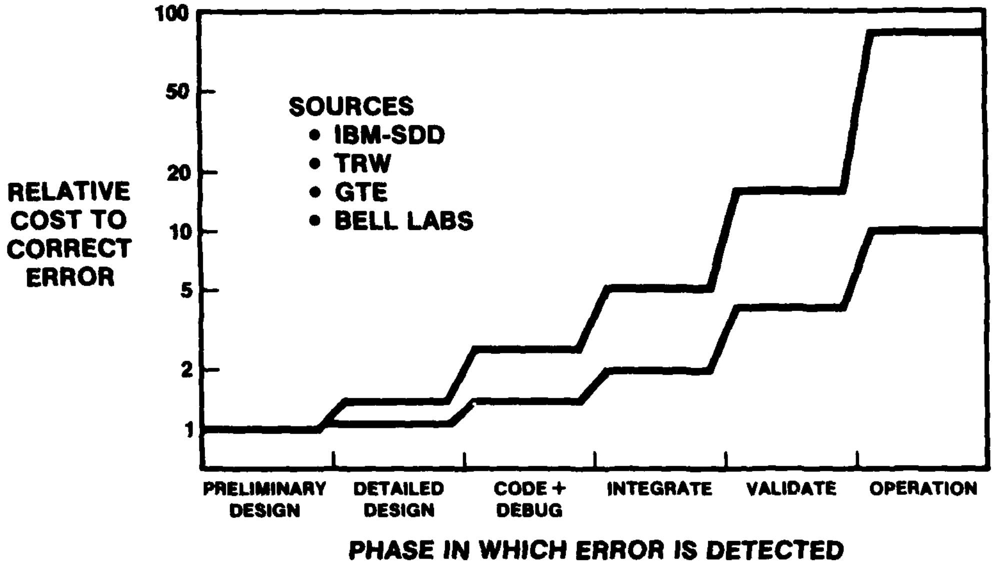


Figure 2 It Pays to Catch Software Errors Early

begun and code is available, then numerous automated tools (logic analyzers, assertion checkers, flow charting programs, etc.) can be used for verification. Stage 5 and subsequent stages often use for validation a "hot bench" simulation facility, which may contain increasing amounts of the actual hardware and software.

Approaches to Early Validation

Few automated or semi-automated techniques, however, have been developed to address the verification of the very early development stages, namely system requirements and system design in Figure 3. Instead, modern practice relies on formal and informal reviews, and analytical studies and trade-off analyses of various aspects of the system design. Recognizing the high potential pay-off of the early detection of requirements deficiencies or system-level design errors, procuring agencies have occasionally resorted to multiple awards of preliminary design contracts to allow selection of the most viable candidate. Often this does not accomplish the goal of achieving a more complete preliminary design; rather, the competition may result in several preliminary designs, none of which is sufficiently mature.

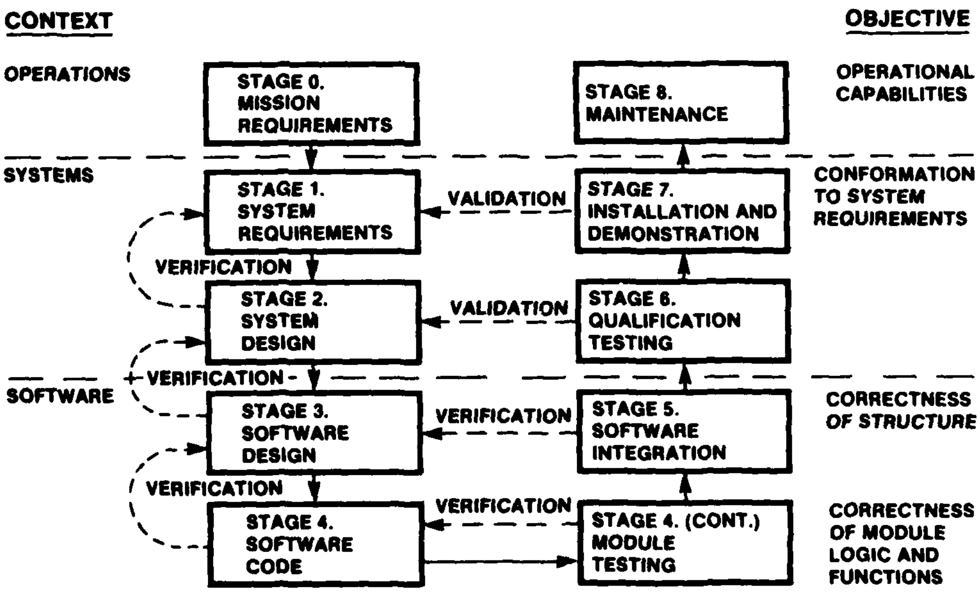


Figure 3 Structured Development Process

A preferable approach is to increase the scope of the preliminary design activity, possibly including the services of an independent assessment contractor to assist in proposing and evaluating trade-off analyses. The independent assessment contractor may be the same organization which is often employed in later development stages to perform so-called independent verification and validation (IV&V) of software. The distinction of this approach is that the independent assessment contractor is brought on early in the development process, before the system design or software design is firm, and his services are sustained thoughout the development process. This sustained support has the advantage of retaining knowledge of the system design in the independent assessment contractor to aid in transitioning the system from development to deployment. Systems Control has performed in this independent assessment role in a number of developments, notably the on-going Advanced Fighter Technology Integrator (AFTI) F-16 development of a digital flight control system and integrated fire/flight control concept. The value of such sustained validation support has been noted in other development programs, for example, the B-1 Navigation System development [4].

QUANTITATIVE TOOLS FOR EARLY VALIDATION

The key to successful performance of sustained system validation is the development and maintenance of a set of tools and techniques for quantitative system evaluation. As mentioned above, modern software engineering provides many options for automated or semi-automated system validation once software code has begun to appear.

What is needed are quantitative tools which can be applied to early requirements analysis and design assessments. In its prior work, Systems Control has developed a number of such tools, ranging from performance analysis simulations to detailed statement-level emulations. Two of these techniques which are particularly applicable to early validation are presented in the remainder of this paper. These techniques are: 1) Generalized Reliability and Maintainability Program (GRAMP) and 2) Functional Emulation.

After a brief description of the two techniques, it will be shown how they can be used jointly by an independent assessment team to provide early design validation.

GENERALIZED RELIABILITY AND MAINTAINABILITY ANALYSIS

The Generalized Reliability and Maintainability Program (GRAMP) is a general analysis tool originally developed by Systems Control for the design of fault tolerant or other more general systems. As discussed below, GRAMP can be an extremely useful tool for performing early and sustained validation of distributed, redundant systems with respect to reliability, maintainability, cost and weight requirements.

GRAMP is based on the theory of coherent system repair models (CSRM) [5], which was modified to account for problems intrinsic to fault tolerant systems design such as detection/ recovery strategies and component and analytical redundancies. Figure 4 presents an overview of the program structure. The system to be modeled is first broken down into components which are replaceable units from a design standpoint and whose failures are approximately independent, i.e., failure in one component will not induce failure in another. GRAMP takes component costs, failure rates, and fault-tolerance coverages in addition to the user-specified system design and maintenance policy, and evaluates such quantities as expected system operating and support costs per unit time, reliability, Mean Time Between Failures (MTBF), and Mean Time Between Repairs (MTBR). Sensitivities of these output values to various component input parameters can be computed.

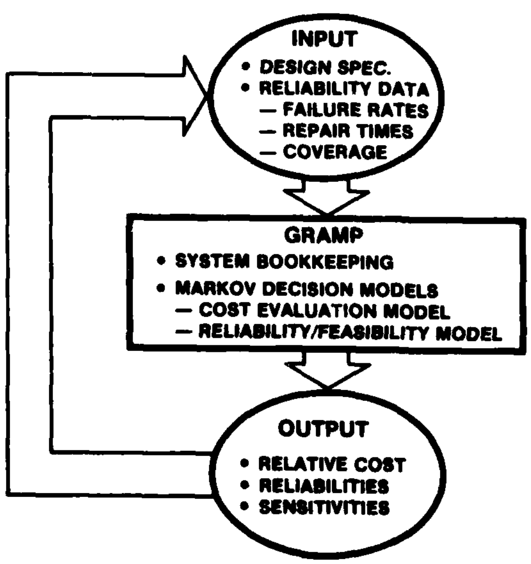


Figure 4 Overview: Generalized Reliability and Maintainability Program

The CSRM formulation is attractive in that, although previously having never been applied to fault tolerant system design, it supplies the framework for solving such problems. It has the capability to handle repairable systems including limited opportunistic maintenance, analytical redundancy and multiple objectives within a single framework. A dual Markov model approach is used to perform the computations: the Cost Evaluator Model (CEM) is a static repairable system model which evaluates cost, MTBF, MTBR, and other steady state quantities. Output from the CEM feeds into the dynamic closed-system Reliability/Feasibility Model (RFM) for evaluation of reliability over specified time intervals or mission lengths.

Advantages of GRAMP

One of the principal advantages of the GRAMP approach to system evaluation or validation is its efficiency. In using it, one makes more restrictive yet realistic assumptions (e.g., constant failure rates, stationary maintenance policies) on the system, thus allowing sustained validation at low cost to the user, i.e., numerous test runs varying system design configurations and input data. Monte Carlo simulations, although applicable and containing more "true to life" assumptions, are often undesirable because of high costs per run. The detailed system input data required to run the simulation is not likely to be available, especially at early stages of the system design process.

Along with its efficiency relative to conventional Monte Carlo methods, GRAMP has several unique features which make it capable of addressing highly integrated systems. For example, GRAMP is designed to handle non-standard aspects of multiple-computer systems, such factors as analytical redundancy, imperfect cross-strapping, and fault-tolerance coverage. In addition, GRAMP will calculate multiple objective functions and the sensitivities of these objectives to input data. This is necessary because reliability, maintainability, and cost are so inter-related that each can not be controlled independently.

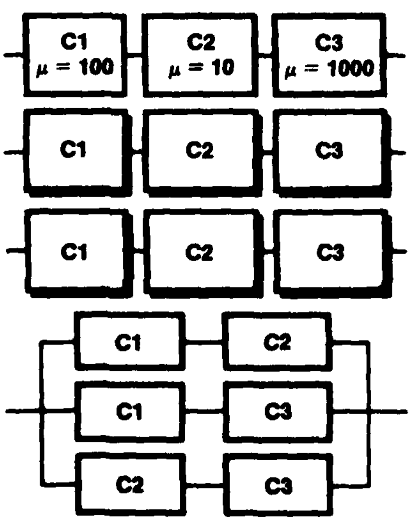
GRAMP Applications

As a design tool, Systems Control has used GRAMP for design of a full authority fault-tolerant engine electronic controller (FAFTEEC). Numerous possible designs and maintenance strategies were tested with the objective of obtaining five or six "good" designs to be analyzed in detail using Monte Carlo methods. System design goals included the simultaneous minimization of weight and cost and maximization of reliability and mean time between repairs (MTBR) subject to minimum reliability and MTBR "standards" or "goals". The "best" candidate configurations were then analyzed in detail using Monte Carlo methods. Results of the study are being compiled and will be available in [6].

GRAMP Example

To illustrate some of these ideas, Figure 5 shows a very simple example of a GRAMP analysis. Here, a three-component subsystem is analyzed, with and without component redundancy and with two different maintenance policies. Maintenance Policy A is to repair whenever a component failure is detected. Maintenance Policy B is to repair only after the total end-to-end function is lost, i.e. system redundancy is exhausted.

The top configuration is simplex, the middle two are duplex by component, and the third uses analytical redundancy. The analytical redundancy means that, in this example, each of the three parallel strings of components shown is capable of performing the subsystem function. Such analytical redundancy, when it is appropriate, is usually implemented by means of software.



MAINT. POLICY	FAILURE PROB. (4 hr. Mission)	MBTR	O & S PER HR.
A	4.4×10^{-3}	901 HR.	\$1.27 x 10^{-3}
B	2.9×10^{-3}	1342 HR.	\$1.62 x 10^{-3}
A	1.1×10^{-4}	450 HR.	\$2.52 x 10^{-3}
A	4.4×10^{-4}	9188 HR.	\$0.27 x 10^{-3}

- Maintenance Policy A: Repair on first failure.
- Maintenance Policy B: Repair on loss of function.

Figure 5 Example GRAMP Outputs

Referring to the figure and progressing downward from the top configuration, the second line offers better reliability due to duplex component redundancy. The duplex redundancy, along with the maintenance policy of not repairing until the total function is lost, yields infrequent repairs and a favorable operating and support cost. Reliability can be improved further in the third configuration by repairing as a single component failure is found, but frequency of repair goes up, as does the operating cost. The bottom configuration exploits analytical redundancy to yield about the same reliability as the third configuration, but at greatly reduced operating cost.

Limitations of GRAMP

A limitation of GRAMP, and of all reliability/ maintainability modeling techniques, is that their results depend on the quality of data concerning component failure rates and fault detection/isolation coverage. While GRAMP can yield valuable "sensitivity" information using inaccurate input parameters, it is much preferable to update continually the failure rate and coverage parameters as more is learned about the system under development.

The ability to consider fault-detection/isolation coverage, a characteristic of GRAMP which qualifies it for analyzing fault-tolerant systems, places the additional burden (over less powerful techniques) of estimating coverage parameters. Self-test coverage can be assessed in straight-forward ways, for example, by injecting a large number of simulated failures into a gate-level emulation and recording the number of errors found by the self-test algorithms. It is more difficult to assess the effectiveness of coverage other than self-test, for instance that coverage internal to the system's redundancy management software, which may use cross-channel monitoring or analytical redundancy techniques. A means of estimating this type of system-level fault coverage is provided as a by-product of the second early-validation tool, Functional Emulation.

FUNCTIONAL EMULATION

Functional Emulation refers to all-software simulation of the architecture of the developing system. As distinct from the conventional statement-level emulation of one machine by another, functional emulation simulates selected hardware and software components of the developing system at a higher level of detail. The level of detail is selected to yield the simplest possible implementation while assuring adequate fidelity for the analyses to be performed.

The technique entails modeling the processing nodes and communications links in the system, both hardware and software components. Within the nodes and links, hardware components are identified to the level of detail commensurate with component failures the user wants to inject. Software components of the flight systems are modeled with varying fidelity, with the most attention given to those software tasks which interact intimately with the hardware, for example, the executive, link interfaces, and redundancy management algorithms. Applications tasks such as control laws need be modeled only to analyze the interactions between basic closed-loop vehicle dynamics and the system architecture from a timing point of view. Questions which can be addressed by this functional emulation are, for example:

- What is the minimum acceptable time to detect and isolate various types of failures? Is that time requirement met by the design?
- Does the system's method for failure detection, isolation, and recovery induce unacceptable transients in aircraft response?
- How do such phenomena as executive interplay and link delays add to the overall computational delay from sensor to control surface?
- Are adequate margins for processing node duty-cycle and link transmissions met by the system design?

Functional Emulation Example

To illustrate the application of this technique, Figure 6 shows the approximate level of detail modeled in the Systems Control emulation of a triplex digital flight control architecture, similar to that of the AFTI/F-16 aircraft in development. The system has approximately fifteen motion sensors, most triply redundant.

The three flight computers, which for illustration operate asynchronously, contain various components for analog-to-digital conversion, processing, transmitting and receiving on inter-computer data links, memory, etc. The Functional Emulator will allow the user to invoke models of failures of any of these components (for example, a link transmitter can be failed to quiet, or memory locations can be made to fail to update). The servo amplifier switching logic, under software control by the flight computers, attempts to remove failed actuator servo amplifiers and failed computers from driving the aerosurfaces. The servo amplifiers are triply redundant with a fourth spare. Actuator hydraulics are only dual redundant, with a mechanical voting scheme for tripping from a hydraulics channel being driven by disagreeing servo-amplifier signals. Models of the flight software functions (executive, failure management, control laws, etc.) are integrated with the hardware models into the all-software emulation. A linear aircraft model, when combined with the control law model, provides the characteristic aircraft response for dynamic studies.

This example is already complex, but it can be made more so when one considers that, for integrated fire/flight control, additional computers for target tracking and director fire control will be issuing commands to the flight control complex over the avionics multiplex data bus. In this case,

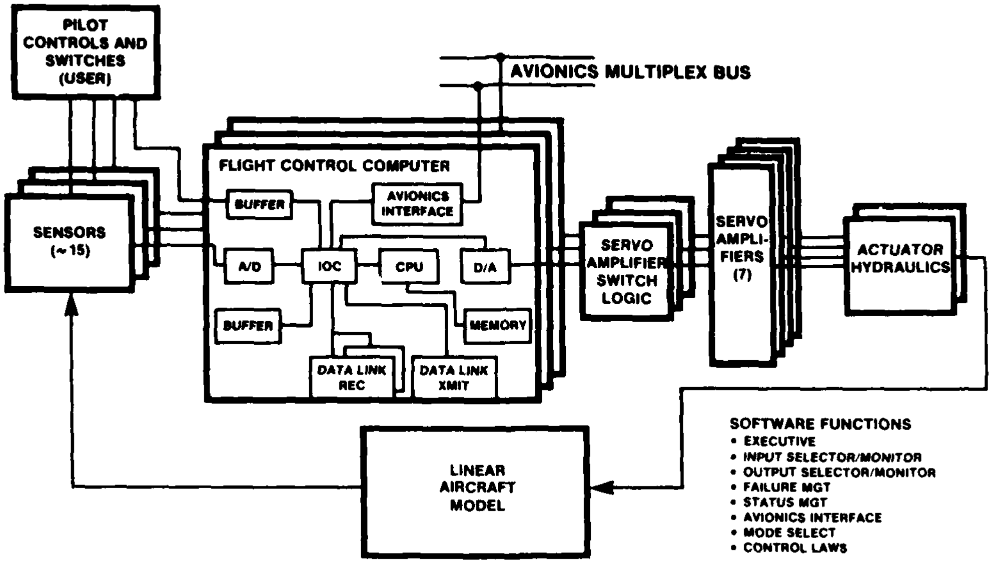


Figure 6 Example Configuration for Triplex Digital Flight Control System

executive interplay and the resulting effective computational delay are additional factors to be considered. This added complexity will be neglected for the remainder of this example.

During an execution of the Functional Emulator, the interactive computer program queries the user to specify initial conditions and parameters (e.g., skews between computers, clock errors, sensor noise, etc.) and failure insertion scenarios. The program then begins a simulated run and flags notable events to be displayed in text on the CRT display and/or printer. Hard copy plots can also be generated, at user option, to examine dynamic phenomena such as the transients induced while failures are being detected, isolated, and accommodated by the flight software models. Figure 7 shows plots from a run where the indicated failures were injected at times when the aircraft is responding to step commands in normal acceleration. The transients shown in the first step response in Figure 7a result from the failed computer's outputs temporarily being passed to the servos until a software threshold is reached, at this time the other computers remove it from controlling the servos. Such thresholds are necessary to allow for normal discrepancies between computers due to asyncronism, or to accommodate temporary low-level noise. The transient observable in the second step response in Figure 7a is due to failure of a servo amplifier in such a way as to cause the hydraulic system to switch between channels. Figure 7b shows corresponding plots of the horizontal tail (elevator) response of each of the three flight control computers; the asynchronism between computers is readily apparent.

Utility of Emulation

Note that developing a Functional Emulator as described above requires the development of models of the flight software to some degree. This can be used to advantage in that the emulator models constitute a validated specification for detailed flight software development. The models are coded in a high-level language which, in effect, serves as a program design language (PDL) for flight software programmers to use as a guide.

The further utility of the functional emulation technique in the system development process is summarized in the following points:

1) Early Design Validation. The software algorithms for executive and redundancy management functions, which interact intimately with the hardware, are integrated functionally at an early stage. This identifies basic design errors, omissions, and inconsistencies, and it forces design ambiguities to be resolved before progressing to the later implementation stages.

2) Architecture Analysis. Emulation provides a quantitative tool for evaluating trade-offs and discovering pathological failure sequences and timing interactions.

3) Assuring Adequate Margins. Given estimates of the processing time requirements of each software task and the block size and transmission rates of the links, the emulator can keep track of the resource utilization of all the nodes and links in the system in order to assess reserve capacity (margin). This job is often difficult to do manually because many software tasks are performed on demand when a particular event happens (e.g., a failure is detected).

4) Design Change Evaluation. The emulation technique provides a means for quick evaluation of proposed design changes, which may occur at any point in the design process.

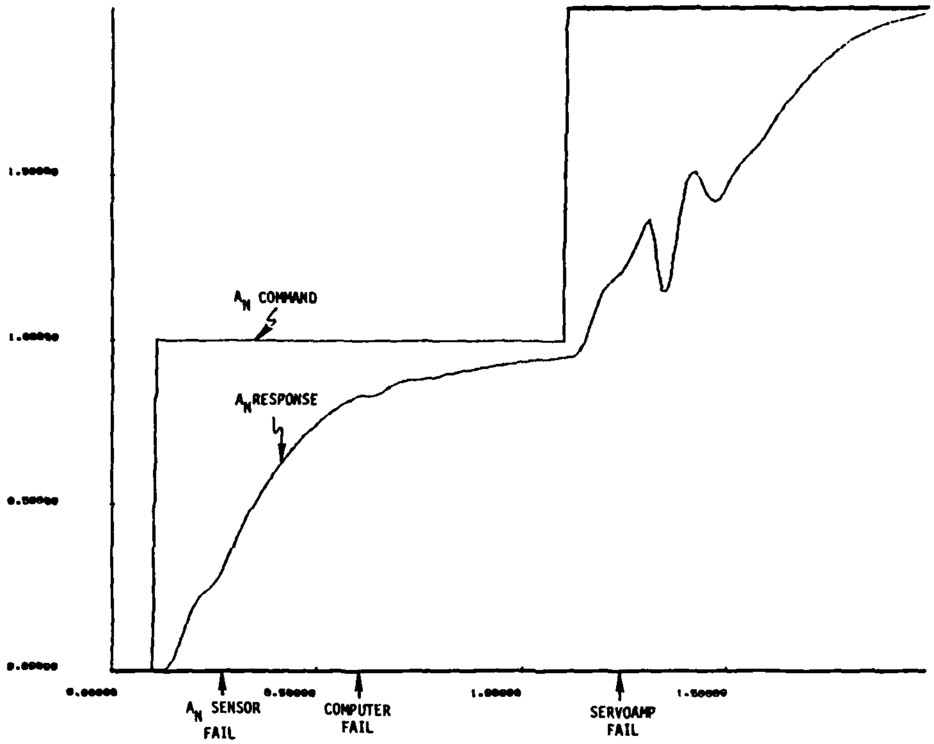


Figure 7a Normal Accelerations (A_N) Command Response with Failures

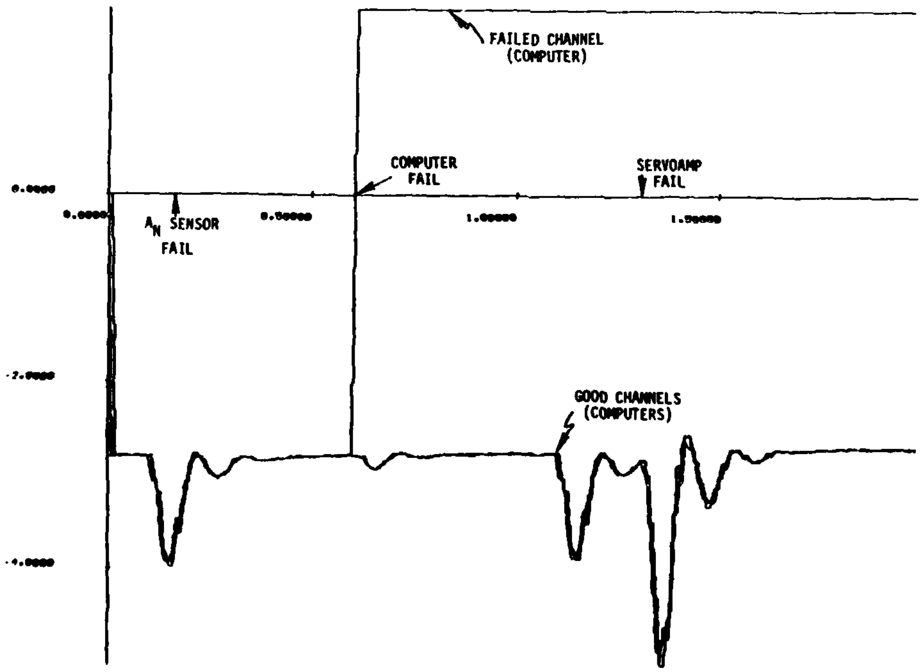


Figure 7b Elevator Command Response with Failures

5) Test Direction. Admittedly, the emulator cannot replace laboratory testing as a final validation of the design. However, it can serve to direct the testing activity towards potential high pay-off areas and reduce the scope of testing in those areas where the emulator results are of high fidelity.

RELATIONSHIP OF GRAMP AND FUNCTIONAL EMULATION

The two tools represented by GRAMP and Functional Emulation can work together to provide a powerful, quantitative capability for early design assessment. The relationship of the two techniques as tools used by an independent assessment team is illustrated in Figure 8.

As shown, GRAMP is used to verify reliability predictions and life-cycle cost projections. It also yields sensitivity results, which can identify weak links in the architecture.

Emulation is used to yield information about how the system architecture will work in a dynamic environment. It estimates whether the architecture will meet requirements relating to fault-tolerance coverage, hardware/software interaction, and processing reserve.

Using these quantitative tools, the independent assessment team is able to provide a specific and authoritative analysis of the system architecture.

CONCLUSIONS AND RECOMMENDATIONS

Experience in the development of highly integrated guidance and control systems has shown the need for early and sustained validation of the developing system. The services of an independent contractor for design assessment and software validation over the full development process is seen to be preferable over intensive but belated independent software verification and validation. The sustained approach would also seem to be more cost effective than competitive procurements for preliminary design, followed by conventional development practices for the winning competitor.

Quantitative evaluation tools, while plentiful for the later software development and testing phases, are lacking for the early system-level development phases where pay-off potential is great. The tools discussed in this paper, Generalized Reliability and Maintainability Program (GRAMP) and Functional Emulation, focus on early validation and have proven to be valuable for design assessment through all stages of development.

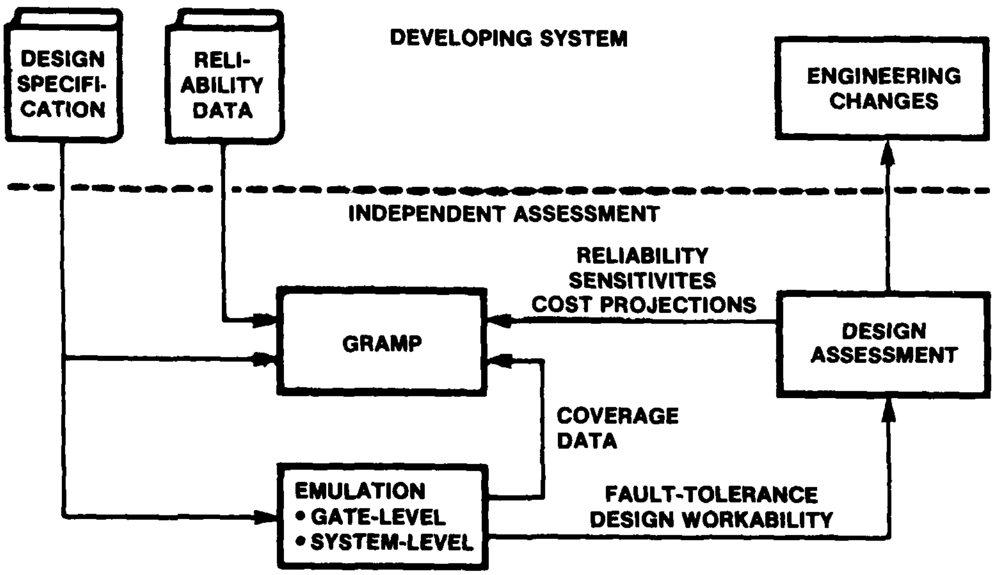


Figure 8 Relationship of Early Design Validation Tools

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REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document
	AGARD-CP-314	ISBN 92-835-0309-0	UNCLASSIFIED

5. Originator	Advisory Group for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France
6. Title	GUIDANCE AND CONTROL TECHNOLOGY FOR HIGHLY INTEGRATED SYSTEMS
7. Presented at	the 33rd Guidance and Control Panel Symposium held at the Agios-Andreas Air Force Base, Marathon, Athens, Greece on 13–16 October 1981.

8. Author(s)/Editor(s)	9. Date
Various	February 1982
10. Author's/Editor's Address	**11. Pages**
Various	184

12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.

13. Keywords/Descriptors

Integrated weapon systems
Autonomous weapon systems
Target detection
Integrated flight fire control

14. Abstract

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A Keynote address on Operational needs in the European Theatre and 22 papers were presented; papers presented covered the following topics:

- Operational Requirements
- Cooperative/Interdependent System Considerations
- Threat and Target Detection and Identification
- Autonomous Integrated Weapons Systems
- Affordability and Survivability Considerations.

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