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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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AGARD LECTURE SERIES No. 101

# Guidance and Control for Tactical Guided Weapons with Emphasis on Simulation and Testing



NORTH ATLANTIC TREATY ORGANIZATION

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# GUIDANCE AND CONTROL FOR TACTICAL GUIDED WEAPONS WITH EMPHASIS ON SIMULATION AND TESTING.

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## PREFACE

This Lecture Series No.101 is sponsored by the Guidance and Control Panel of AGARD and implemented by the Consultant and Exchange Programme.

As a result of significant and extensive developments in modern control theory in recent years there is a need to keep under continuous review their possible impact upon the design of tactical guided weapons. It is the purpose of this Lecture Series, therefore, to summarize the state-of-the-art of guidance and control for tactical weapons and to pay particular attention to GW simulation techniques (digital, hardware-in-the-loop development, validation) and the testing of missile guidance and control systems.

The other principal subject areas to be reviewed are weapon delivery (including targeting and acquisition), missile control techniques, and current guidance techniques (both mid-course and terminal, guidance sensors, and processing). Finally, consideration will be given to future trends.

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## CONTENTS

## INTRODUCTION

Tactical air launched missile development continues to be accomplished at a very rapid rate. The field includes a broad set of technologies in which frequent synergistic bursts of new capability occur. Some of these synergistic developments in missile guidance and control which are underway will be discussed during this series. One of the more significant of these synergistic developments is the impact of microminiaturization of integrated circuits combined with expanded state variable theory and new developments in guidance sensors and information processing. These innovations are making it possible to accomplish measurements or estimations of aerial target behavior and resulting determinations of most desirable trajectory modifications of the attack missile on a real time basis. New navigation laws are also possible and are being developed in support of this enhanced capability. Lower cost and higher quality midcourse and terminal guidance systems are being developed as well.

Some of you may have participated in the first lecture series on tactical missiles given by AGARD back in 1972. You may recall that that series emphasized techniques of development of subsystem requirements for control components and guidance devices. Two sessions, however, were devoted to evaluation methodology and simulation.

In the intervening years since the first tactical missile series, significant advances have occurred in all of the missile technical fields. Among the more exciting are those associated with missile navigation, guidance, and simulation. The discussions in this series will emphasize these areas.

In developing our review, we will first consider in general terms the rationale for a general analysis of the needs for guided weapons and concurrent needs for simulations. We will then match these discussions with reviews of some emerging technologies and simulation methodologies.

Let us begin by outlining the issues which drive guided weapon technologies. Several of the basic goals of tactical warfare relate to these issues. The most obvious of these goals is the desire to accomplish an effective strike against an enemy with a minimum cost of resources. Of equal importance, however, is the ability to have an effective surge capability without maintaining inordinately large or costly forces. Achievement of both of these goals in the modern environment drive the development of guided weapons.

As threat capability becomes more significant in both defensive and offensive modes, the cost to accomplish a strike and the cost to defend ones own resources increases to the point that significantly improved weapon systems become imperative. This is precisely why there is currently so much interest in precision guided tactical weapons. With this in mind, let us consider the issue of the rising cost of strike accomplishment. The netted air defense concepts common today permit defense commanders to detect strike forces through early warning radar and then be able to intercept the penetrating forces numerous times with various gun and missile defense units along routes to deep targets as well as at the targets themselves. Enroute attrition in many cases therefore may be severe. As terminal defenses tend to be more intense and because the task of the strike crews becomes more demanding, the expected attrition is even higher during the actual strike phase. This higher level of attrition forces tactical air warfare planners to develop minimum exposure tactics, and this in-turn generates the need for strike weapons which require a minimum of attention from the aircraft command and which are sufficiently accurate and powerful so that a single pass will be sufficient for successful mission accomplishment. In response to this situation, weapons have, over the past decade evolved from unguided bombs to laser guided bombs to television and infrared guided air-to-ground missiles. Moreover, the weaponry evolution has extended into longer range systems and off-boresight capability weapons.

Consider also the evolution of need for improved capability of air-to-air weapons. As the speed and general performance of combat aircraft have increased, the time during a given engagement in which a pilot may fire weapons at aerial combat opponents has significantly decreased. This shortened reaction time has in-turn led to a tactical weapon family of "beyond visual range" and "within visual range" missiles requiring a minimum activation time. Moreover, it has also led to the need for launch - leave - weapon capability.

Most, if not all, of the tactical air-to-air missiles currently in use utilize some form of infrared seekers or semiactive radar seekers. The concepts under development incorporate incremental improvements such as better terminal guidance as well as some radical improvements as for example dual mode guidance.

During this lecture series we shall discuss the evolution of weapon guidance and control technology to date for both air-to-air and air-to-ground applications and will pay particular attention to the role played by simulation and testing.

We will set the stage by highlighting the tactical problems which depend to some degree on guidance and control technology for resolution. This will include a general discussion of environmental factors which drive the needs for guided weapons. We will review the evolution of guidance laws and the current new developments which are making innovative guidance laws possible. We will discuss the benefit potential of new control concepts.

The series also includes a comprehensive review of two and three point guidance laws. We will address modern and optimal control theory applicability to guidance subsystems and will include several indepth discussions of simulations and testing.

Throughout the series active participation by the audience is encouraged. Questions and comments to each speaker are encouraged – both at the time of the particular lecture and during the round table discussion.

C.T.MANEY
Lecture Series Director

# TACTICAL MISSILE PERFORMANCE REQUIREMENTS

## A METHODOLOGY FOR DEVELOPMENT

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SUMMARY

This paper addresses the philosophy by which tactical air launched missile design requirements are developed using the development planning process used by the United States Air Force. Basically, this process consists of an analysis of the mission to be accomplished including the countering influence of the threat, a comparison of the alternative approaches to the solution, and a trade off design analysis of subsystem and system performance, effectiveness and costs. Typical generic factors of speed, maneuverability, range, and payload are discussed.

INTRODUCTION

The decision to develop a new tactical missile is based upon corporate assessments of the relative significance of planning factors associated with the threat, present inventory level, inventory weapon performance, potential for performance improvements, development and life cycle comparative costs, and relative need compared to other needs. A thorough examination of each of these factors, though useful, is considerably beyond the scope of this paper. Attention is limited in this discussion to those technical factors which impact strongly on that portion of the development planning process which leads to the determination of the specific missile performance parameters. A further limitation of this discussion is that though much selection methodology commonality exists amongst all types of missiles, this paper considers only air launched conventional missiles.

As is the case of all new Department of Defense system development programs, the initial formal activity commences with the publication of a Mission Element Needs Statement or MENS. The purpose of this document is to formally describe the corporate rationale for program initiation. Development of the required arguments to be included in the MENS demands considerable indepth thought, analysis and data gathering. This paper addresses much of the type of work which is accomplished in support of this formal process and further which supports subsequent preliminary design efforts. In brief, we shall consider methodologies for answering questions of two classes: those associated with the requirements for the weapon system and those associated with the design of the specific weapon.

GENERAL WEAPON REQUIREMENTS

Let us first consider the development of rationale for a weapon system. The perceived need for a new system frequently originates with combat personnel. Comments or complaints about observed inadequacies stimulate discussions within the using command. Then during periodic planning meetings between operating personnel, headquarters specialists and development personnel, the perceived deficiencies crystallize into a need for a systems study. The study can take the form of a concept formulation or a mission analysis. The concept formulation study assumes that a need for a particular capability has been identified. The study then develops specific rationale and trade off analyses which finally result in technical descriptions of promising weapon concepts along with parametric designs and estimates of the performance, costs, and effectiveness of alternative concepts. A mission analysis, on the other hand, examines a total mission area and sometimes produces a wide variety of system concept needs. For example, sometime ago, the Air Force conducted a study on defense suppression. The results of the study illuminated the need for improved command and control reaction time, improved reconnaissance, new aircraft self-defense weapons and an assortment of strike weapons.

After extensive review and consideration of the mission analysis and other related factors, one or more specific concept formulation studies are usually initiated. If the results of these studies are promising, specific concept definitions or preliminary designs are developed. This three step process, with extensive participation and review, serves to surface all of the significant technical facets of the perceived technical deficiency, the threat ramifications, alternative approaches to the problem, and preliminary estimates of cost, performance, and system effectiveness.

It is of interest to note the methodology used in the mission analysis portion of the process. The usual approach is to utilize an ad hoc team to develop an overall study approach and identify (or develop) the appropriate threat model. They will then develop particular tools called for by the mission and the selected approach, such as a penetration model and associated supporting models. Another portion of the team is tasked to develop input data for the models and a third section of the team is made responsible for developing conceptual designs for systems to be considered in the problem. At this point, it is perhaps useful to digress briefly to comment on efficient study management.

Once the study leadership determines the definition of the problem and develops an overall approach, the scope of investigation becomes critical. Judgments at this period readily identify a skillful manager for the decisions on depth and scope of the analyses are made at this stage. Much time--even months--can be wasted if the study director chooses an unnecessarily grandiose approach to the modeling problem. A simplistic approach is almost always preferred as it is usually the best understood by all concerned. This approach, moreover, has the advantage of requiring less time to execute (less data detail required and less sophisticated computer model development and shorter computational times). The major disadvantage to this; however, is that it is bound to be very difficult to know in advance of the analysis which factors will prove to have the predominant influence on the results and thus to be able to know which of many possible simplifying assumptions may be made without loss of either usefulness or validity.

Regardless of the degree of sophistication employed; however, the performance model is usually a hierarchy of models. Typically, a set of reasonable scenarios are postulated in order to gain a preliminary estimate of threat potential. Opposing force doctrine, size, and equipment are postulated and red versus blue actions and reactions are developed. The level of sophistication of the selected war game model serves to help bound the problem.

Once again let us refer to the study of defense suppression as an example. In this study, a specific section of land in Central Europe was chosen as an area of interest. Representative targets were identified and reasonable defense levels at the targets and enroutes to the targets were postulated. Then, with models of threat defense, early warning, command and control, electronic countermeasures, and air and ground defense systems, it became a straightforward process to determine exposure time and probability of encounter for airborne penetrators in a parametric fashion.

If one couples to these models the parameterized postulated performance for defensive radar, electronic countermeasures, and lethal defense systems, it is possible to derive reasonable estimates on the relative importance of penetration speed, altitude, in-flight maneuvering, radar cross-section, vulnerability, flight size, etc. This phase of the study develops very illuminating data on the value of near real time reconnaissance against mobile defense units. It also develops, at the same time, useful data on the payoff for various levels of target damage assessment.

Depending upon the desired level of detail required and on the availability of intelligence data concerning threat defense systems, penetrator survivability data may be developed either from a deterministic or Monte Carlo point of view. Both approaches are frequently employed. If it is desired to develop deterministic data, better estimates of penetrator survivability are potentially possible but this approach require development of credible assessments of threat defensive detection, command and control, and particularly in defense weapon fly out and end game performance.

The goal of penetration is the survival while accomplishing successful strikes against critical targets. To this point we have merely discussed penetrator survivability. Target destruction, however, is also a function of warhead numbers, size, type, penetration, fuzing, accuracy, reliability and delivery conditions. These factors must be superimposed on the penetrator survivability data.

It is reasonable to assume a sufficiently wide range of penetrator parameters so that the penetration air vehicle may be typical or manned aircraft, remotely piloted vehicles, or standoff missiles. As the baseline case, a strike system using manned aircraft may be considered. Flight profiles may be developed for predicted optimum survivability enroute to the target area. Once the aircraft is in the target area, however, the analyst should address the question of target acquisition and required strike flight profile (e.g., low and fast penetration with terminal pop-up and delivery). Reasonably good data is usually available for use in models for target acquisition and weapon accuracies for delivery of conventional unguided bombs. Somewhat less data are available, however, for the case where the pilot launches inventory guided weapons. For the case of lock-on-after launch type guided weapon systems, of course, even less experimental data are available for use in model validation and use.

The case of manned aircraft and direct attack weapon strike systems requires that the analyst address survivability of the weapon as well as that of the strike aircraft. This, of course, influences the overall probability of target kill and cost of target kill estimates. In completing the baseline case, estimates are developed for weapon probability of success as a function of launch conditions and target geometry. This is usually done using broad approximations; however, point mass simulations may be employed to permit calculation of reasonable estimates of encounter probabilities.

After development of the baseline case for penetration and strike survivability, subsidiary cases of standoff weapons and remotely piloted vehicles with weapons may be studied. From an analysis point of view, this may be accomplished by assuming flight profiles, observable cross-sections, vulnerabilities and accuracies appropriate to those particular vehicles.

In general, the conclusions reached may be summarized as follows:

For the manned aircraft with unguided weapons:

Good target acquisition.

Fair target destruction.

Excellent bomb damage assessment.

Low cost of sortie except in high threat areas.

High cost of mission due to need for repeated strikes.

For the manned aircraft with guided weapons:

Excellent target acquisition.

Excellent target destruction.

Excellent bomb damage assessment.

Low cost of sortie except in high threat area.

Low cost of mission except in high threat area.

For the standoff missile:

Excellent target acquisition for fixed targets in good weather.

Excellent target destruction.

Poor bomb damage assessment.

Moderate cost of mission in good weather.

High cost of mission in poor weather.

For the strike RPV:

Excellent target acquisition for fixed targets in good weather.

Excellent target destruction.

Fair bomb damage assessment.

Moderate cost of mission in good weather.

High cost of mission in poor weather.

It is obvious that these factors, though influencing, are not sufficiently inclusive. Many important factors are simply not amenable to modeling--for example--credible assumptions on the effect of possible jamming of data links. Moreover, other factors such as surge capability, on the other hand, frequently can be assessed only in larger war game models. Human and environmental factors can frequently be handled best in a subjective manner. It suffices to say, therefore, that a mission analysis study of this sort is useful in bounding the weapon system design problem.

At the completion point of a mission analysis, one can have a rather good first cut of desired weapon characteristics required for all three delivery modes. These characteristics include speed, range, flight profiles, radar cross section, optical cross section, need for maneuverability, required levels of midcourse and terminal guidance accuracy and warhead size and type for RPVs, standoff missiles and direct attack weapons.

SPECIFIC WEAPON REQUIREMENTS

An analysis of the type just discussed could lead to a number of valid recommendations. For the sake of discussion, let us assume that the analysis supported the development of a standoff missile. Let us then review the methodology by which the preliminary design requirements for this concept may he developed.

The previous study probably would have addressed the warhead size and type required to destroy representative targets under optimum delivery and accuracy conditions. The issues which should now be addressed indepth are the tradeoffs concerning midcourse and terminal guidance technologies and the desired range and flight profile. Point mass three degree of freedom simulations are probably no longer adequate at this stage. The designer, therefore, begins to look at five or six degree of freedom models in order to integrate the overall weapon performance predictions.

For a tactical standoff weapon useful in deep interdiction, for example, three basic flight profiles would probably be considered: a subsonic cruise, a supersonic cruise and a boost glide. Coupled with these vehicle concepts would be guidance concepts for midcourse and terminal phases. The designer then must consider the basic vehicle to be a turbojet, ramjet, or solid rocket. These basic concepts would be expanded to address size and type of payload, aircraft launch weight and size limitations, and reasonable ranges of flight profiles--range, speed, and associated altitude. Using appropriate simulations, the designer then develops configurations which are compatible with the assigned design constraints.

The baseline concepts become the basis for a series of parametric design changes and sensitivity analyses. One of the initial study phase parametric investigations is the tradeoff of range, speed, and warhead weight. While these design variations are being developed by the airframe, aerodynamic and propulsion engineers, another series of tradeoffs are accomplished in the weapon lethality area. This portion of the study addresses variations in expected target destruction as a function of warhead size and type and delivery accuracy. Required inputs to this substudy, of course, include target vulnerability data, assumed aim points, fuzing types, and weapon orientation at impact.

Design and performance estimates to this point provide the preliminary basis for serious cost effectiveness studies. Life cycle cost effectiveness evaluations form the primary rationale for technical selection criteria. Factors which are considered from this point on are part of the complete weapon system, and the overall considerations in cost to destroy a target and cost to maintain in peace time inventory.

Before we comment further on the evolution of missile performance requirements, let us review the impact of our choosing as a decision tool the system life cycle cost effectiveness rank. Perhaps the greatest weakness of the overall life cycle cost effectiveness approach is that it assumes that dependable data are available for the exercising of the models. It is precisely because dependable information is not always available and available data are often not dependable that the weapons anslyst should in general limit himself to making summaries of his analysis and be extremely reluctant to make recommendations. Recommendations should be based on objective evaluation of all the influencing factors and the weapons analyst is usually in a poor position to state objective evaluations of his own assumptions. The analyst can; however, be of great service to the decision makers if he prepares confidence or validity statements regarding all of the data used in his analysis.

Though the life cycle cost effectiveness measure of merit suffers from the weakness of valid data, it is still an excellent tool. For a particular example we are considering the design requirements for a standoff missile--the basic questions briefly mentioned earlier should be readdressed and expanded. Namely:

Are the targets under consideration worth the effort to destroy them?

What are reasonable alternative ways to accomplish the task?

What are the probabilities of success of each of the alternatives?

What are the corollary requirements of each of the alternative approaches in:

Reconnaissance requirements?

Command and control?

Aircraft compatibility?

Personnel?

NATO interoperability?

Maintainability?

Reliability?

Operating costs?

Etc.

and particularly:

What are the relative costs in time, physical resources, and money for each of the alternatives for accomplishing the task?

Let us return now the evolution of specific missile performance requirements. Our analysis has progressed through baseline concepts and with suitable models and model input assumptions, we may make predictions of the probability of given concepts being able to fly to the target area and to destroy the target.

Cost to destroy the target will include the costs of the system plus the costs impacting as a result of assumed levels of attrition for the carrier vehicle and for the strike vehicle. At this point it is very useful to consider the cost impact of modifying the baseline designs to permit higher velocities, lower altitudes, inflight maneuvers, changes in radar, optical, and audible characteristics, etc. As each change is assumed, the system performance models should be utilized to predict changes in survivability and overall effectiveness. Once again it is emphasized that the analyst should exercise extreme caution and assure himself that he is presenting performance, cost and effectiveness data to a level of accuracy that is fully supported by facts. Moreover, if the facts are not sufficiently plentiful, he should alert the decision maker as to just where the analysis deficiencies lie.

At this point in the development we have a series of very preliminary conceptual designs along with some broad estimates of relative "costs to do the job." We have rough cuts of desired accuracy, payload, observables, flight profile, range, maneuverability, aircraft load-out and cost to say a factor of two or three. It is prudent at this stage to take a close look at the "illities" of the concepts. The desired system characteristics permit a preliminary selection of subsystems--midcourse guidance, terminal guidance, warhead, fuze, propulsion, etc. Each subsystem candidate then in-turn is studied from the point of view of performance potential, state-of-the-art, reliability, maintainability, manufacturability and overall cost. Also addressed is the relative vulnerability of each subsystem to various countermeasures. Candidate concepts may be ranked at this point on the basis of costs, performance, and effectiveness. Moreover, representative subsystems are usually identified at this stage to provide credence to the postulated rankings.

For the United States Air Force, the developments thus far described are frequently the result of combined industry and government studies. These study results are used primarily to assist in the Air Force activity preparation of the mission element needs statement discussed earlier. This document is then reviewed at the highest levels of Air Force and the Defense Department. If the consensus of the decision makers is that of a go-ahead, the Air Force frequently asks the aerospace industry to develop the next level of specifications.

The data thus far generated serves as the basis for a formal and funded concept definition on preliminary design studies, the goal of which is to provide data somewhat more refined than before and also with independent industrial assessments. Frequently during this phase as many as four separate systems contractors will participate.

Using the data and study results developed by the system concept definition contractors, Air Force then decides whether or not to proceed into a weapon prototype development. If the decision is affirmative, several firms will compete to build operating prototypes. Incidentally, the competition for selection of these contractors is normally not limited to those firms who participated in the earlier studies.

This stage of the determination of subsystem performance usually demands an extremely large amount of experimental data; measured seeker detection capability and tracking rates--hard wind tunnel data for various aerodynamic shapes and control concepts--test performance data for proposed signal processing devices are representative of the required activity for each contractor. The performances achieved by these flyable systems will then determine the specific weapon performance goals or specifications required.

Development of pilot or limited production runs for the weapon concept are based on design specifications developed in the previous phases. Additional changes of course will occur during engineering development as a result of new ideas, manufacturing difficulties or improvements, test results, etc., but the basic design specifications are established.

## TECHNOLOGY TRANSFER

One of the more significant challenges in evolving the performance specifications is that of determination of the most appropriate state-of-the-art for each of the missile subsystems. There are three major aspects to this problem. First, is the determination of the state-of-the-art of the candidate technologies; second, is the selection of the most promising candidates, and third, is the particular mechanics for accomplishing the transfer insofar as the first problem is concerned. The Air Force makes a concerted effort through the several Air Force laboratories and development organizations to maintain communication with both domestic and allied country aerospace contractors.

One of the primary means of staying abreast of United States contractor developments is through the independent research and development (IRAD) program. Defense contractors are permitted to set aside a percentage of the value of each contract (included in overhead) to finance defense related research and technology development. In return, each contractor doing independent research on new technologies under this program reports periodically to the Defense Department. By means of on-site reviews and report reviews, the Air Force maintains cognizance of progress being made in these pertinent technologies.

The weapons related technologies under development outside the United States are also reviewed and examined for applicability to new missile development. The International Systems Technology Evaluation Program (INSTEP) of the Air Force utilizes the facilities of the European Office of Aerospace Research and Development (EOARD) in London to invite European research organizations to communicate their activities to interested USAF agencies. The INSTEP office also makes use of the Office of Defense Cooperation (formerly Military Assistance Group) in each United States Embassy to facilitate the cooperative exchange of technological progress and needs. NOTE: INSTEP is presently in the process of becoming a triservice support function and may eventually be operated at the Office of the Under-Secretary of Defense, Research and Engineering.

The second aspect of the technology transfer problem is the selection of the particular technology for use in the several subsystems. This is done by means of an extremely thorough scanning and review process by teams of technical experts supported occasionally by outside consultants. The factors considered are performance, reliability, maintainability, cost, availability, compatibility, etc.

The third portion of the problem, i.e., mechanics of technology transfer is not a uniquely defined procedure. The techniques for its accomplishment are widely varied throughout the community. The approach used at ADTC is that of transferring certain key people along with the project as it moves from exploratory research to advanced development and finally to engineering development.

SUMMARY

The development of tactical missile system performance is a step by step procedure which places very heavy emphasis on the front end of the program. Extensive analysis is conducted concerning the merits of relative ways of doing the job of destroying a tactical target as well as whether or not the target should even be destroyed in view of known heavy demand for limited Air Force resources for other tasks.

A critical problem involved in the development of missile performance specifications and especially in the development of the missile hardware is that of technology transfer. This transfer is vital to the determination of the best compromise of performance, cost, and reliability for credible specifications. It is absolutely indispensable in the satisfactory implementation of these specifications. The missile design specifications which evolve as a result of this planning process are based on proven technology and reasonable design-to-cost goals. Heavy emphasis is placed on not exceeding state of the proven art so as to assure acceptable reliability, maintainability, manufacturability, and interoperability.

# WEAPON DELIVERY AND ITS EVALUATION

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## 1 INTRODUCTION

The object of this first paper in the lecture series is to set the scene in which the guided missile and its associated weapon delivery system must perform effectively if it is to offer a real capability, and then to follow this discussion by presenting some of the modelling and simulation techniques that enable the weapon with its delivery system to be assessed.

The fundamental problems of successfully delivering tactical weapons, be they guided or unguided, from air vehicles, stems from the increasingly hostile environment that faces the aircraft and hence the tactics that have to be employed to ensure an acceptable level of attrition. Traditionally aircraft have been able to adopt operating tactics that assist weapon delivery accuracy; *ie* dive attacks, stable flight prior to weapon release, and a good surveillance of terrain aiding early target detection. Today, it is essential to establish an operating concept that ensures a high survivability of the aircraft and, within those operating constraints, to design a weapon plus system that can function effectively and wherever possible use to advantage the constraints of the attack. However, in general many of the features that offer protection (leaving apart ECM) also make the engagement conditions more difficult, for example, screening ranges, the oblique geometry, etc. Ground to air detection probabilities are reduced by flying as low as possible, at speeds in excess of $M=0.8$ and by using small vehicles with low signatures. The constraints that apply to operations under such conditions are discussed later.

The systems and missiles must therefore offer the chance of being deployed from small high speed low-flying aircraft, and support for the low level attack in heavily defended air space is now growing rapidly. In the case where line of sight contact to the target is required, terrain masking and weapon launch limitations normally determine the time that the pilot has at his disposal to complete all phases of the attack. In general this is likely to be less than 10 seconds. Under such rigorous engagement conditions, the reaction time of the weapon delivery system has to be capable of 'out smarting' that required by the local target defences, if a satisfactory kill rate and exchange ratio are to be achieved. Key features in determining this overall success rate are therefore the constraints that bound the problem: target detection/recognition, precise sighting systems that minimise the required time for aiming/designation, wide missile coverage envelopes that enable weapons to be deployed over a range of engagement geometries, terrain masking, which to some extent is under the control of the attacking vehicle, atmospheric characteristics, and finally limitations imposed by the weapon on the flight profile.

The following sections aim to address many of these problems and outline the way in which the assessment and testing of weapon systems can be conducted with these constraints uppermost in determining effectiveness of specific weapon system options.

## 2 OPERATIONS

Before proceeding with a detailed discussion of weapons and their delivery systems, it is valuable to set the stage by exploring the use of the tactical weapon from the air, the types of mission on which it may be valuable, and the types of target against which it can be deployed.

If air power is to be used effectively, its three prime characteristics of flexibility, mobility, and fire power have all to be exploited. The aircraft itself is inherently flexible and mobile providing a number of roles against different targets at short notice. The tactical weapon system with its high lethality, precision aim, and compactness, complement these aircraft characteristics, and serve to add even greater flexibility to the delivery vehicle in the final attack phases. Tactical operations themselves cover a wide range of activities but for the purpose of this paper three classes of mission can be identified, covering counter air, air interdiction, and close air support.

In a limited engagement against an enemy with an effective air force, the main tactical activity in the early stages will be counter air measures designed to gain a favourable air situation. This is likely to involve attacks against:

(i) enemy air fields, missile complexes and immediate support facilities,

(ii) surface to air defences system,

(iii) air vehicles in general

in fact, any target which supports the enemy's counter air objectives. Targeting for such air strikes should be of a planned nature based upon the latest intelligence concerning the enemy battle order. Targets would be assigned a priority and great benefit accrues from a precise knowledge of target position in an absolute co-ordinate frame of reference. Additionally, precise co-ordination between different aircraft is beneficial in order to pose a multi-threat to the defences. Also if the target does not possess a prominent signature in either visual, radar, or electro-optical bands, then a position relative to a local prominent feature in one of these wavebands becomes highly desirable. The air to air aspects of counter air will not be discussed in this paper.

The second group of missions identified above are interdiction operations, which are normally conducted some distance behind the enemy lines. Attacks may be aimed at typical targets such as armour reinforcements being brought up to the forward battle area, road and rail transport, communication centres, and logistic and supply routes in general. To be effective this type of operation has to be maintained on a 24-hour basis. Targets are likely to be seen in groups and probably on the move, but will not in general be of known geographic position. Cueing however may be forthcoming from local features such as roads, rail tracks, rivers etc. Timing of such attacks may also be an important element of this type of operation. The effects of air interdiction are achieved through destruction, neutralisation, delay and of course harassment. In particular they reduce the enemy's capability to mount or sustain an offensive, restrict his freedom of action, and generally increase his overall vulnerability to be attacked. When targets are found, however, they are likely to be associated with local defences and the effects of search procedures, stand-off range, etc are important elements in this type of engagement.

Finally, there is the close air support operation which involves action against enemy targets in close proximity to our own forces. The targets are therefore of immediate concern to the surface forces and the use of air power is envisaged only when the surface forces cannot produce the desired effects or the disposition of the targets prevent their successful attack from the ground. The use of air power can often give quick results and raises the morale of the ground forces. The operations must be integrated closely with the land forces and hence pose special command and control problems. It is usual for target area control to be through a Forward Air Controller and aids such as laser target marking, either directly on the target or indirectly, become a valuable asset in establishing visual contact with the desired target. Typical targets are tank groups, troop concentrations, surface to air missile systems, radar controlled gun systems, and various fighting vehicles. Target identification in this highly mobile and cluttered environment is important. Problems associated with debris, smoke, destroyed targets etc, all serve to compound the difficulty of the decision making process.

In general therefore the tactical operation is often against the smaller mobile target, difficult to plan in precise detail, but likely to be against a target rich scenario. The small size, precision and mobility of the tactical guided weapon can therefore reap good dividends under these conditions, for the launching vehicle can carry a large number of weapons and retain considerable agility and flexibility throughout the whole attack phase. This latter point impinges significantly on achieving a high survivability in the hostile environment.

## 3 CONSTRAINTS OF THE SCENARIO

In order to assess, simulate or test any weapon system option, it is extremely valuable to quantify the limiting conditions within the scenario that impinge upon the system during the delivery phase. This is always a complex area and one often open to intense debate. The aim of this paragraph is to identify many of the key effects and to indicate the type of data bank that is required in order to initiate a valid assessment of particular options, typical examples of which will be presented later in this paper.

### 3.1 Terrain

At low level the terrain is paramount, and to a large extent the pilot's instinct for survival will determine how low he actually flies. The general terrain features, however, do provide important cover and the intelligent well trained pilot maximises the benefits that can be obtained from the local geographical effects. Considerable data exists to describe the general coverage that can be achieved from terrain and radial plots such as shown in Fig 1a-c, serve to illustrate the increased cover that is obtained when making attacks at the lower altitudes and by choosing the approach path with care. The alternative way to present such results is shown in Fig 2, and represents the probability of achieving a specific unmasking of the target as a function of approach range. Clearly different curves exist for each approach height, and the resulting family of curves will be terrain, target location, and approach path dependent.

Much of this type of data has been derived from measurements made from ground sites and does not precisely represent the dynamic conditions experienced from the air, where details and location of small objects can become important especially for the guided weapon delivery. This detail must embrace information that relates to local obstacles such as trees, bushes, hedgerows, local buildings etc, and their relative statistical distribution on the ground in and around the target area.

Continuous line of sight to the target is necessary during any extended detection process. Local features interrupting this sightline are therefore clearly undesirable, but usually only prolong the period of time that is needed to establish the correctness of specific targets. However, once the guidance line of sight has been established and the weapon is committed, then the interruptions to this sightline usually prove disastrous unless special precautions are taken within the weapon system. The statistics of the fine screening detail thereby become important. Currently much interest exists in establishing a broad based data bank to describe the terrain statistics for a number of representative scenarios. The need or otherwise for short term memories in either the delivery systems or missiles, or both, is greatly affected by information of this type. By way of an example to illustrate some of the effects, Fig 3 shows a number of forward looking scenes recorded at 250 ft during recent trials in Northern Germany. The photographs were taken at 1 s intervals, whilst flying at 450 kn, in winter. Note the very significant effect of the trees despite the lack of foliage.

### 3.2 Atmosphere

In establishing a complete all-year round capability, the atmosphere plays a key role in determining what can be seen when, at what contrast, and on how many occasions. Classically, the weather scenarios are taken from climatic data, which it is argued, presents a picture that is close to the expected value for each parameter. In practice, of course, meteorological range is not distributed in a Gaussian manner, as illustrated in Fig 4. This shows data collected from the Hannover region of Germany indicating the frequency of occurrence of conditions when the visibility exceeds the indicated value. Note 5 km range is only achieved on 30% of occasions.

In evaluating the performance of electro-optical viewing systems a more complex combination of environmental parameters becomes critical. In fact, as in the case of visibility discussed above, it is the occurrence of wide range departures from the mean of the critical parameters that more often than not determines the limiting conditions of operation. Detailed atmospheric modelling now becomes essential in order to establish for example the performance of thermal sensors of heavy fog, rain, etc. This work has been reported in some detail[1,2], where weather statistics gathered at one hourly intervals throughout the year are presented. These observations recorded parameters such as dew point, temperature, visibility, and cloud and wind, and have enabled atmospheric models to be upgraded and validated with some certainty in the appropriate wavebands.

### 3.3 Cloud coverage

Cloud cover is usually only considered a major problem when weapon delivery takes place from altitudes in excess of 1000 ft. However, as can be seen from Fig 4 some difficulties can be experienced in the number of occasions on which cloud obliterates the ground line of sight at the lower altitudes. This problem is especially serious during winter month operations, and as can be seen something in excess of 10% occasions will be lost in this period when operating from 200 ft. Under some conditions, clouds can also increase the level of difficulty in target search because of the additional clutter that cloud shadows introduce into the overall scene, and the wide variation that then occurs in scene contrast levels.

Additionally, cloud coverage affects the thermal scene by providing equilibrium between sky and ground with the result that fine image detail will become washed out. This can be considered a double-edged effect since although the image detail essential for safe low level flight management may be lost, a general reduction in ground clutter will occur which will aid the search process as discussed below.

### 3.4 Target detection

Target search and detection are key parameters in establishing the 'starting line' for the delivery process, and much has been documented on likely detection ranges for various targets and groups of targets. Refs 3 and 4 give a comprehensive summary of the expected values for detection range over a wide range of scenario conditions.

The actual range at which detection occurs on any one attack, however, is a very complex issue, and one that has received a great deal of attention during the last decade or so. Numerous mathematical models have been produced for a variety of requirements and in the main these relate the contrast of the target against its background, with its angular size, with some search method, to specific detection criteria. Looking briefly at target contrast, Fig 6 illustrates the typical way in which the target contrast, with an initial inherent value of 30% relative to the background, decreases with increasing viewing range. The data, which is for visual conditions, is shown for clear and overcast conditions and for dark and light targets, when the meteorological range is 10 km. A number of detection criteria exist but time does not permit any detailed discussion in this paper. However, again for the visual case, the criteria in their simplest manifestation usually take the form of:-

$$k_2 \frac{d_o}{R_o}$$

where $d_o$ is a target size parameter and $k_2$ is a constant dependent upon the precise criteria being employed.

By relating these functions, values of $R_o$ can be obtained from unlimited viewing time conditions, for a range of targets and backgrounds and atmospheric conditions. For electro-optical viewing systems, the interactions between the crew, the image and its quality, and the target scene, are more complex and embrace the performance of the viewing system in total including the display and the environment, together with the man. To address this problem the concept of Minimum Resolvable Modulation (MRM) is now often used. The technique has wide application and enables the relationship between the spatial frequency of a stylised four bar target and the modulation necessary to distinguish that target, to be defined for any electro-optical system or part system. As an illustration, Fig 7 shows a set of hypothetical MRM curves for a television system. A series of curves exist defining the performance under different background conditions:- $B_1$ high light level, $B_5$ very low level of illumination. To recognise a target, say using the Johnson criteria, for any specific system, implies that frequencies '$f_r$' must be distinguished. The curves then indicate the modulation that the vehicle has to present against its background if it is to be seen, by means of the system. Atmospherics effects modify the expected levels of modulation.

The air environment of course does not permit the 'luxury' of long viewing times, for even with the helicopter operation, viewing time can be equated with exposure, and for the aircraft the vehicle rapidly closes on to the target area. To illustrate this and to indicate the large step that still exists in going from the laboratory to field trials, Fig 8 shows the interaction of target size and contrast for a number of situations. Curve A represents laboratory data relating to uncluttered scenes with unlimited search time. Curve B shows the same situation but with the search time reduced to around 1 s. Finally, curve C gives a plot representative of trials data, and suggests that larger target contrast or size is needed to effect detection in the real world conditions. Targets in reality are rarely detected in the strict laboratory sense but have to be perceived sufficiently well to differentiate them as objects in a cluttered scene.

For the purposes of weapon system modelling therefore, I believe, there is much to recommend a less rigorous approach that employs empirical relationships to overcome some of inherent difficulties discussed above. Considerable success has been obtained by relating the cumulative probability of detecting a target, and thereby the chance that detection will occur at a specific range, as a function of the major scenario parameters. For example:-

$$P_{acc} = f(n, R_o, R_s, R)$$

where $P_{acc}$ = cumulative probability of detection

$n$ = a weighting task difficulty factor

$R_o$ = the range at which detection would just occur given infinite viewing time

$R_s$ = the range at which the target becomes unscreened

$R$ = the variable range.

The general form of this relationship is shown in Fig 9. The range data has been normalised for a detection range of $R_o = 1$ for the benefits of illustration, and the curves plotted for values of $R_s = R_o$ and $R_s = 0.5R_o$, and $n = 0.25$ and $n = 1.0$. It is found that airborne trials data are well represented by this form over a wide range of targets and conditions.

The parameter 'n' remains the most difficult area to tie down and at present the spread illustrated represents the uncertainty. Additionally, the effects of target motion, specific groupings of targets, various disturbances of the scene caused by the target's presence, are all difficult effects to quantify, with little useful data existing to predict their effect on likely change in detection range.

### 3.5 Human factors

This important area follows naturally from the previous paragraph and is high-lighted because in the past the emphasis of system development has been on the problems related to technology rather than on optimising the performance of the human and the system. Task loading and the allocation of tasks among crew members during the critical attack phase needs careful addressing in system design and optimisation. An even more challenging area within the field of human factors, is the goal of achieving successful delivery of the more advanced guided weapon concepts from single seat aircraft, where the tasks of flight management, choice of operational tactics, and the final deployment of the weapon have all to be integrated into a single operator system. An essential feature of this total system is that it should retain a large degree of flexibility inherent in air systems. The application and use of automatic aids within a flexible robust weapon system is a key area for future design.

A related human factor topic is the level of degradation experienced by the crew, be it one man or two, as a function of the environmental conditions under which they are required to perform. Often, these are serious effects which are hard to quantify for data is usually scarce. By way of examples in this area, consider the aerodynamic ride quality given by the vehicle at low level, high speed, which proves especially crucial during all the phases of weapon delivery. A simple example of this effect is given in Fig 10, which shows a typical variation of aiming performance as a function of ride quality for a helmet mounted sighting system. Note a degradation factor in excess of three can occur.

Such a device is used for initial or coarse sightline determination and the accuracy and consistency with which this sight can be brought to bear on the target critically determines the ultimate performance of the weapon system or the performance requirements of any fine aiming device incorporated in the system or missile. Such considerations impose requirements on the aerodynamic properties of the vehicle. It is worth noting that aircraft optimised for the ground attack role, such as the RAF Jaguar provide an environment well to be left of the curve in Region A, whereas aircraft designed for the combat role having a low wing-loading tend to be to the right of the axis.

Performance against the time available to complete an aiming task is a further area where degradation can occur under operational conditions and where data in the past has again been scarce. This position is slowly being rectified and Fig 11 gives a typical example of the degradation that can occur in the pilot's ability to aim a head-up display under time limited conditions. It can be seen that compared with the unlimited time-aiming performance a serious degradation will occur when aiming times are reduced below 2 s. A good system design will ensure adequate performance under time-critical situations.

### 3.6 Navigation

Navigation accuracy is a key factor if successful low altitude engagements are to be undertaken. The system aids the pilot and crew in three key ways. Firstly, it prevents the crew from getting lost, especially under conditions where tactical considerations have dictated that a new route should be flown, for example where vulnerability could be reduced by a new approach path. It is interesting to reflect that in recent competitive exercises as many as 15% of the crews do not find the target.

Secondly, there is the problem of workload. Managing a navigation system that requires frequent manuel updating, monitoring, and keyboard insertions, imposes a high burden upon the crew. System developments to minimise the attention that the navigation system requires in flight, for example, automatic updating acceptance, represent significant improvements to system deaign.

Finally there is the crucial issue of precision cueing into the target area that a good navigation system can give in terms of a lead-in to planned targets. Uncertainties in defining the correct sightline to the target arise not only from system inaccuracies of course, but also in terms of prior knowledge of the target's geographical position in the applicable aircraft co-ordinate system.

Modelling techniques are very powerful in establishing the interaction of the statistical uncertainty in target position about the flight path, as the aircraft approaches the target with any specific weapon delivery system. This point is pursued further below. Also of considerable interest in this cueing role of the navigation system, is work relating to the pilot search process about the defined sightline, and the way in which uncertain target knowledge should be indicated to the pilot. Ref 5 discusses this point in some detail and concludes that it is essential to indicate to the pilot the likely uncertainty in target position when this can be determined. Computer defined sightlines that give such indications to the pilot are extremely compelling and it is found that targets displaced by only small angles from the defined sightline can be missed.

As mentioned above accurate navigation is only beneficial if accurate target knowledge is available to complement it. This raises severe problems for attacks against mobile targets and high-lights the need for a rapid reconnaissance capability if such techniques are to be used efficiently.

## 4 WEAPONS

Before addressing the problems of weapon delivery and the accuracy thereof it is valuable to consider the various types of weapon guidance that are appropriate to this area. The intention is not to prejudge the various concepts, but to identify fundamental issues that govern delivery.

Two broad classes of weapon are often identified; namely 'launch and leave' and 'fire and forget'. The essential difference separating these two techniques lies in the requirements of the weapon on the aircraft after the weapon has been launched. The former type requires the aircraft and system to be committed to a specific target up until the point of impact of the weapon, whereas in the case of the latter once the missile has been released, it has no further requirement of the aircraft.

Within these two classes a number of clear sub-groups emerge, each of which change the constraints for a specific operational attack and in general each type can either be powered or unpowered.

### 4.1 Launch and leave

The Launch and Leave concept is best known in terms of the laser seeking weapon that is autonomously aimed from the air vehicle. Guidance within the weapon is generated by a small quadrant detector mounted in the homing head which responds to the reflected laser radiation scattered from the target. The homing head aligns itself with the target and control signals are generated from the position of the head relative to the missile. Both powered and unpowered versions of this weapon concept have been considered and the potential of the unpowered system was graphically demonstrated during the Vietnam conflict. Many releases were made that resulted in pin-point accuracy and the general phrase of 'smart' as opposed to 'dumb' weapon was aptly coined. More recently interest has been centred around the low level release of 'smart' weapons to encompass the attack philosophy generally considered applicable to the European theatre. Guided bombs using a PAVEWAY kit enable autonomous low level toss attacks to be completed against large easily detectable targets with localised defences, whilst at the same time maintaining a measure of stand-off range. However, during the escape manoeuvre the aircraft reaches a considerable altitude and questions have to be raised concerning its ability to survive in some future scenario whilst undertaking this manoeuvre. Additionally line of sight problems as discussed in section 3 limit the general use of this weapon to a considerable extent. Although concepts embracing two aircraft or one aircraft together with a ground marker do enable a low level delivery mode to be made, it is unlikely that the weapon will gain acceptance for land operations until it evolves in an accelerated form. The problems of co-ordinating such attacks are not insignificant.

Much thought has been given to small powered missiles of this class and these offer a wide flexible launch envelope. Fig 12 shows such a typical coverage. The engagement envelope is wide and extends several kilometres in range ahead of the aircraft. Its short-range performance is good and satisfies the general need to provide a weapon system with a rapid reaction capability.

### 4.2 Fire and Forget

A number of weapon options fall within this general Fire and Forget class, which merit consideration. The first of these is the laser seeking weapon that has just been discussed when used in a homing mode, against targets that are externally illuminated. To the launching aircraft it becomes a fire and forget weapon delivery. Examples of this attack are forward air control marking or marking from other air vehicles, be they aircraft, helicopters or unmanned aircraft. The problems however of such constraints were discussed in section 4.1 and there is a greater interest in obtaining a truly autonomous capability from the weapon system.

Traditionally Fire and Forget has only embraced the truly self-seeking missile and used electro-optical guidance methods to home onto the target, *ie* using essentially passive techniques. However the use of inertial guidance is also representative of this class and under some operating circumstances offers attractions in terms of overall capability.

#### 4.2.1 Inertial weapon

Although these weapons exist in many forms, I intend to limit my remarks in this paper to essentially the short-range variants, by this I mean less than 10 km stand-off range.

The inertial unit provides three-dimensional navigation and is usually of the strap-down kind. Prior to launch from the aircraft, the unit is initialised with the co-ordinates of the target and the release velocity. During its flight which is usually less than 25 s, command signals are generated within the guidance unit and fed to conventional control surfaces to achieve the desired impact point. A lateral coverage of a typical unpowered weapon is shown in Fig 13. An initial along track forward throw is followed by an envelope that expands at a rate proportional to the 'g' capability of the weapon. Energy within the weapon is limited to that imparted at release and hence maximum range will be strongly dependent upon the launch velocity and the induced drag that is generated during the lateral manoeuvre of the vehicle. The coverage plot therefore assumes a roughly pear shape. By adding a power unit to the weapon, these limitations can largely be overcome. The value, though, depends on the specific engagement range that is needed and the ability to know the target position with sufficient precision at longer launch ranges. Where line of sight information is required, the value of the powered unit is questionable.

#### 4.2.2 Electro-optical guidance

The most commonly known fire and forget weapon relies upon electro-optical techniques for guidance. They permit accurate discrimination between small targets that are distributed in a cluttered background and a precision homing capability. Usually the homing head has to cope with low contrast situations, and therefore demands a high standard of tracking from their homing algorithms. This point is especially critical under European conditions, or conditions where a significant stand-off range is desired. This type of system is epitomised by the MAVERICK missile.

Choice of operating wavelength is important and although early systems employed the visible spectrum, a number of problems relating to target contrast, atmospheric attenuation, etc, have now caused most interest to centre around the use of thermal devices operating in the 3 to 5 and 8 to 13 micron wavebands.

A wide range of tracking devices have been tested and evaluated including techniques employing edge trackers, area correlators, centroid trackers, multi-edge trackers, and more recently, intelligent tracking algorithms to obtain a higher integrity in the rapidly changing scene that the missile homing head sees when approaching the target at low level. Developments in signal processing between the basic homing head and the tracking algorithm, now enable a better discriminating signal to be presented to the tracker.

The lateral coverage of such a missile is very similar to that shown for the semi-active laser guided weapon in Fig 12. The outer boundary, however, will be target shape and contrast dependent, since an adequate 'lock' has to be achieved prior to launch if an acceptable level of success is to be assured.

Rapid developments are taking place in this processing area and these are likely to result in a weapon of greater integrity that can be used against a wider range of targets. Increasing interest is also centring around exploiting the total spectrum that lies between the upper radar frequencies and the visual band in order to optimise the target signature and weapon intelligence, again to achieve high credibility tracking, and reduce the occasions when false target lock-on occurs.

4.3 Comparisons

Before proceeding it is valuable to identify some of the benefits and disadvantages offered by each option. Looking first at the benefits of the semi-active laser weapon, it is capable of good delivery accuracy that will be dependent upon the designation system. It operates over a relatively large range against a wide variety of targets, hence it is flexible, and it does not rely on any specific target characteristic other than it should reflect part of the incident illuminating radiation in a non-specular form. Valuable both by day and night its performance in poor weather is limited to essentially visual range if 1.06 micron lasers are the basis of designation. The guidance systems are relatively cheap, hence expendable with little cause for concern. The precise aim on to the target need only be accomplished a very short time before weapon impact, and hence, although desirable, there is no mandatory requirement to have a refined aim before the weapon is launched, provided the target has been positively identified.

On the debit side are limitations on the frequency of target engagement since the process is essentially sequential. For very long target flight weapons, some restriction occurs on vehicle manoeuvre before the weapon impacts the target, although it is considered that from low flying aircraft this limitation is minimal. The designator equipment imposes a penalty on the aircraft in terms of performance, (weight, drag, structural limitations which affect mission range, speed and operating envelope) and the number of stores that can be carried. Finally, although of well-defined nature, the system is active with the aircraft radiating energy on the target for a short time prior to the missile's impact. In addition of course two sightlines need to be clear during the missile's flight time, that is the sightline between target and aircraft for the designator and secondly the sightline between target and missile for homing. The problems therefore of atmospheric attenuation, smoke, dust, clouds etc impact the system on two fronts. It should be aoded however, that the inertial capability of the designating system to retain line of sight to a stationary ground target enables some of the transient difficulties to be overcome.

Turning now to the inertially guided fire and forget weapon, after weapon release the aircraft is totally free. The weapon is also passive and hence very difficult to counter-measure by direct interference with its guided system. However the initial aiming process is likely to be active unless operations are limited to pre-planned engagements only.

On the debit side it cannot possess the inherent precision accuracy of the homing weapon and clearly the accuracy capability has to be tailored closely to the requirement in terms of warhead capability and the flight time of the weapon after release. The cost can also be high and a carefully tailored performance requirement is essential. It is, however, highly adaptable and completely independent of target signature. It can be launched in salvo and does not suffer from sightline problems or difficulties arising from battle-smoke and debris.

And lastly the electro-optical option offers the usual fire and forget benefits, together with the possibility of a rather simpler aircraft fit. On the debit side are largely questions of confidence. The capability is limited by target signature, its luminance/radiance in relationship to the background, although the development of cheap fast image processing intelligence is helping to remove difficulties in this area. The day/night poor weather capability is similar to the semi-active laser guided weapon.

Obvious questions are posed concerning its susceptibility to relatively simple counter-measures, but any discussion of this topic is clearly outside the scope of this paper. And finally the cost of the weapon in the immediate future is likely to be greater than that of either the semi-active laser GW or the inertial guidance kit for short range weapons, and even though striking advances have been made in solid state technology especially in the areas of detectors and fast processors, it is difficult to imagine that the weapon will become a cheap option, and therefore the number of weapons within the inventory will be limited.

5 DELIVERY SYSTEMS

Airborne weapon systems have evolved at a rapid pace during the last decade. Prior to the mid-60s weapon delivery was generally a matter of deciding upon a standard set of delivery condition, *ie* speed, height, dive angle, and then flying the aircraft to achieve these specified conditions relative to the target. Little flexibility was available for operational judgement when in the target area, and against unfamiliar target scenarios large errors were likely. Limited corrections could usually be made for variables such as wind and target motion, but in general these were somewhat crude. The success of the attack depended totally upon the skill of the pilot achieving his desired flight conditions.

The advent of airborne sensors and computing power greatly aided these otherwise intractable problems by removing many of the hitherto constraints of the attack, and gave the pilot some degree of preferred operational choice in the delivery of dumb ordnance. The so-called nav/attack systems emerged and these covered delivery of a wide range of weapons over most operational flight profiles. The systems consisted of three key areas of technology to generate this enhanced capability, Fig 14.

Firstly there is the sensor area, designed to give the necessary flexibility of not demanding that the pilot shall achieve the necessary standard release conditions. In practice, this allows tactical considerations to rule. This area is vitally important for the low-level attack since very small errors in achieving the accurate release conditions can result in unacceptable miss distances occurring. The sensors can be divided into:

(i) ranging sensors/radar, laser, barometric height, radio altimeter, geometric, etc;

(ii) velocity sensors - inertial, air speed, Doppler;

(iii) attitude sensors - inertial, attitude/heading reference, gyros, etc.

Secondly there is the computing power essential to process the above measurements into useful, meaningful information for the crew. A steady increase in both the number of processors and the amount of processing has occurred over the last 10 years. A single computer with 8K words was at one time considered a large machine. Now, many processors, each with several tens of K words are considered average for the present day system. Much of this increase has gone towards my third key area, the system crew interface, where interesting developments in the areas of controls, display formats, moding options, etc have taken place. Typically ground attack aircraft may have head-up displays, helmet mounted sights, head-down displays, and other electronic display surfaces. In the area of input controls, multi-function keyboards, proportional hand controllers, and many other discrete input switching devices have to be set up for the mission. Additionally, devices that enable automatic data entry are becoming more common.

Such systems are admirable for the delivery of unguided weapons from both single and dual seat aircraft and depending upon the detail fit, provide either visual attacks with moderate precision or blind attacks using target information gained from the radar or navigation sub-systems with an accuracy compatible with those devices, against a wide range of targets. However, some important constraints are still imposed upon the launching vehicle. For although a number of unique release points can be achieved by flying different elevation flight profiles, a dedicated azimuth steering solution has to be achieved in order to register an accurate 'line' delivery. Additionally this range of elevation release points can only provide stand-off delivery at the expense of weapon accuracy. The need for guided weapons to enhance further the delivery flexibility is therefore apparent.

I would now like to turn to typical system make-ups that are appropriate to the guided weapon types as discussed above. The first important point is to recognise that as weapons progress and become more complex, then the interaction between the weapon and their system becomes more involved. No longer is it adequate for the system to possess an accurate ballistic model together with an interface that enables a single or series of release pulses to be generated. If the maximum capability of guided weapons are to be used to operational benefit then the performance of the aiming system has to be tailored to meet these needs.

The essential elements in the attack process are well-known; to detect the target, to establish the correctness of the target, to bring the system to bear on the target, to release the weapon, and finally to effect a successful escape by evading enemy defence systems. Detection was covered in section 3.4 where it was shown that essentially the process could be defined for any set of conditions in terms of a target range and a probability that a line of sight exists.

For the low level semi-active laser GW option, the essential additional system elements are a designator and tracking devices. Examples of such systems in current usage are PAVE SPIKE, PAVE TACK ATLIS, LATAR, etc. The designator is a high technology equipment and usually comprises the following sub-system: laser, electro-optical camera, common stabilisation and optics, tracking devices, and environmental control units. Although it is a self-contained piece of equipment many benefits arise from integrating its performance with the standard aircraft nav/attack system. Its principal performance characteristics enable it to be pointed to a very high accuracy over a wide ground coverage and to completely decouple the aircraft movement and vibration from sightline that is presented to the crew usually via the electro-optical link. This performance enables it to track targets in highly dynamic manoeuvres and to move rapidly from one target to the next.

The equipment can be used from both single and dual seat aircraft. It is usual for the designator to be cued onto the target by either, the on-board radar, the inertial navigation system, visually by the pilot using a helmet-mounted sight or head-up display, or in the case of dual seat aircraft by the weapon system operator manually steering the sight. This process determines a coarse sightline to the target, albeit not an accurate one. Time of course is involved in this process to achieve the necessary accuracy.

The use of the sight now differs between single and two-seat crew. Looking firstly at two-seat crew operation; a hand-over to the rear operator enables him to acquire the target, recognise it, and refine the sightline aim to the necessary precision. The pilot, who is completely free of this final aiming task can concentrate on the task of survival by monitoring his electronic warfare system if appropriate and undertaking evasive manoeuvres to defeat ground defences. The time taken to complete this fine aim process and the accuracy to which it can be completed are key input parameters to assessments as discussed below. The weapon sequencing especially against multiple target engagements are under the control of the weapon system operator and the fire control system itself. The ability of the system to engage several targets depends upon the speed with which the sightline can be changed from one target to the next together with the distribution of the targets on the ground. Design optimisation of the overall system in terms of field of view, its ability to be rapidly changed from target to target, the tracking aids to achieve accurate fine aiming, etc, are the keys to the success of the system.

The tasks in a single-seat aircraft, although similar, have important differences and impose penalties which usually manifest themselves in terms of some lost flexibility. After the coarse aiming sequence, the pilot has to recognise the target from the cockpit display of the designator image. This demanding task requires a period of concentration 'inside' the cockpit and detracts from the ability to fly low and safely. Following recognition the pilot has to complete the fine aiming task, and manage the fire control system. Considerable care in system design suggests these tasks are realistic and much simulation and flight assessment has taken place to establish likely levels of performance in terms of the times needed to complete the various phases. It is very likely, however, that evasive manoeuvres in the target area will only become possible after the sighting system has been locked to the target.

Typical line diagrams for both the single and dual-seat operation of such missiles are given in Fig 15.

Fire and forget weapons allow many system options to be considered, but for this paper I would like to limit my discussion to systems appropriate to the inertial weapon and the electro-optical homing weapon discussed above.

Looking firstly at the inertial weapon, in its most simple manifestation, the present nav/attack systems can be successfully used for weapon delivery. The target co-ordinates can be determined in standard ways, usually from the head-up display plus the appropriate sensor, and these values used to initialise the weapon. Again, the time to aim, $t_{ca}$, plays an important role in the effectiveness of the total system as does the angular cut-off in azimuth of the display system. This is illustrated in Fig 16 together with the typical weapon coverage plot. The delivery sequence after detection is therefore relatively simple, with the pilot flying his weapon coverage over the target, completing the aiming process, releasing the weapon and escaping. All tasks are 'head-up' and represent extensions of the present techniques which form the basis of ground strike training.

To achieve greater accuracy, especially at significant stand-off ranges demands a better fixing aid. This may well become available with further navigation aids for planned targets, but for 'visually' detected opportunity targets, the need exists for better sighting systems. The use of the designator in this role has many attractions, and can be used for precision fixing. The event sequences are very similar to those discussed for the semi-active laser guided weapon and will not be repeated here. However, one difference to note is that once an adequate fix has been taken against the required target, the weapon release sequence can be initiated, and the designator and pilot become 'freed' for other tasks.

Harmonisation errors are critical for this type of system updating and careful procedures need to be ...ted to achieve adequate performance. This additional accuracy element has to appear in the system b... ...

...ning now to the imaging 'fire and forget' weapon this presents a more complex range of system pos...ities. In its broadest concept, the weapon requires nothing beyond a coarse wide angle aiming devi..., whereby the pilot points towards the target, commits the weapon and repeats the process as required. The weapon takes care of the rest, by responding to the initial direction given by pilot, automatically capturing the target via its imaging head and initiating a release sequence when in range. This conceptual process, however, requires a degree of back-up from a supporting system if it is to become a robust delivery option. In particular, target recognition may be ssential before weapon release, or it may be operationally necessary to identify high priority targets in a relatively cluttered target rich scenario against which to mount the initial attacks. Additionally, assistance to the missile in determining an adequate locking success rate may require a human involvement, *ie* a value of $t_{fa}$ may still be appropriate. To achieve this may or may not dictate an electronic cockpit display of missile imagery; but a crew commitment is implied. Finally range is an important parameter when the weapon is being released under conditions that exploit its full potential. Some savings do arise, however, from the lower precision needed from on-board sensors and computing but again to achieve 'near limiting' weapon releases, caution needs to be applied in implementing such savings.

In summary, the critical processes that govern the overall system capability appear as common themes throughout the guided weapon options considered, effecting one or the other to a greater or lesser extent. These are:

(i) targets - how many? what distribution signature? etc.

(ii) target detection - what range, offset and probability?

(iii) time to coarse aim - $t_{ca}$ values for HUD, HMS, radar, navigation system etc.

(iv) target recognition - human, automatic, what range? how long does it take? etc.

(v) time to fine aim - $t_{fa}$ values for pilot with and without auto aids, how accurate?

(vi) weapon launch sequence - series, salvo, etc.

(vii) boundary constraints - missile coverage, aiming sights field of view, missile homing head limits, aiming sights angular coverage.

(viii) aircraft flight envelope - ground, defence coverage, evasion manoeuvre, manual or automatic flight control etc.

The interactions of these eight areas determine to a large extent the success of the mission and are the principle driving forces that govern the need for modelling, simulation and flight assessment, to determine the real capability and effectiveness of particular systems. The remainder of this paper considers the use of such techniques in understanding, assessing and evaluating various weapon systems.

## 6 SIMULATION

Simulation is a much used and valuable technique for investigating the principle effects that determine the performance of specific weapon systems either in terms of absolute values or as is more usually the case, in terms of comparisons between systems or the same system under different conditions.

Simulation, however, often means different things to various people and in the following discussion I would like to make a distinction between three types of simulation, to treat each in its own right, and to identify the role that each can perform in giving the utmost confidence that a specific system will give the required performance on the day.

The three areas identified are:

(i) the use of mathematical analysis to assess performance, to establish the critical parameters, and to arrive at a situation where the most promising system has been selected and optimised to a first order of magnitude. This process is often termed mathematical modelling and with the advent of high power ground computing complexes, is used increasingly.

(ii) the use of 'human-in-the-loop' simulations, to overcome the almost intractable problem of representing the human response in mathematical form and to expose experienced operators to the system design at an early stage. Facilities usually cover the research and development phase through to in-service training.

(iii) the use of flight vehicles to validate the previous two stages, to expose the system to a 'real world' environment and to demonstrate the engineering of the system.

Each of the above phases represents important stages prior to the realisation of any system in-service.

## 6.1 Mathematical modelling

The use of mathematical models to describe a process or processes is relatively well known in many major fields. The approach that is generally adopted is to represent the total process by a number of related interactions and then to examine the way in which the whole responds to changes in each interaction. In weapon system design, the need for such techniques has grown rapidly over the last five years or so as the complexity of the weapon system and the environment in which it is to operate has increased, and the constraints become more severe. Trivial examples are, a 20 km engagement is of little use at low level if a need exists in the weapon system for a line of sight between the target and aircraft. Likewise any weapon system that needs 20 s to deliver after target detection will have (in general) a zero capability from low level flight profiles. The model therefore identifies the credibility of specific performance parameters within the range of conditions likely to be presented by the operational scenario.

Two distinct schools exist as to the best approach to be adopted for arriving at the total answer and before citing examples it is valuable to identify the merits of both techniques. These are generally known under the titles of 'Monte-Carlo' modelling and 'Root Mean Square' modelling. Looking first at the latter, this is the simpler approach and relies upon the fact that the total process governing the attack effectiveness is driven by a series of forcing functions that are either totally uncorrelated or whose correlation functions can easily be identified. For example, target detection range, attack speed, terrain screening angles, aiming accuracy, aircraft flight manoeuvre, etc. Each aspect of the model can be treated separately, and provides a parametric output which can be combined with other error sources on an rms basis to give a final answer. Its great merit lies in the fact that the total model is visible to the assessor, a positive understanding of critical events is quickly achieved, a good insight to the process is obtained and the model can build in an evolutionary way without becoming overwhelming in its demands or computing power and time. It is also relatively easy to simplify if subsequent running shows that some process can be represented by a less rigorous global treatment.

The 'Monte-Carlo' technique on the other hand has achieved greater acceptance in recent years as adequate computing capacity has become relatively easy to obtain. As the name suggests, the technique is a means of handling the total probabilistic nature of the process. In effect each happening has a most likely value, together with a distribution of likely spread about this value. By inference the success or failure of the method rests on the modeller's ability to specify adequately the statistical nature of each process involved in the whole. Given a definition of the events, their sequence and their parameter statistics, the technique involves running repeated attack sorties using individual values of all parameters drawn independently from statistical populations with the stated distributions. Each run of the model provides a single answer to any one output parameter, and by repeated modelling runs a distribution of output parameters is built up. To obtain a picture at each condition at least 30 runs are needed. Since each run represents an attack, and may take more or less than real time, it can be appreciated that a significant computing commitment is undertaken when adopting this approach. The technique, however, is extremely rigorous with the appropriate statistics being applied at each part of the model, be they normal or not.

Having used both techniques, I believe each does have a valuable contribution to make. As a first cut at the problem, the rms method gives a good rapid insight into a process, and identifies many of the driving factors. For a better refined answer usually involving more of the contributing factors and to check that important interactions have not been overlooked, the MC approach gives excellent results.

To expand on these points further, the following example illustrates the use of modelling techniques to assess the delivery capability of a hypothetical weapon delivery system typified in its capability and coverage by Fig 17. The analysis covers the period of entry to the target area to weapon impact, but excludes any aircraft losses on the way to the target area and during the attack, and weapon lethality. Fig 18 describes the attack process. The aircraft enters the target area with a navigation error that is distributed normally with values dependent upon the accuracy of the system, and the time since and the position and accuracy of the last fix. A number of probability conditions are now summed in order to determine the range at which the target will be detected (if at all). Included in this process are:

(i) a free line of sight to the target;

(ii) detection as a function of range, height, speed and offset;

(iii) atmospheric effects;

(iv) search aids and any angular limits imposed, etc.

This initial compounding is a critical aspect of the model, since in this example it dictates the time available in the final attack sequence and hence is a major outer constraint in determining system usage.

The next sequence block to be modelled is the final engagement phase. Essentially this consists of flying the aircraft to bring the target within the weapon's coverage envelope, aiming the system and releasing the weapons. The aircraft model for the first of these stages consists of the statistics describing:

(i) a pilot response time;

(ii) a kinematic aircraft model, in terms of speed, roll rate, lateral 'g', etc;

(iii) a stabilizing time before any aiming can be established.

Finally, system aiming is covered in terms of event sequencing together with the aiming times and accuracies. This latter point is of some special interest because in general, systems fall into two classes; those that require the crew to achieve the best aiming accuracy, in whatever time is available, see Fig 11, and those that require a 'critical' level of performance to achieve for example a lock-on. Both types can of course be embraced by the model given the necessary data to define the known performance. By way of an illustration of the latter technique, Fig 19 shows a typical time history of a helmet mounted sightline in moving from position (1) to position (2) under vibration. An initial rapid transfer is followed by a period of hunting around F2 until a satisfactory set of conditions are obtained. Clearly the tighter the angular constraints needed at F2, the less likely the task is to be achieved, and hence the longer it will take.

Outputs from such a model are many, but by way of examples, Fig 20a-c, show examples of the variations of attack probability against:

(i) navigation accuracy;

(ii) detection range, and the influence of terrain screening;

(iii) aiming time.

Each shows the robustness of the model to the assumptions and where these become critical. Clearly it is desirable to determine a performance that is on any plateau of the curves.

Finally the 'MC' technique can be readily extended to multi-aircraft attacks against multi-targets; the limitations arising only from the amount of processing required to undertake the task.

### 6.2 Man-in-the-loop simulation

A. identified in the above, the event sequencing, the time taken and the consistency/accuracy of performance that can be achieved are very vital inputs to the success of the majority of systems. Precise measurement of the man's performance is important and ground based simulation facilities are valuable tools to explore and quantify these areas. Most of my remarks will therefore refer to facilities aimed at research and a number of such complexes exist.

The last decade has seen a considerable increase in the amount of weapon system simulation undertaken. Traditionally, the aerodynamicist and flight control engineers were the major customers, but it is now recognised that the system designer has much to gain from using the dynamic ground rigs. The problems of providing adequate simulation performance, however, are not few and in many cases considerable investment is needed if full mission ground attack simulations of repeatable accuracy are to be made. The increasing use of electro-optics in the systems to be simulated, aggravates these difficulties. The problems are in achieving good ground detail, good 'on-line' accuracy, good representation of narrow field of view electro-optical sensors together with of course standard good responses from the flying controls.

I do not wish to cover the details of the various complexities but to highlight some of the different approaches that have been used in this field.

The UK has had some notable success in developing the special purpose weapon aiming simulator that can be produced within a small budget. This has been adequately reported in the past, Refs 6 and 7. Suffice to say that the use of high quality cine film has much to commend itself in terms of realism, cost and accuracy and the disadvantages of limited flight manoeuvre and change in aspect have minimum effect. On-line measurement is practical without the use of video analysis equipment. For designator assessments, the cine system can be complemented by a small high detailed model area, viewed with a television system. The total package, Fig 21a,b, provides all the facilities necessary within a modest framework. Other film-based facilities have been used for this work, notably the 70 mm film system at Boeing Seattle.

The main deficiencies with the film-based systems lie in their limited coverage and response to flight path changes, and although these have been identified as areas of limited importance, for tactical simulations against opportunity targets there is clearly a need for something better.

Undoubtedly something better implies improvements to the outside world representation and these are inevitably costly. Two approaches are in use. First, there is the large detailed model based systems that employ closed circuit television systems viewing the model in order to reproduce the forward scene for the pilot, and any electro-optical sensor simulation. The facilities, of which a number now exist, provide a good flying area with realistic motion and orientation cues for the pilot. The television image may either be projected on a screen or viewed directly on a collimated display. In the past image detail has been a problem, but modern modelling techniques coupled with providing a smaller flying field now permit.

excellent presentations to be produced. Typical scale ratios cover the range of 500-2000:1. The second window, essential for example for designator or missile homing head simulations, can usually be accommodated but some compromise in terms of image performance and co-location of viewpoint is likely.

These facilities enable realistic target scenarios to be set-up and realistic missions to be flown and assessed. A further important point of application is the capability inherent within such facilities to assess real hardware in a dynamic environment. Tests relating to displays, controllers, video processing, auto-tracking equipment, missile homing heads, etc, can usually be integrated with the tasks of the crew to provide an extremely realistic on-line examination of the system, "wrinkles and all", prior to field trials.

The second fruitful avenue to pursue for better outside world representation comes from the technology of computer derived images. The recent cost effective advances in computer capability now bring highly detailed computer generated imagery within the grasp of the simulation designer and, for the future, developments in this field will probably set the pace, enabling effects such as atmospheric attenuation, partial cloud cover, and thermal signatures of the scene and targets to be adequately represented. Some limitations, however, are likely in the direct application of flyable system hardware for evaluation purposes, because of the software nature of the scene. However it is possible that some evaluation of tracker algorithms may be possible if a very high degree of spatial resolution can be achieved.

The key feature of work conducted on any of these facilities is a 'controlled environment', that is, the performance of system should be assessed under a set of tightly defined conditions. Wherever possible, the degree of randomness in the experiment should be minimised to that introduced by the pilot or crew.

By way of an example, Fig 22 shows the C&P performance of two systems, A and B, in terms of the position of the target relative to the flight path at the moment of target detection. The exercise constrained the pilot to engage the target from four precisely controlled ranges and two target offsets, balanced equally left and right. The results show system A better than B at the larger ranges but *vice-versa* when time is short. Note the cross-over point favours system B as the offset is increased. This type of result is typical of the data that needs to be measured and has considerable significance when fed into the modelling studies discussed in section 6.1. For under poor engagement conditions it may prove quite impossible to ever realise the anticipated benefits of 'system A'. However, if the starting conditions of the experiment had been left to chance, it is likely that little variation would have been found in the results.

The final example relates to the use of eye and head movement recording equipment. Fig 23 shows a stylised set of results obtained from a head-out, head-in the cockpit task typical of an attack that requires visual cueing and coarse aiming followed by head-down recognition and lock-on sequence. Such results showing the times spent at the relative tasks, the variability of these times as functions of engagement conditions, and to some extent the measure of confidence that the pilot has in his automated system, are all key inputs to assessments.

## 6.3 Flight demonstration

My third category is flight demonstration, which is essential element of testing in order:

(i) to validate the expected performance of the system,

(ii) to establish the hardware realisation of the system,

(iii) to demonstrate the viability of the system in .e air to the customer.

This phase is of increasing importance as systems become more complex, more costly and hence more difficult to prove adequately in a research or piecemeal sense. It is an essential part, together with the two previously discussed areas, in establishing a total understanding of the weapon system. However, being the most costly part too often it is neglected or inadequately covered.

To manage such a trial successfully requires a represented aircraft, although not necessarily a specific type, a representative system, especially in terms of cockpit functions and displays, and a weapon or good weapon emulation dependent upon the issue being resolved.

As generalised in the above aims, the trials should be conducted against an expected performance, for in flight, conditions are never controlled precisely and for example trials safety considerations may exclude interesting parts of the delivery envelope. The trials therefore are used to pin down the expected performance curves at a number of points from which interpolation or at the worst small extrapolations can be made to all the areas of interest. Exposure to more than one pilot is desirable, although resources usually limit this activity to about four at the most. Finally, target scenario is important and representative arrays are important especially where electro-optical techniques are being assessed.

Hardware realisation is an area that is now being taken more and more for granted by the assessor and to some extent this is as it should be. However, both in determining whether the equipment meets an adequate standard and as a final cross-check that the standard is adequate, such demonstrations are highly desirable. Equally, no theoretical assessment has ever included all the significant effects experienced by the practical application and demonstration has a vital role to play here.

A further important area in the total proving and procurement of weapon systems, is the need to expose the system to the users. This is particularly important if new concepts and technologies are embraced, for it is often very difficult for Service personnel to relate well to academic assessments performed by the design engineers. The proof of the pudding is in the eating, and aircrew feed best in the air environment; a well equipped flight vehicle therefore can become an important sales tool in demonstrating new systems to a wide Service interest.

Finally, to seal the feedback loop from flight assessment to modelling, the flight trials enable a first look to be taken at the tactics and deployment of the system under pseudo real conditions. This usage of the system is an important element, since without it, the analytical studies will have only been working with best estimates in the absence of a practical system. Many is the time when the real value of a technology only emerges after trials, and indeed was not even considered during the early design of the equipment.

An important aspect of the flight assessment is instrumentation, for it almost goes without saying that if a close validation is to be obtained between ground modelling and flight trials, then all aspects of the trial need careful recording to make this possible. Adequate thought is usually given to the 'on-board' aircraft problems, where a full range of system parameters together with the pilot/crew responses should be covered. Of equal importance is the relationship of the aircraft with the real world. The position of the aircraft relative to the target can be found either from specially installed recording equipment in the target area or alternatively by using photographic techniques from the aircraft itself. However, the area often neglected is that of conditions pertaining in the outside world itself; conditions which can dramatically effect the results obtained from the trial. These include:

(i) the target, its position relative to others in the vicinity, its aspect relative to the aircraft's flight path, its signature in the appropriate wavebands relative to the background, and any motion effects;

(ii) the atmosphere, its transmission properties, etc;

(iii) the terrain, its broad and local features where they affect the flight path and intervisibility.

The value of such a data base cannot be emphasised too much, since many of these features are the driving influences in relating the measured performance to the expected result as obtained from the theoretical modelling. Indeed, wherever possible the conditions of the trial need to be selected with care, to ensure that results are obtained over a sufficiently wide range of engagement conditions, to cover the expected operational in-Service usage of the system. Past experience suggests this is done when deciding the aircraft attack configuration etc, but often not with sufficient care on environmental matters.

## 7 CONCLUSIONS

The aim of this paper has been to review the influences and constraints that govern the delivery of tactical air-to-ground guided weapons, and in doing so I hope I have been able to illustrate the need to consider the total weapon system delivery performance in the context of the real scenario. The start must be made at the target, with information relating to its likely position, concentration, motion, aspect, signature relative to its local surroundings, etc, and followed by the influences of terrain and atmosphere at the likely places and times where attacks may be necessary. Given this essential framework, the weapon system in total can be joined on, to establish an attack capability. The crew form an essential element of the 'total system' and determining their spectrum of performance is a vital link in this process.

The distinct roles of modelling, 'man-in-the-loop' simulation and flight demonstration all have valuable contributions to make, but when related each to each other, form a powerful methodology for enhancing the understanding of the system, for optimising performance, for establishing the range of operating conditions under which a worthwhile performance can be achieved and enabling a sound technical choice to be made between alternative options.

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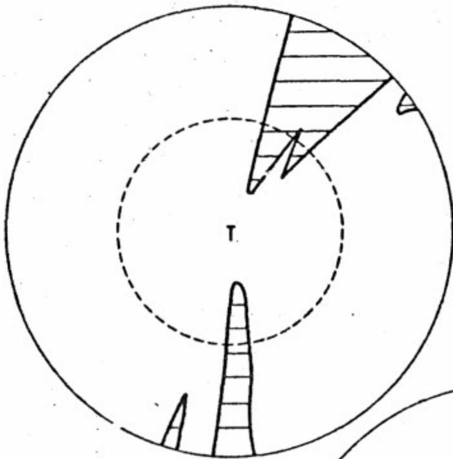
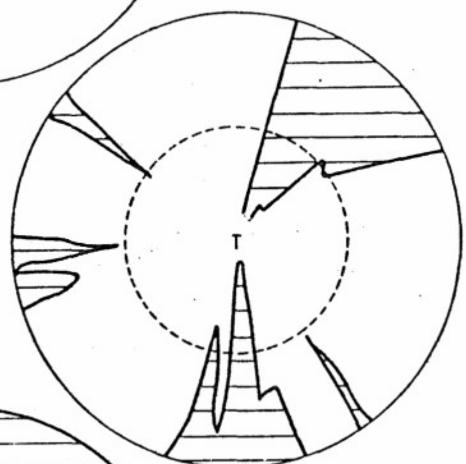


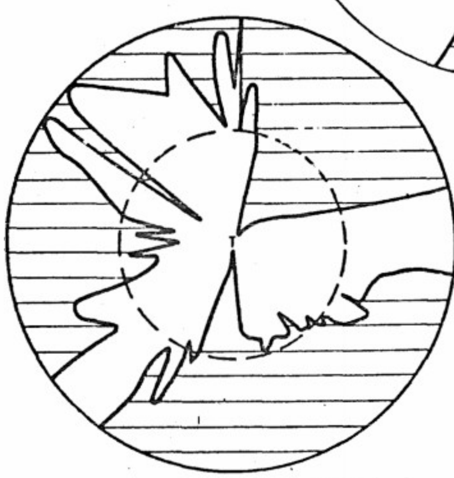

Inner circle = 6 km range
Outer circle = 12 km range
Target at 'T'
Shaded areas denied line of sight to target

Fig 1a Approach height = 1000 ft




Inner circle = 6 km range
Outer circle = 12 km range
Target at 'T'
Shaded areas denied line of sight to target

Fig 1b Approach height = 500 ft




Inner circle = 6 km range
Outer circle = 12 km range
Target at 'T'
Shaded areas denied line of sight to target

Fig 1c Approach height = 200 ft

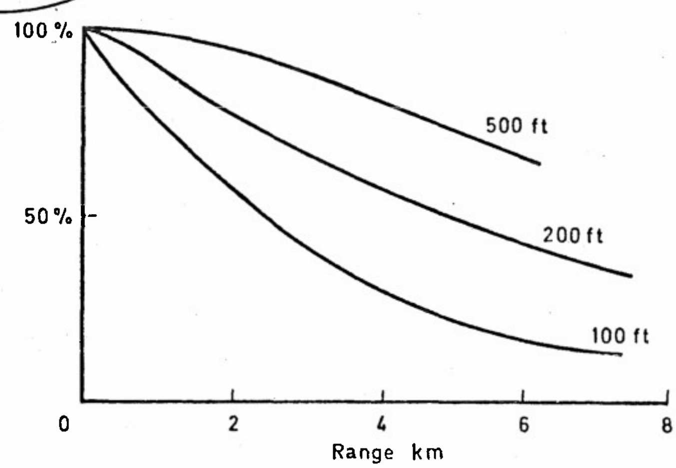



Fig 2 Cumulative probability of unmask as a function of range

Fig 3a

Fig 3b

Fig 3c

Fig 3d

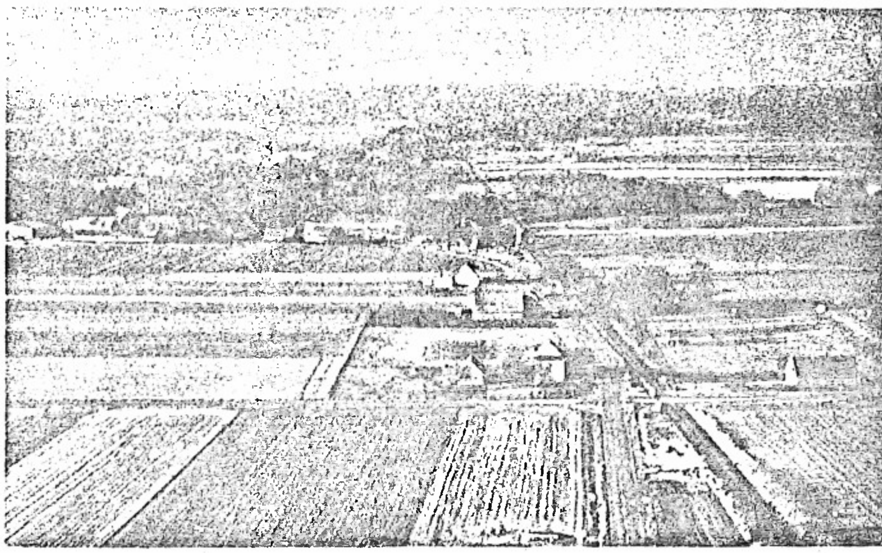


Fig 3e

Fig 3f

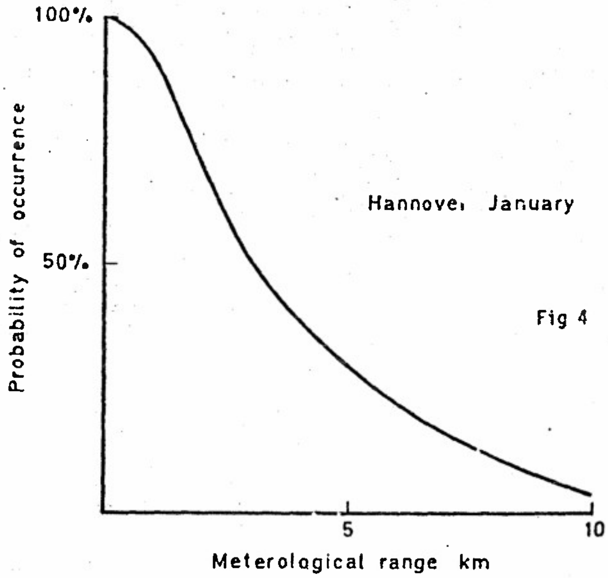
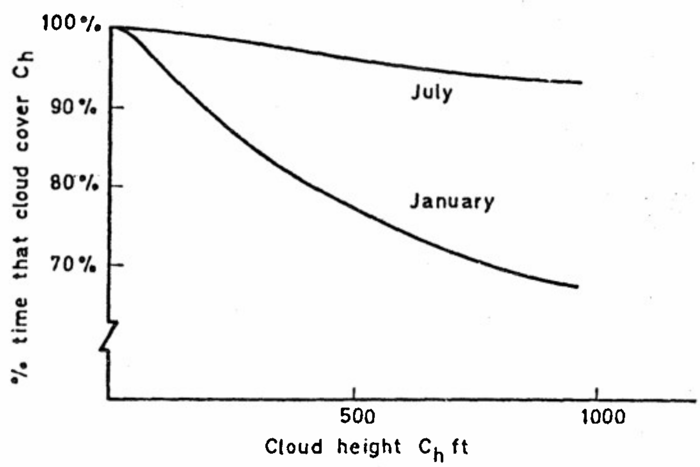



Fig 4 Frequency of occurrence of met range exceeding stated value

Fig 5



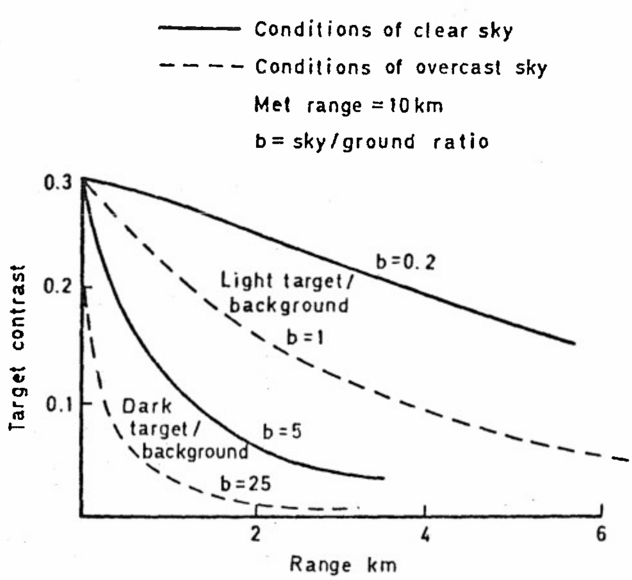




Fig 6 Loss of target contrast in visual range

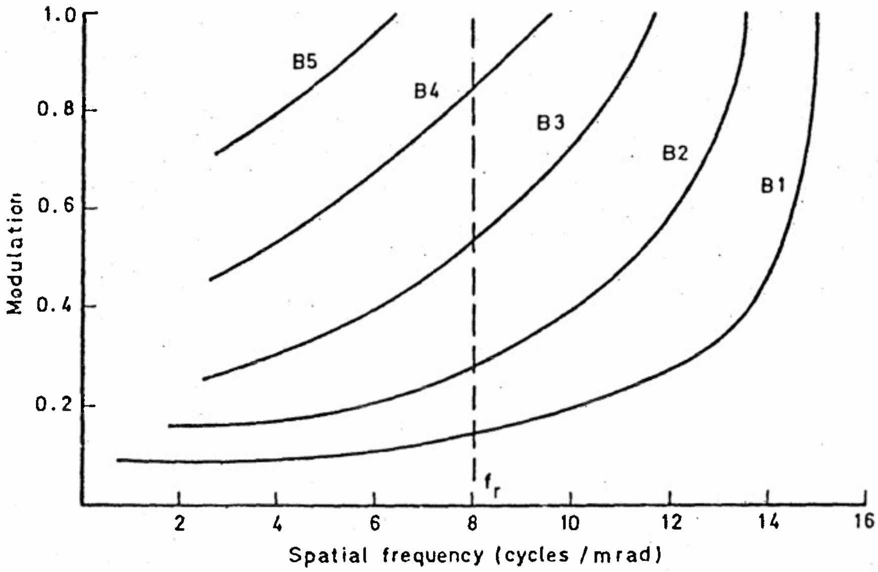



Fig 7 Hypothetical 'MRM' curve

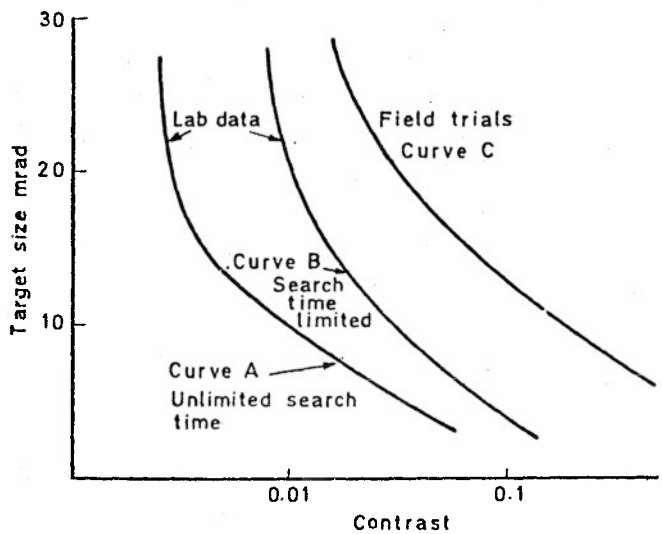



Fig 8 Target threshold as a function of its contrast

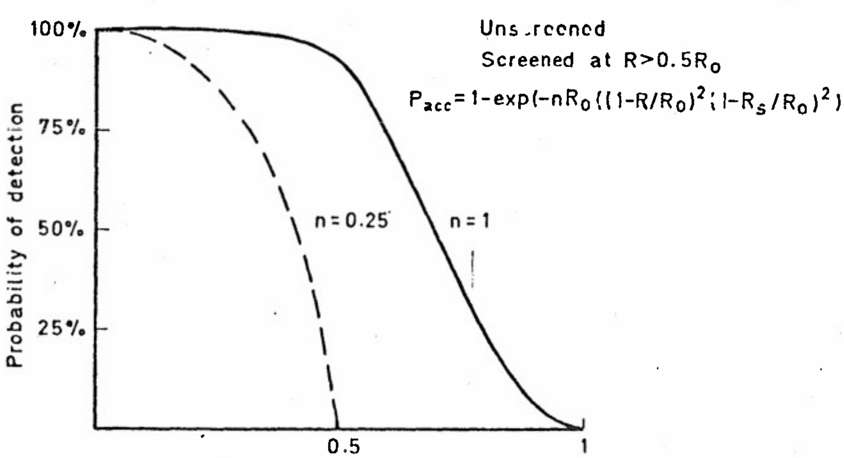



Fig 9 Cumulative acquisition probability as a function of range

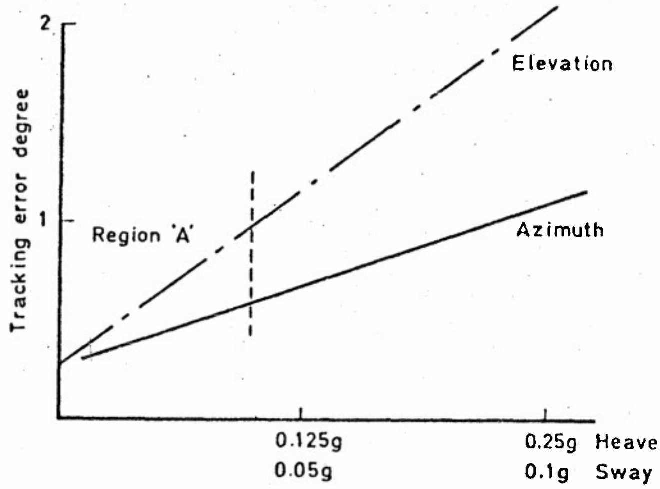



Fig 10 Variation of aiming error as a function of ride quality for a helmet mounted sight

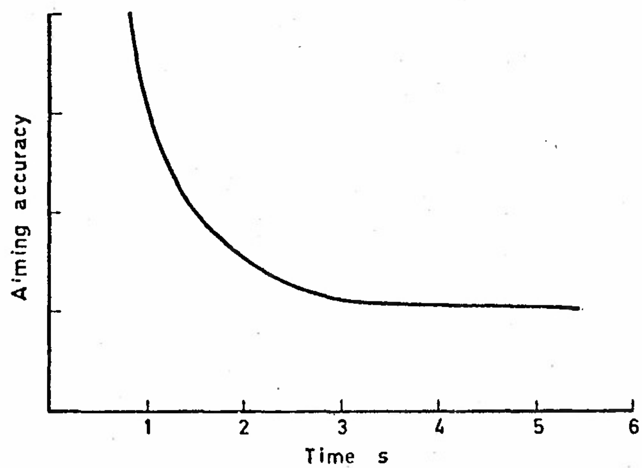



Fig 11 Variation of aiming accuracy as a function of time

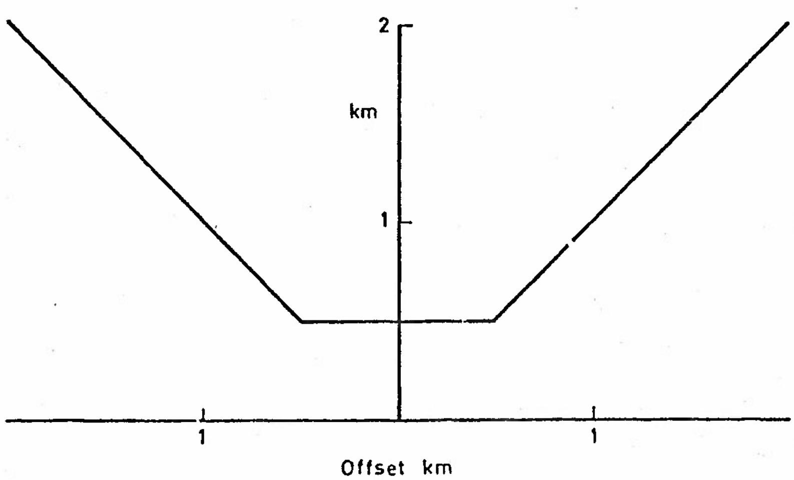



Fig 12 Typical lateral coverage plot for a SALGW

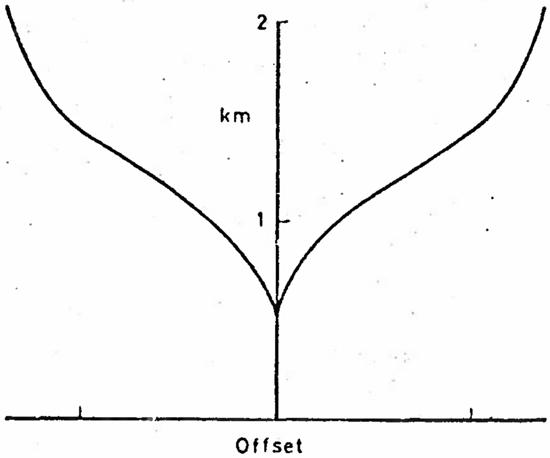



Fig 13 Typical lateral coverage of an unpowered inertially GW

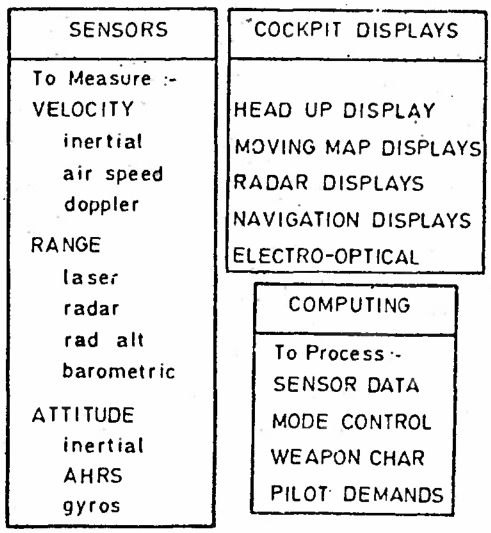
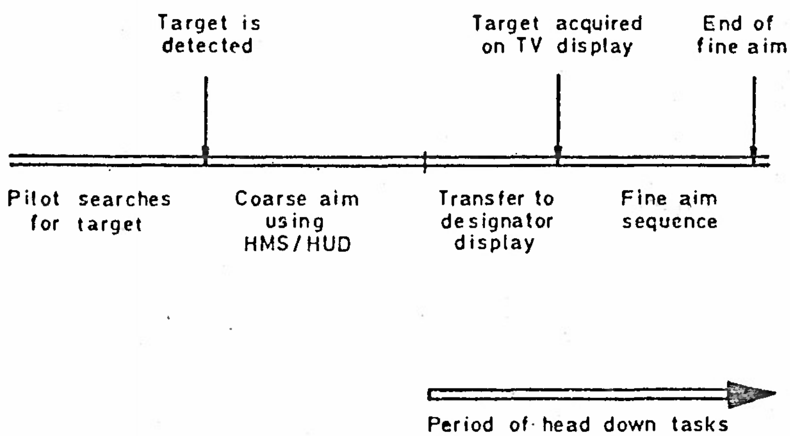
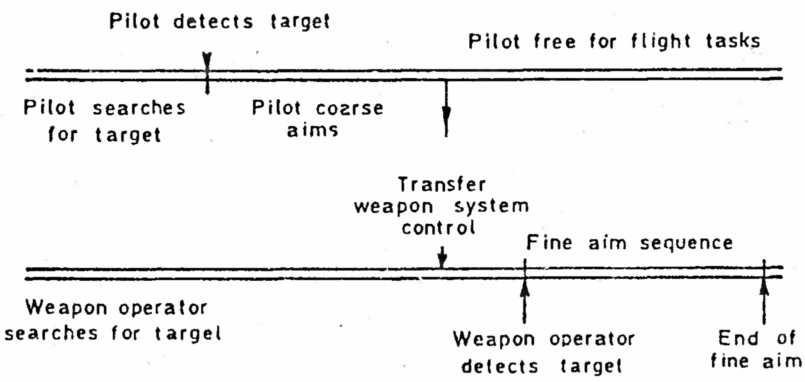



Fig 14 Technology areas of Nav/attack systems




a Single crew aircraft




b Two crew aircraft

Fig 15 Task sequence

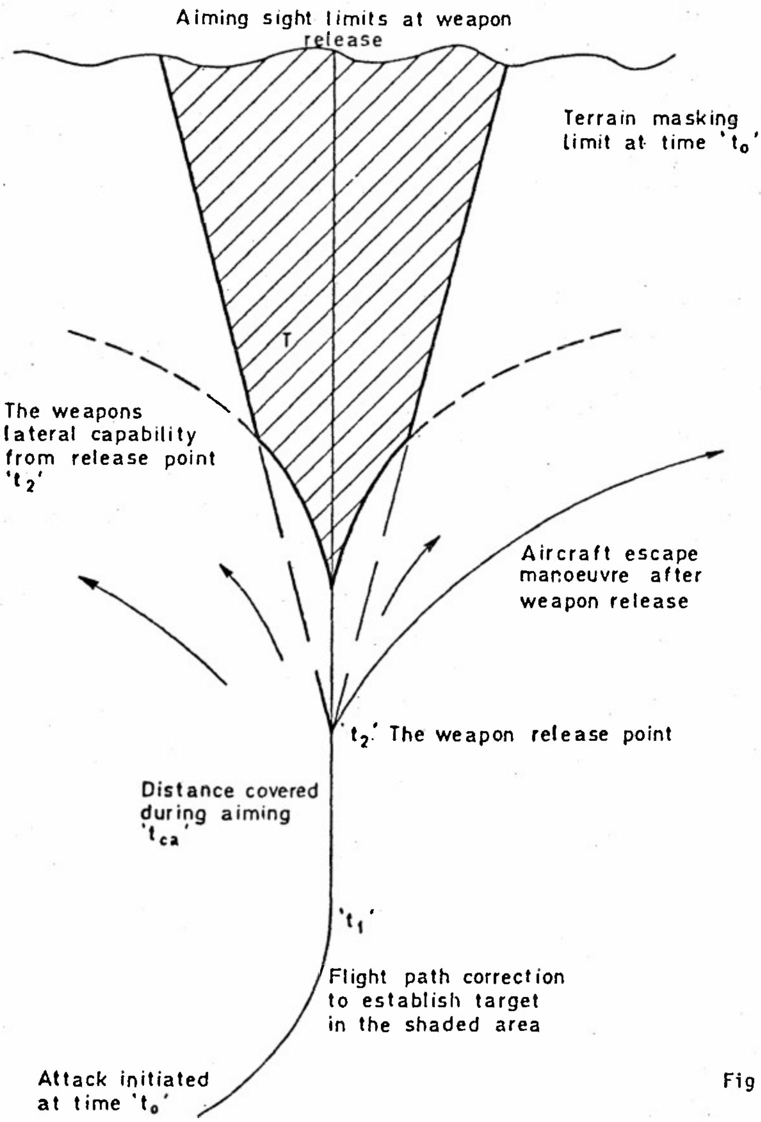

Aiming sight limits at weapon release
Terrain masking limit at time 't$_0$'
T
The weapons lateral capability from release point 't$_2$'
Aircraft escape manoeuvre after weapon release
't$_2$' The weapon release point
Distance covered during aiming 't$_{ca}$'
't$_1$'
Flight path correction to establish target in the shaded area
Attack initiated at time 't$_0$'

Fig 16

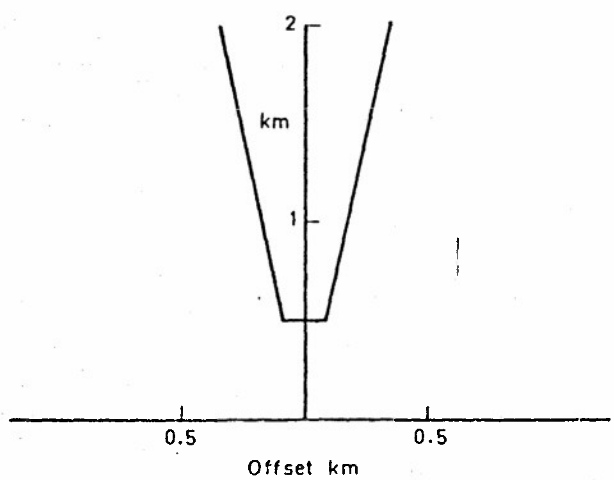

2
km
1
0.5
0.5
Offset km

Fig 17

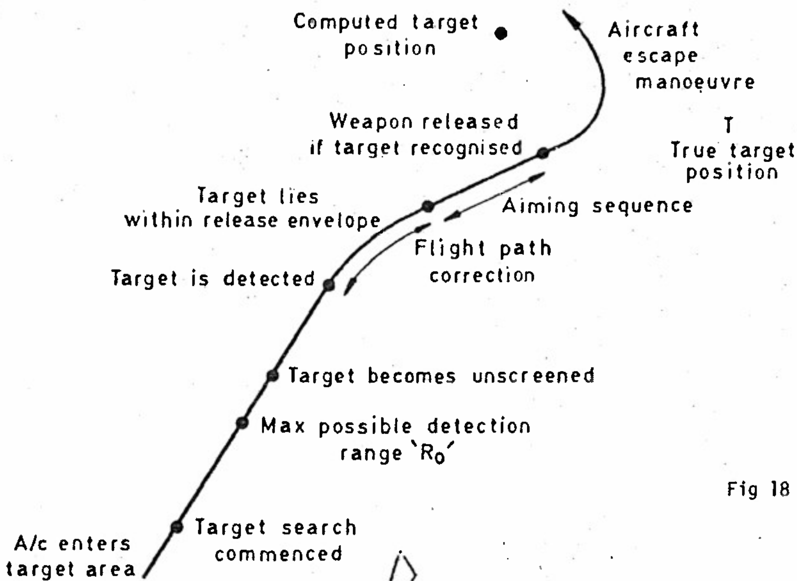



Fig 18

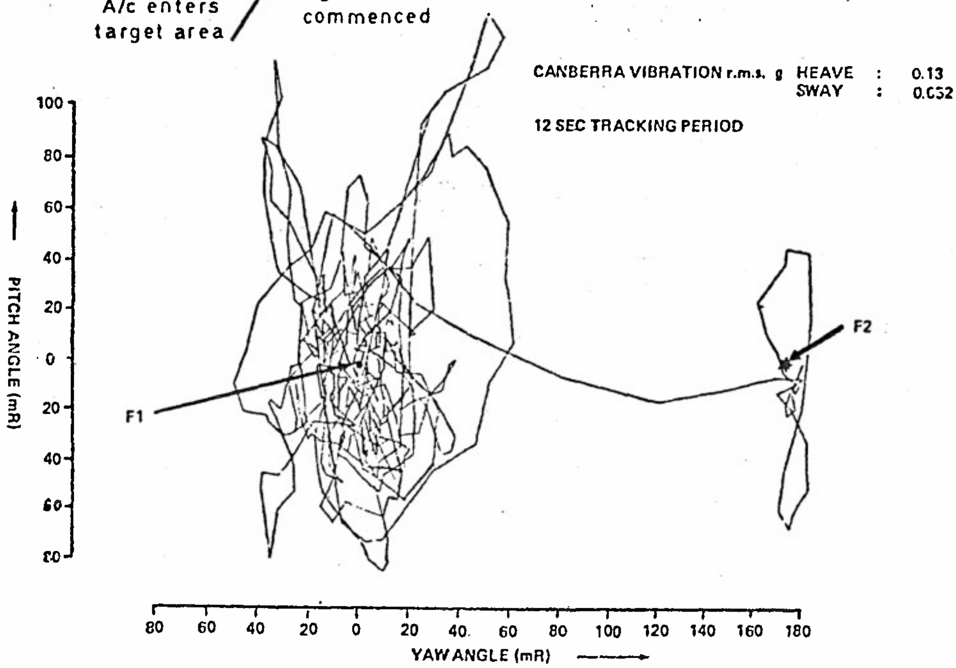



Fig 19 Helmet mounted sight: radial pointing error

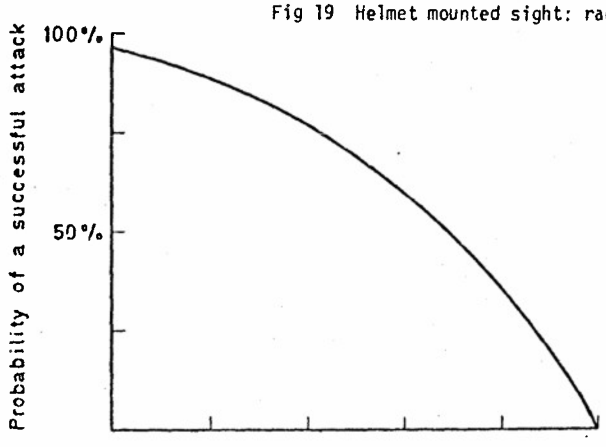



Fig 20a Variation of attack success with system aiming time

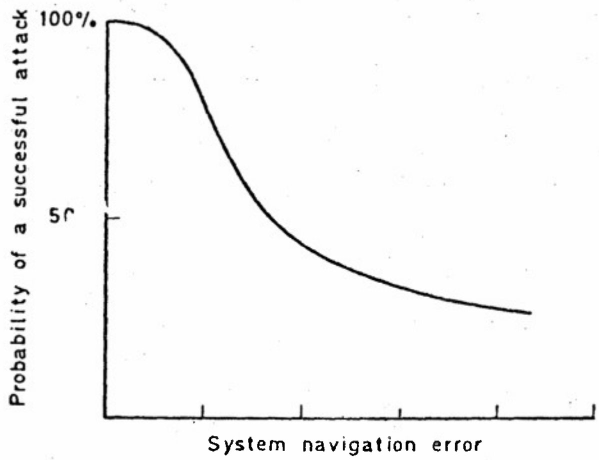



Fig 20b Variation of attack success with navigation performance

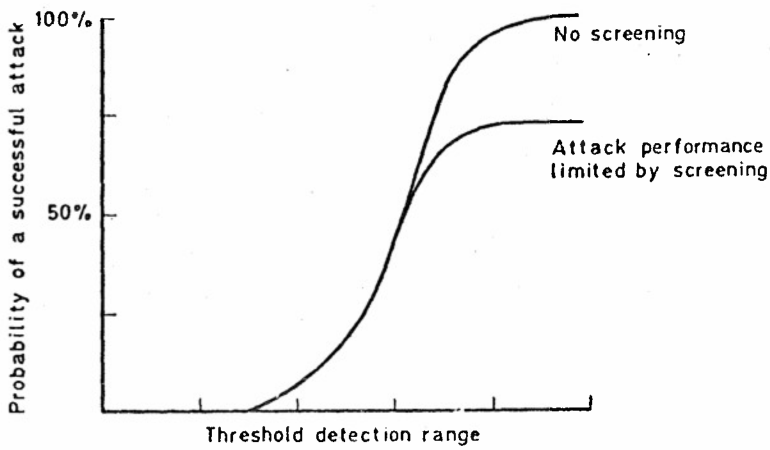



Fig 20c Variation of attack success with target detection range '$R_o$'

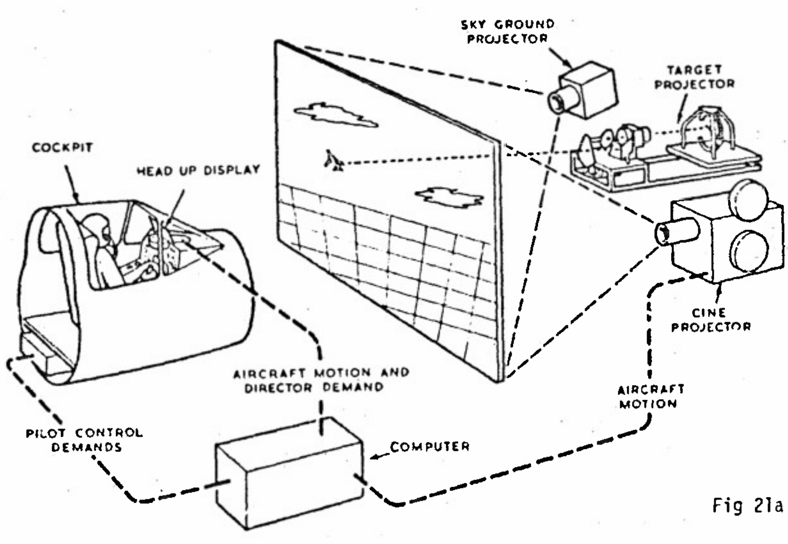



Fig 21a

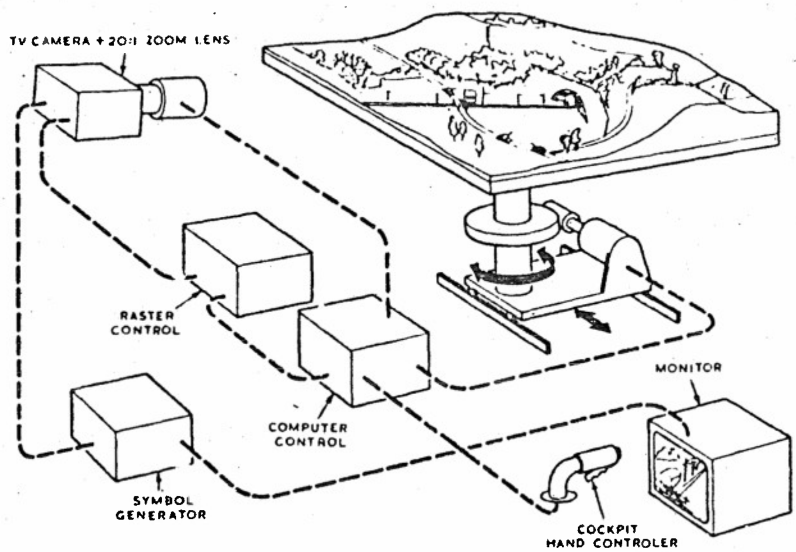



Fig 21b

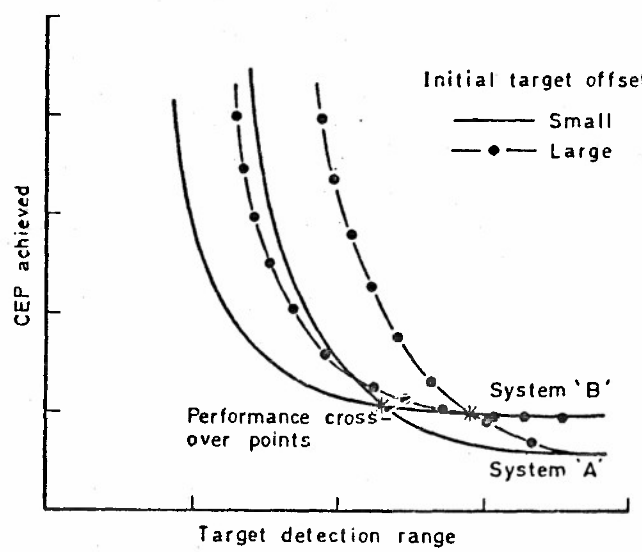



**Fig 22** Variation of weapon system delivery performance as a function of starting conditions

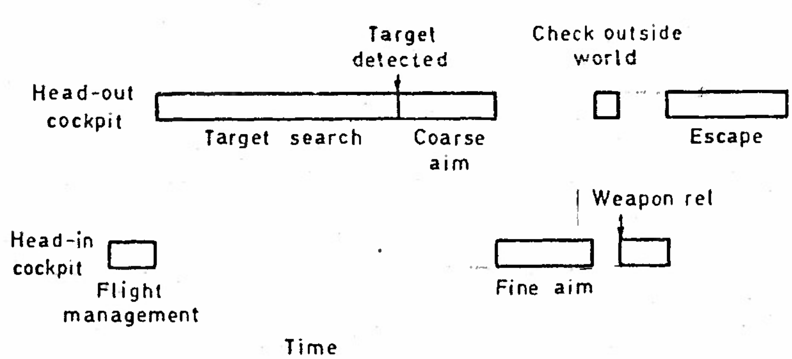



Fig 23 Typical split between head-in/head-out of cockpit during air-to-ground delivery

# NEW METHODS IN THE TERMINAL GUIDANCE AND CONTROL OF TACTICAL MISSILES (Air-To-Air)

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## SUMMARY

The purpose of the guidance law is to determine appropriate missile flight path dynamics such that some mission objective might be achieved in an efficient manner. There are many classical terminal guidance laws which have been used for tactical missiles - each characterized by varying degrees of performance, complexity and seeker/sensor requirements. The increased accuracy requirements and more dynamic tactics of modern warfare render contemporary guidance laws unsatisfactory in many applications. This is especially true in terminal air-to-air missile engagements. Improving performance involves a tradeoff between more sophisticated hardware or more sophisticated software. Increased hardware sophistication almost always results in increased costs. With the advent of new theoretical methods and low cost/high speed microprocessing techniques, the potential exists for tremendous increases in missile brainpower with little or no corresponding increase in cost.

## 1. INTRODUCTION

Before I describe some of the newer methods which are currently being investigated for the terminal guidance and control of tactical missiles, I would like to put this topic in perspective by outlining the overall tactical missile guidance and control scenario. Consider the mission for tactical missiles being described by two different (but not mutually exclusive) scenarios: air-to-air vs air-to-surface combat and standoff vs short-range combat. Standoff (or mid-course) guidance is required when the missile is launched at such long ranges from the target that either the missile seeker cannot "see" the target or, if it can, the available guidance information is of sufficiently poor quality that it is unusable. In such case the guidance law usually consists of some pre-programmed strategy such as "maintain launch heading and a constant altitude" or "fly directly at where you think the target might be". In some cases tactical missiles don't have seekers and the complete trajectory can be thought of as a type of midcourse guidance. In those cases where a terminal seeker is "locked" onto a target and providing reliable tracking data (short-range combat), the strategy is called terminal guidance. The dynamic requirements of terminal guidance are usually more stringent because all the trajectory errors which have accumulated must be corrected in a very short time. Figure 1-1 graphically depicts the midcourse and terminal guidance phase of a tactical missile.

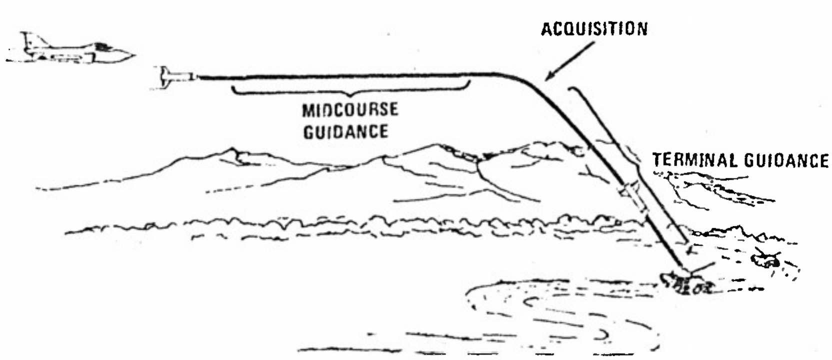


FIGURE 1-1 TYPICAL TACTICAL MISSILE TRAJECTORY

Another way to characterize missile guidance and control is by considering the types of targets involved. Surface targets are generally stationary or slow moving, although they may be difficult to detect and track. On the other hand, aerial targets are highly maneuvering and unpredictable, but usually easier to acquire. From a guidance and control point-of-view, aerial targets stress terminal guidance the most.

I limit this paper to an examination of terminal air-to-air missile guidance and control concepts against maneuvering aircraft, not because the other three cases are unimportant or without problems, but primarily because I view this as the area of greatest payoff for the application of advanced guidance and control techniques. Certainly the terminal air-to-surface guidance and control problem is a subset of this more general problem. Midcourse guidance, both air-to-air and air-to-surface, is primarily an energy management and inertial instrumentation problem. Although advanced control and estimation techniques are applicable to this problem as well, the objectives are sufficently different from terminal guidance and control that unified treatment of both problems is not within the scope of this lecture.

## 2. MAJOR MISSILE SUBSYSTEMS

Although the exact configuration and subsystem description of a tactical missile depends upon many factors, it is possible to generically describe the subsystems and their inter-relationships with the aid of figure 2-1, which is a functional diagram of the major subsystems. Figure 2-2 illustrates where these subsystems might be physically located on a typical missile. A brief description of each subsystem follows.

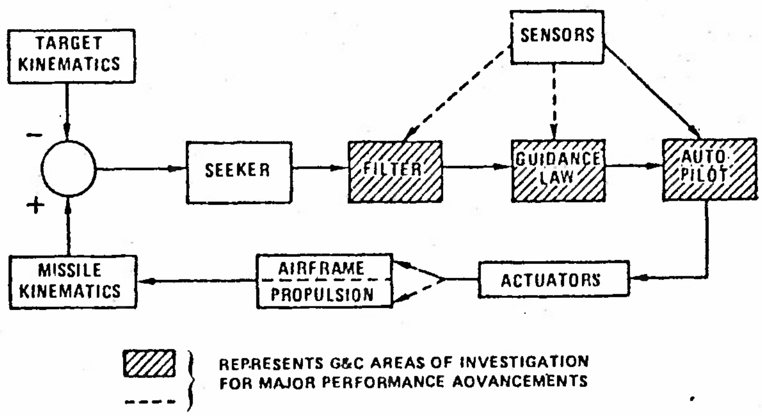


FIGURE 2-1 MAJOR MISSILE SUBSYSTEMS

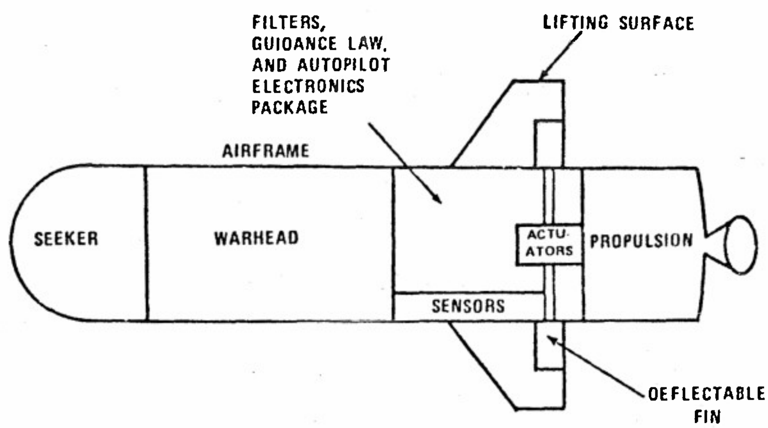


FIGURE 2-2 TYPICAL PHYSICAL LOCATION OF MISSILE SUBSYSTEMS

The seeker can be thought of as the "eyes" of the missile. Its purpose is to detect, acquire, and track a target by sensing some unique characteristic associated with it. This unique characteristic usually consists of the radiation or reflection by the target of energy in a specified region of the electromagnetic spectrum. Typical regions include ultraviolet, infrared, laser, visible, millimeter wave, and radar frequencies. Some missiles may have seekers which can operate in more than one region at the same time or at different times.

Detection is the process whereby the seeker senses a certain amount of energy (in some region of the spectrum) above that normally expected from background or internal seeker noise. Acquisition is the process whereby the seeker, after experiencing one or more incidents of detection, decides (according to some pre-established criteria or algorithm) that a valid target has been located. Tracking is the process whereby the seeker continually specifies the angular location of the target relative to some fixed coordinate system.

There are several methods available for tracking a target, depending on whether the seeker has a wide or narrow field-of-view or whether the seeker is gimballed or fixed to the airframe. The instantaneous field-of-view is the angular region (usually conical) about the seeker centerline which is capable of receiving useful energy.

If the seeker has a large field-of-view, it is possible to fix the angular orientation of its centerline relative to the airframe centerline. (See figure 2-3.) The type of tracking information avialable in such case is an indication of the angle between the line-of-sight (straight line from missile to target) and the missile centerline. This, plus possibly other information, is available for missile guidance.

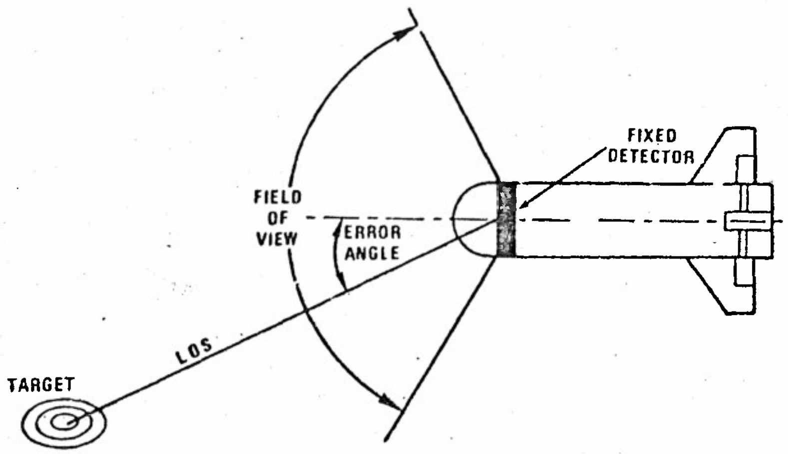


FIGURE 2-3 BODY-FIXED (STRAPDOWN) SEEKER

If the seeker has a narrow field of-view, it is usually mounted on a gimballed platform. The seeker maintains the target within the narrow field-of-view by rotating the platform. (See figure 2-4.) If the platform is inertially stabilized, the rotation is accomplished by applying torques which are proportional to the target displacement from the field-of-view center. The tracking information provided by this type of seeker is an indication of the inertial rotational rate of the line-of-sight. This, plus possibly other information, is available for missile guidance.

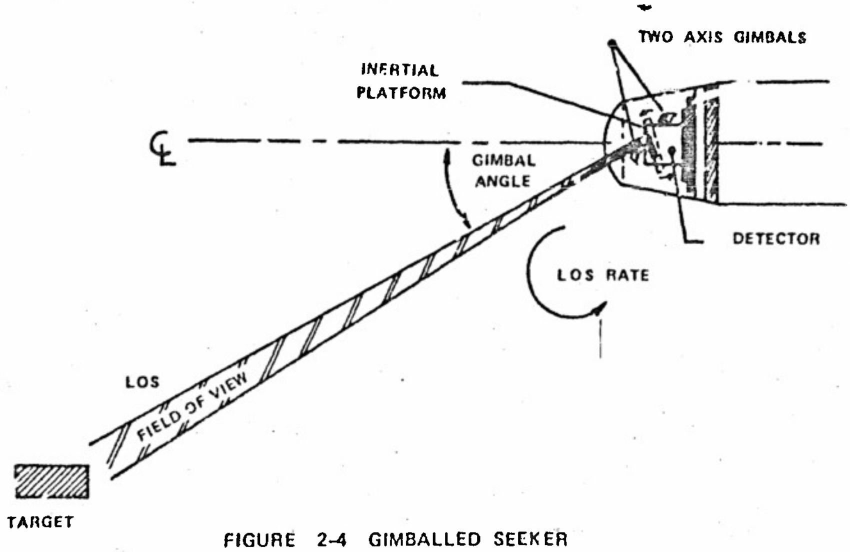


FIGURE 2-4 GIMBALLED SEEKER

Other information which the seeker might be capable of providing to a guidance law is missile-to-target range and/or range rate. Radar seekers are the only ones which currently provide such information. (Active radar seekers can provide both, semi-active radar seekers can provide range rate, and passive radar seekers can provide neither.) Techniques involving modern estimation theory are being studied which might provide this same capability for passive seekers and/or other frequency spectrums.

The next three subsystems can be thought of as the "brain" of the missile. The filter operates on the seeker data to produce a clearer "image" of the target behavior by extracting the pertinent kinematic variables. The guidance law decides the best trajectory (physical action) for the missile based upon its knowledge of the missile's capability, target capability, and desired objectives. A command is then sent to the autopilot, whose function is to determine what "muscle" control (actuator position) is required to best execute the command. The guidance law and filter are the major subjects of this paper, and each will be discussed in great detail later. However, the makeup of this "brain triad" depends heavily on the nature of the other subsystems, so I will finish my review of these before I dismiss them from further consideration.

The purpose of the actuator is to alter the external geometry of the missile such that the net forces which result will approximate the guidance law command. This alteration may take the form of a wing deflection, tail deflection, canard deflection, thrust control, or some combination of these. The first three alterations change the aerodynamic properties in such a manner that the proper moments and forces are achieved. Actuators require an external energy supply to accomplish their task.

The airframe serves two purposes. First, it is the container for all the other subsystems (including the payload). Secondly, by proper design and in partnership with the propulsion, it can be used effectively to produce the required lift and drag forces for accomplishing the mission objectives.

By virtue of Newton's Second Law and all its ramifications, these net forces determine the missile's kinematic variables, such as position, velocity and acceleration. These variables, in combination with those produced by the target, result in something new for the seeker to see.

## 3. CLASSICAL GUIDANCE LAWS AND THEIR PROPERTIES

The guidance and control laws used in current tactical air-to-air and ground-to-air missiles are based largely on classical control design techniques. These control laws took birth over 25 years ago and have evolved into fairly standard design procedures. Though the specific guidance and control law varies from one missile to another (depending on its size, weight, cost and manufacturer), the following basic characteristics are common to all of the missiles in the present Air Force inventory:

(1) The overall control of the missile is divided into two or more loops. The outer guidance loop controls translational degrees of freedom, while the inner, autopilot loop controls missile attitude.

(2) Proportional feedback is used to correct missile course in the outer loop (commonly referred to as proportional navigation or "pro nav"). Figure 3.1 illustrates the pro nav concept. Pro nav is quite successful against nonmaneuvering targets.

(3) In the inner loop, the roll, pitch and yaw channels are uncoupled and are typically controlled independently of each other.

(4) Sensors typically measure aspect angles in pitch and yaw planes and rates may also be available. Advanced sensors may measure other variables.

(5) No explicit state estimators are used and the signals are filtered to reject high frequency noise.

(6) All commands are amplitude or Torque constrained to ensure autopilot and missile stability.

Classical controllers have two major advantages, simplicity in design and simplicity in implementation; but they also have several problems. Table 3.1 indicates how characteristics of classical short range air-to-air missile guidance and control laws lead to advantages and disadvantages in design and implementation.

Early missiles used a pursuit form of navigation in which steering commands were generated to drive the angle between the line-of-sight (LOS) and missile velocity vector to zero. That is, the missile steers to head straight for the target. This worked well for non-moving or slowly moving targets but was seriously degraded for fast targets such as those found in the air-to-air environment. In the air-to-air mission the missile trajectories were clearly suboptimal and usually ended in tail chases. However, this guidance law does have the advantage of being relatively insensitive to system noise.

| FEATURE OF CLASSICAL MISSILE CONTROLLER | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| Pro Nav Guidance | (1) Requires only LOS rate information from the seeker. LOS rate is often available from somewhere in the seeker tracking loops.<br><br>(2) This information is used in a simple, easy to implement manner. Also autopilot design becomes more universal; more independent of the properties of the seeker. | (1) Much more information about the pursuit/evasion problem is known or available. Not using this information in the guidance law degrades performance. |
| Uncoupling of Steering and Roll Motions | (1) Can use classical control theory to select autopilot gains. Effects of variations in parameters is well understood.<br><br>(2) The resulting autopilot design is not very involved. The autopilot can be implemented with either digital or analog circuits. | (1) The missile angle of attack, and thus the missile's maneuverability, is limited by the autopilot. These limits are imposed to keep the missile flight stable, but the limits could be raised if a different method of autopilot design were used. |
| Dither Adaptation Scheme | (1) Allows a larger flight envelope, but still is easy to implement. Can be implemented with analog or digital circuits. | (1) Only partially adaptive. Other adaptive methods, perhaps more involved, could be used that would improve missile performance. |
| Direct Use of Sensor Data | (1) Reduces on-line data processing requirements. | (1) An optimal control law will require use of information that is not directly measurable. |
| Burning All the Engine Fuel at Once | (1) Engine design is simpler. | (1) May not be optimal solution for minimizing miss distance. |

TABLE 3-1 MAJOR ADVANTAGES AND DISADVANTAGES OF CLASSICAL G&C DESIGNS

Several methods which would have the missile lead the target were considered. The goal was to have the missile travel a shorter path to the target, which would in effect increase missile range capability. One method considered was a fixed lead in which the missile steered to a heading that was a fixed angle ahead of the target. When the target merely changed course, the missile performance degraded leading to excessive maneuvering in an effort to re-establish the lead angle when simply using a new lead angle would have resulted in a more favorable trajectory.

The development of proportional navigation was a major break-through in homing missile guidance. In pro nav, steering commands are given so as to drive the line-of-sight-rate to zero. Subsequent studies using optimal control have found pro nav to be an "optimal" guidance law when the missile and target have constant velocity, the missile is inertialess, and the only optimal criterion is to minimize terminal miss distance. However, assuming constant missile velocity is a serious assumption that neglects considerable thrust and drag effect. Because thrust and drag are present, pro nav is not optimal even against constant velocity targets (see Figure 3.1). Moreover, the targets seldom have constant velocities. Despite its shortcomings, pro nav is easy to implement and, for many years, provided satisfactory missile performance. Therefore, it has seen considerable use, although it is somewhat sensitive to unfiltered system noise.

Several varieties of pro nav are in current use. For the most part, these differ as to how the navigation gain is determined. The navigation gain is the ratio of commanded steering rate to the LOS rate ( $\dot{\gamma}/\dot{\sigma}$ ). The overall navigation loop gain is called the effective navigation ratio and is equal to:

$$\Lambda = ( \dot{\gamma}/\dot{\sigma} ) \frac{V_m}{V_c} \tag{3.1}$$

where

$V_m$ = missile speed

$V_c$ = closing rate

(It should also be mentioned that the "optimal control derivation" of pro nav results in a time-varying $\Lambda$ .) In most missiles, normal acceleration ($a_c$) is commanded instead of turning rate. Since $a_c = V_m\dot{\gamma}$, then

$$a_c = \Lambda V_c \dot{\sigma} \tag{3.2}$$

In many missiles a constant gain is used,

$$a_c = K\dot{\sigma} \text{ , where } K \triangleq \Lambda V_c \text{, assumed to be a constant.} \tag{3.3}$$

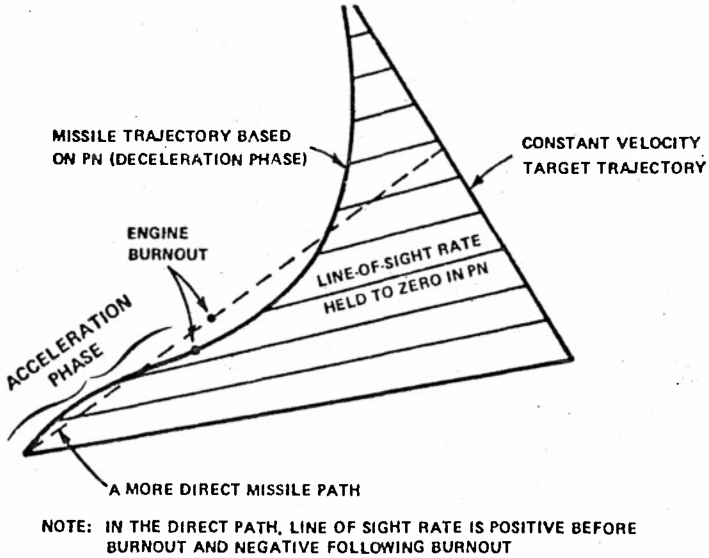


**FIGURE 3-1 PROPORTIONAL NAVIGATION GUIDANCE AND A MORE DIRECT APPROACH PATH**

In this case the effective navigation ratio will vary with the closing velocity. Studies have shown that $\Lambda = 4$ is a good compromise for low miss distance (high $\Lambda$) and small time-to-go before instability occurs. To be stable, $\Lambda$ must be greater than two, but the greater $\Lambda$ the sooner the homing loop goes unstable (higher time-to-go). Because of these effects the constant gain K must be carefully selected. In practice, the gain is picked for a tail, beam, or head-on chase and the performance degrades for other cases. Radar missiles can directly measure $V_c$ and are therefore able to keep the effective navigation ratio constant in spite of varying target aspect. Another scheme uses filter theory to provide an estimate of $V_c$ based upon LOS rate and inertial body motion (rates). This method could be applicable to passive seekers.

There have been several attempts to combine the good features (while simultaneously eliminating the bad ones) of proportional navigation and pursuit guidance into an overall composite guidance law. One such approach is to compute guidance signals based on both laws, provide a time varying weighing factor for each, and sum the result. Such an application usually weights pursuit guidance heavily at long ranges where the noise problem is most severe and the accuracy requirements less severe. Of course, a knowledge of time-to-go or range is required.

Dynamic lead guidance provides results similar to the weighting technique but for different reasons. At small line-of-sight rate frequencies (which typically occur at long ranges) the guidance law behaves like pursuit guidance, at large line-of-sight rates (which typically occur at short ranges) it behaves like pro nav. The advantage is that no estimate of range or time-to-go is necessary; the behavior transitions "automatically" based upon the frequency of the input signal. It also has the advantage of better performance in atypical situations (e.g., large line-of-sight rates at <u>long</u> ranges). However, stability problems can occur if significant noise is still present when the guidance law transitions to a pro-nav type behavior.

Table 3.2 summarizes the advantages and disadvantages of each of these guidance laws when they are used in combination with classical low pass noise filters. A law called command-to-line-of-sight is also included although, strictly speaking, it is not a terminal guidance law beacuse it requires no terminal missile seeker. The launch aircraft merely tracks the target, tracks or computes the missile position, and sends commands to the missile to guide it along the launch-aircraft to target-aircraft line-of-sight.

| GUIDANCE LAW | ADVANTAGES | DISADVANTAGES |
|---|---|---|
| 1. Command-to-Line-of-Sight Guidance | No terminal seeker required | Very inaccurate against moving targets and with winds. Data link required |
| 2. Pursuit | Noise insensitive Easy to use with strapdown seekers | Inaccurate against moving targets and with winds |
| 3. Proportional | Accurate against constant velocity targets. | Inaccurate against accelerating targets Stability is sensitive to noise. |
| 4. Pursuit + Pro Nav | Between 2 and 3 in terms of accuracy | Between 2 and 3 |
| 5. Dynamic Lead | Between 2 and 3 in terms of accuracy. Easy to use with strapdown seekers. | Between 2 and 3. Stability problems if transition to pro nav occurs when significant noise is present. |

Table 3.2 Comparison of Classical Guidance Laws

As mentioned already, the autopilot performs the function of translating the guidance law command from the "brain" into some signal which the "muscle" can understand. In general, this translation depends upon the aerodynamic and kinematic properties of the airframe and the physical properties of the surrounding air mass. For example, a 20g lateral acceleration command for a high lift missile going Mach 2 at 10,000 ft requires a much different actuator command than for a lower lift missile going Mach 1 at 20,000 ft.

The reliance on classical control techniques in autopilot design usually results in an autopilot with three independent channels for yaw, pitch, and roll. These three motions are assumed uncoupled because classical control techniques are in general limited to single input, single output linear systems (their extension to multi input, multi-output systems is quite complex). In flight, inherent aerodynamic interactions comprise coupling modes between steering and roll motions. Therefore, the channels of the autopilot are not independent and this leads to stability problems. The cross-coupling stability problem gets worse with increasing angle-of-attack. To partially decouple the roll and steering control systems, autopilot designers limit the steering response speed so that the roll system bandwidth is two to four times the steering system bandwidth. Also, the designers limit the angle-of-attack missile can use.

The autopilot gains in each of the channels are often variable. This variation is required to produce the optimum performance for different Mach numbers, dynamic pressures and control effectiveness. Two approaches have been used to vary the gains. In the first, the gains are scheduled based on Mach number, density and possibly other states. In the second, a high frequency dither signal is used to converge the missile gains to desired values.

To simplify our discussion, we will assume that the guidance law and autopilot are designed independently. Not only is this assumption not necessary, but better guidance laws can be designed if the autopilot characteristics are included in the guidance law derivation. To do so, however, makes the problem too vehicle dependent which in turn further dilutes generality. In addition, autopilot design and mechanization techniques are now available which result in very good guidance law command execution, regardless of the airframe or guidance law characteristics.

## 4. ADVANCED GUIDANCE CONCEPTS

A review of Table 3.2 quickly reveals that there is no perfrect guidance law. Even the most accurate one (proportional navigation) is susceptible to noise and accuracy degradation against accelerating targets. (It also has poor performance in large off-boresight angle launches for short range air-to-air missiles. The off-boresight angle is the angle between the launch aircraft's velocity vector and the line-of-sight from launch aircraft to target aircraft when the missile is fired.) It is this quest for more accurate missile performance at lower cost that has prompted the initiation of our research and technology programs in advanced guidance and control theory.

The research program is concentrating on the application and extension of modern control and estimation theory to tactical missiles. Figure 4-1 is a functional comparision of how our current modern approach differs from the classical approach to missile guidance and control. The classical techniques of using low pass filtering to attenuate the noise inherent in the guidance signal and using proportional navigation to steer the missile toward the target were well developed before the advent of modern control and estimation theory. They have become firmly entrenched in guided missile designs because they worked well in the rather benign environment of past air-to-air engagements and were easily implemented with analog circuitry. Because of such precedence, missile designers have often tried to satisfy the increased performance requirements of modern day air-to-air missiles by increasing the complexity of associated hardware such as seekers, gyros, accelerometers, airframes, and engines. Such approaches in many cases have improved performance, but the resulting cost has often been so high that the systems were never developed for operational use or were purchased in small quantities.

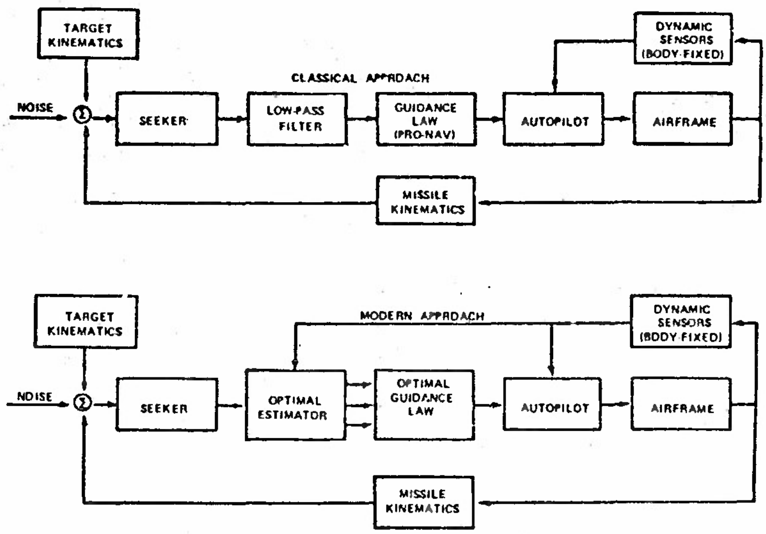


**FIGURE 4-1 FUNCTIONAL COMPARISON BETWEEN CLASSICAL AND MODERN GUIDANCE AND CONTROL**

During the late 1960's and early 1970's, a few missile designers did take a cursory look at applying the modern control theory developed during the late 1950's and early 1960's to tactical missiles. Basically, such an approach would replace the low pass filter with an optimal estimator such as the Kalman filter. In theory, this would allow one to "optimally" separate the signal from the noise by using information about the missile dynamics and noise covariances rather than filtering based only on frequency content. In addition, missile/target states other than line-of-sight rate could be estimated, even if not measured, provided they were mathematically observable. This, in turn, would allow one to design more advanced guidance laws based upon optimal control theory, because such theory usually requires complete information concerning the missile states.

Unfortunately, the conclusion reached at that time was that, except in the most simplistic and unrealistic cases, the mechanization of such algorithms in real time on board a small tactical missile was unfeasible because most of the calculations involved procedures which could not be accomplished efficiently with analog circuitry. Thus, control engineers continued to design missiles based upon the classical "intuitive" concepts.

Fortunately, several things have changed since the early 1970's to give us renewed optimism. First, new theories have appeared and old ones have been extended and refined. Second, several new numerical techniques for solving complex equations have been developed, and such techniques require fewer and less time-consuming calculations. Finally, and probably most important, we have witnessed the birth of the microcomputer. This revolution now allows us to perform more calculations, more often, more accurate, at less cost, and in a smaller volume than anyone would have imagined just ten years ago.

In October of 1977, the United States Air Force Armament Laboratory initiated a basic and applied research program to investigate and extend those modern control and estimation techniques that have potential application for improving short range air-to-air missile performance. Figure 4-2 outlines in broad generalities the major areas of investigation. During the first 18 months of the program, the problem was formulated using standard textbook theories, such as Linear Quadratic, Linear Quadratic Guassian, and Extended Kalman Filtering. The results of this initial investigation provided a theoretical baseline by uncovering the deficiencies of these standard approaches when applied to the short range air-to-air missile scenario and by comparing the resulting performance with classical techniques such as proportional navigation and low pass filtering. The overall conclusions showed that a simplistic and straight-forward application of modern control theory results in very little performance improvement.

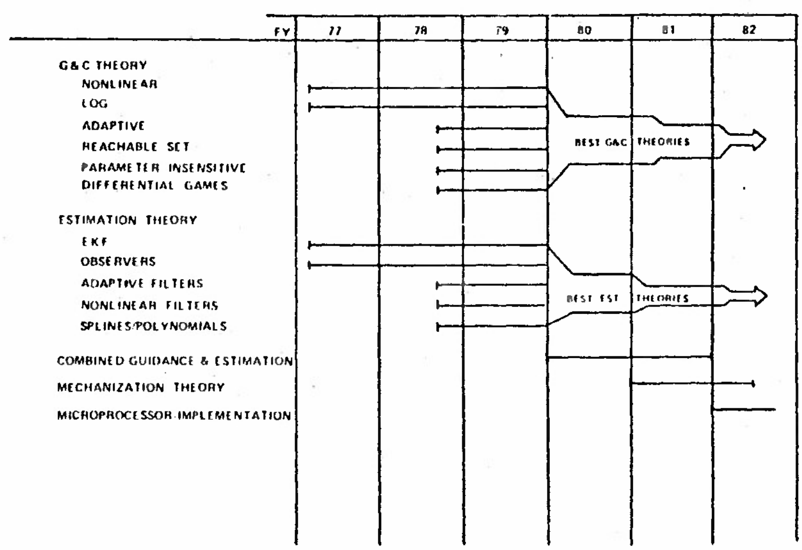


FIGURE 4-2 MAJOR TOPICS OF INVESTIGATION FOR USAF G&C RESEARCH FOR TACTICAL MISSILES

But this initial investigation did accomplish a rather detailed evaluation of those theoretical aspects which need further development and those practical aspects which are important in increasing missile performance. We have now extended our study into several new areas, many of which have never been examined in any great detail. Time and space limitations will not allow me to elaborate on all the methodologies listed in figure 4-2. (See references 1 and 2 for more details on all these techniques, plus many more. Reports on preliminary results from our study will be available in early 1980.)

My approach will be to review the highlights of nonlinear and linear optimal control theory, pointing out the general advantages and disadvantages of the theory for our particular application. Then I will apply the linear quadratic (LQ) theory to a rather simple air-to-air missile control problem. The purpose of this is three-fold. First, it illustrates the procedural techniques for applying such theory. Second, it provides an excellent framework for discussing the advantages and disadvantages of various assumptions which can be and often are made. Third, it vividly illustrates the limitations of proportional navigation by showing that pro nav is but a special case of LQ theory based on some rather restrictive assumptions. Finally, I will briefly describe the role of optimal estimation in missile guidance and control, showing how the estimation and control problem are actually inseparable under most conditions.

## 5. GENERAL OPTIMAL CONTROL THEORY

Before I give an example of how optimal control theory can be applied to the derivation of tactical missile guidance laws, I would like to review some of the more salient features of this theory. Time and space limitations will not allow me to examine the theory in its most general form, nor will it allow me to present it in any mathematically precise detail. (See reference 3 for a complete treatment.) However, I do intend for this presentation to highlight its usefulness and its limitations.

Consider a dynamical system represented by the following set of nonlinear differential equations:

$$\underline{\dot{x}} = \underline{f}\,(\underline{x},\ \underline{u},\ t), \quad \text{where} \tag{5-1}$$

$\underline{x}\ \Delta$ state vector of the system,

$\underline{\dot{x}}\ \Delta$ the time derivative of the state vector,

$\underline{u}\ \Delta$ the system control vector input, and

$\underline{f}(\cdot)\ \Delta$ a vector function whose components are nonlinear functions of the state and control vector components and of time.

Such a system may also be subject to terminal equality constraints of the form

$$\underline{\psi}(t_i, t_f, \underline{x}_i, \underline{x}_f) = 0 \quad \text{where} \tag{5-2}$$

$t_i\ \Delta$ the initial time

$\underline{x}_i\ \Delta$ the initial state,

$t_f\ \Delta$ the final time,

$\underline{x}_f\ \Delta$ the final state, and

$\underline{\psi}(\cdot)\ \Delta$ a vector function whose components are nonlinear functions of the initial and final state vector components and the initial and final times.

The theory can also handle inequality constraints on both the control vector and state vector, but this generality will be omitted here for the sake of brevity, even though it is an important consideration in practical applications.

Now the optimal control problem can be stated as follows: Select a control vector $\underline{u}(t)$, for $t_i \le t \le t_f$, such that we minimize some performance index (or sometimes referred to as a cost functional) of the form:

$$PI = g(t_i, t_f, \underline{x}_i, \underline{x}_f) + \int_{t_i}^{t_f} L(t, \underline{x}, \underline{u})\,dt, \quad \text{where} \tag{5-3}$$

$g(\cdot)\ \Delta$ a scalar function of the terminal times and states and

$L(\cdot)\ \Delta$ a scalar time-varying function of the state and control vectors from $t_i \le t \le t_f$.

Let us pause for a moment and consider the generality of the problem.

(1) It includes *any* system that can be represented by a set of nonlinear time-varying differential equations. (It can also be applied to systems represented by difference equations, but we will not consider these here.)

(2) The system and controls can be subject to a large class of equality or inequality constraints.

(3) The performance index includes both initial and final conditions, plus the time history of the control and state vectors. Note also that the performance index is more flexible than it appears upon first consideration. For example, if we wish the system state to follow some predescribed state reference trajectory $\underline{x}_r(t)$, then our performance index might be

$$PI = \int_{t_i}^{t_f} [\underline{x}(t)-\underline{x}_r(t)]^T A(t) [\underline{x}(t)-\underline{x}_r(t)] dt$$

where A(t) is some time-varying weighting matrix, which allows us to choose what portions of the state trajectory we feel are most important in terms of being "close" to the reference trajectory.

If one determines that the only important performance objective is to force $\underline{x}_f$ to be as "close" to some $\underline{y}_f$ as practicality allows, then we might choose

$$PI = (\underline{x}_f-\underline{y}_f)^T \; A(\underline{x}_f-\underline{y}_f) + \int_{t_i}^{t_f} \underline{u}(t)^T B(t)\underline{u}(t)dt$$

The integral term is included to add realism, since omitting it will result in a mathematical solution which will require an "infinite" control. (An alternative approach could be to omit the integral term and place an inequality constraint on $\underline{u}(t)$ instead.)

Although this formulation has tremendous generality, the practical disadvantages become evident when we examine the solution. There are many representations of the solution, all of which of course give the same answer. Perhaps the most popular representation is in terms of the Hamiltonian. Define the following quantities:

$H(t, \underline{x}, \underline{u}, \underline{\lambda}) \triangleq L(t, \underline{x}, \underline{u}) + \underline{\lambda} \;\; \underline{f}(t, \underline{x}, \underline{u})$, where H is called the Hamiltonian and $\underline{\lambda}$ is the vector of Lagrangian multipliers so often used in the calculus of variations;

$G(t_i, t_f, \underline{x}_i, \underline{x}_f, \underline{\nu}) = g(t_i, t_f, \underline{x}_i, \underline{x}_f) + \underline{\nu}^T \underline{\psi} (t_i, t_f, \underline{x}_i, \underline{x}_f)$, where $\underline{\nu}$ is also a vector of Lagrangian multipliers.

It can be shown that the solution to the problem stated in (5-1) through (5-3) is given by

$$\dot{\underline{\lambda}} = -\frac{\partial H}{\partial \underline{x}} \tag{5-4}$$

$$\dot{\underline{x}} = \frac{\partial H}{\partial \underline{\lambda}} \tag{5-5}$$

$$\frac{\partial H}{\partial \underline{u}} = \underline{0} \tag{5-6}$$

$$\frac{\partial G}{\partial \underline{x}_i} = -\underline{\lambda}\Big|_{t_i} \tag{5-7}$$

$$\frac{\partial G}{\partial \underline{x}_f} = \underline{\lambda}\Big|_{t_f} \tag{5-8}$$

$$\frac{\partial G}{\partial t_i} = H\Big|_{t_i} \tag{5-9}$$

$$\frac{\partial G}{\partial t_f} = -H\Big|_{t_f} \tag{5-10}$$

Now let us examine a typical solution procedure in order to illustrate how difficult the solution can be in general.

Step 1: Solve eq. (5-6) for $\underline{u}(t)$.

Step 2: Solve eq. (5-5) for $\lambda(t)$. Note that, in general, this involves the solution of nonlinear differential equations, which may or may not have an analytical closed-form solution. Also note that this differential equation may be coupled with eq. (4-5), another nonlinear differential equation.

Step 3: Substitute the solution for $\lambda(t)$ from step 2 into the solution for $\underline{u}(t)$ in step 1. Then substitute this form of $\underline{u}(t)$ into (5-5).

Step 4: Solve eq. (5-5) for $\underline{x}(t)$. This is also a nonlinear differential equation which might be coupled to eq. (5-4).

Step 5: Note that the solution to $\underline{x}(t)$ and $\lambda(t)$ involves 2n unknown constants, where n is the dimension of the state vector. Use all given initial and final conditions for $\underline{x}(t)$ along with the solutions to eqs. (5-7) through (5-10). This should result in 2n equations in 2n unknown constants, which in theory can be solved completely.

It should be obvious by now that there are very few conditions under which closed form solutions for $\underline{u}(t)$ exist. In general, complex numerical techniques must be employed, involving a large amount of data and numerous calculations. There are two other disadvantages to this formulation which should be noted. First, the solution is initial and/or final condition dependent. Hence, for each launch condition and target maneuver in an air-to-air missile engagement, the solution must be completely re-calculated. Also note that, in general, the solution for the optimal control depends only on time. This is what we refer to as an open-loop solution since it does not depend directly upon the missile state $\underline{x}(t)$. (If it did, we would refer to it as a closed-loop or feedback solution.) This fact has severe consequences in pratical solutions, since the actual state trajectory will in general diverge from the optimal one if there is any error in our dynamic model (eq 5-1).

Before we leave the general nonlinear theory, I want to discourage any pessimism concerning its potential application to future tactical missiles. Recent studies (Reference 4) which we have conducted show that, when advanced numerical techniques are used in combination with the computational power of modern microprocessors, reasonable solutions can be obtained for somewhat simplified nonlinear formulations. In addition, a feedback solution can be approximated by re-solving the problem at appropriate time intervals in real-time on-board the missile. Although such a solution would not be the optimal one from launch to fuzing, it does offer significant advantages over proportional navigation and may be the only approach when the system involves significant non-linearities. The disadvantages of the general non-linear theory led researchers in the early 1960's to search for less general, but more tractable, formulations of the optimal control problem. The result was Linear Quadratic Theory, some features of which I will discuss now.

## 6. LINEAR QUADRATIC THEORY

Linear Quadratic Theory is a subset of the general nonlinear optimal control theory. The key elements in the formulation are the same: a dynamical system model, a performance index (or cost functional), and appropriate constraints. The difference in formulation lies in the fact that the dynamical system model must be linear, the cost functional must be quadratic in nature, and only a limited set of constraints are allowed. The linearity assumption is the most severe for air-to-air missiles. Nonlinear aerodynamics, nonlinear equations of motion, and nonlinear kinematics are prevalent in air-to-air missile engagements. The nature of these assumptions will become more obvious when we work a simplified example later.

The limited nature of the allowable constraints are somewhat less of a problem. Two of the more important constraints (terminal state $\underline{x}_f = \underline{0}$ and $\underline{u}(t) \leq u_{max}$) are still allowable. The problem of allowing only quadratic cost functionals is usually workable. This is primarily because we are still allowed to use a time varying weighting matrix and most intuitively reasonable costs are of a quadratic (or positive definite) nature.

There are several techniques available for applying this linear theory to nonlinear systems. Some of the most common ones are:

(a) Ignore the nonlinearities by postulating what seems to be a reasonable linear model of the nonlinear system and hope that this will not significantly decrease the overall optimality of the solution (See Reference 6).

(b) Compute some optimal nominal trajectory using nonlinear theory. Then linearize the nonlinear system equations about this nominal trajectory, using small perturbation theory. Apply the optimal linear theory to the linear perturbation equations. This will result in two control functions, one for the nominal trajectory ($\underline{u}_N(t)$) and one for the perturbation trajectory ($\underline{u}_L(t)$). One disadvantage of this approach is that it forces the missile trajectory to follow the optimal nonlinear trajectory for our model, and this trajectory may be far from the true optimal trajectory for the actual missile. Another drawback is that the optimal nominal trajectory is a function of initial conditions. Hence, one either has to compute a new optimal nominal trajectory for each launch condition (using the complex solution process outlined in section 5) or contend that the differences in optimal nominal trajectories for various launch conditions are unimportant in the overall optimality of the solution (See Reference 7).

(c) Linearize the nonlinear equations about the current value of the state vector and re-solve the linear problem on-line at various points along the trajectory. This technique will usually cause the solution to "forgive mistakes" made in the past due to invalid linearity assumptions (See Reference 8).

References 1 and 2 should be consulted for other examples of applying Linear Quadratic Theory to nonlinear systems. There are two major deficiencies associated with all these methods. First, there is no apriori analytical global method of determining how much we sacrifice in optimal performance (i.e., how much does the performance index increase) when we use these approximations. The only real way to evaluate this is through extensive computer simulations. (All promising techniques studied in our research program are evaluated using extensive and detailed computer simulations.) Second, and more important, there are not even analytical methods available to ascertain whether or not the solutions remain stable! (This is not exactly true. There are a few special types of nonlinearities for which analytical methods have been developed to ascertain stability. See Reference 5).

If linear theory has all these drawbacks, why do we continually pursue it? The answer lies in the eloquence of its properties and relative ease of implementation. To illustrate this, I will now formulate the problem and discuss the solutions. (See References 3 or 5 for a complete treatment.)

Let our linear dynamical system be represented by:

$$\dot{\underline{x}}(t) = F(t)\,\underline{x}(t) + G(t)\,\underline{u}(t), \tag{6-1}$$

where F(t) and G(t) are matrices and the rest of the notation is the same as that used in section 5.

Our quadratic performance index is denoted by

$$PI = \underline{x}_f^T A\, \underline{x}_f + \int_{t_i}^{t_f} \left[\underline{x}^T(t)Q(t)\underline{x}(t)+\underline{u}^T(t)R(t)\underline{u}(t)\right] dt, \tag{6-2}$$

where R(t) is a symmetric positive definite matrix, Q(t) is a symmetric semi-positive definite matrix, and A is semi-positive definite.

Although certain constraints can be applied to the state and control vectors, they are too specialized to examine here. Minimization of the performance index (6-2) results in

$$\underline{u}(t,\underline{x}) = -R^{-1}(t)G^T(t)P(t,t_f)\underline{x}(t), \tag{6-3}$$

where $P(t,\ t_f)$ is found by solving the matrix Ricatti nonlinear differential equation

$$-\dot{P}(t)=P(t)F(t)+F^T(t)P(t)-P(t)G(t)R^{-1}(t)G^T(t)P(t)+Q(t), \tag{6-4}$$

subject to $P(t_f,t_f) = A$.

Sometimes it is possible to make additional assumptions to simplify eq. (6-4). For example, if F, G, Q, and R are constant matrices and $t_f \to \infty$, eq (6-4) reduces to the algebraic Ricatti equation given by

$$PF+F^TP-PGR^{-1}G^TP+Q = [0] \tag{6-5}$$

The solution to equation (6-4) is not immediately obvious, but several techniques have been developed throughout the years to solve it, many of which are extremely efficient on a digital computer. Equation (6-5) is even easier to solve on a digital computer. The solution for $\underline{u}(t)$ in eq (6-3) has several attractive properties. The most important ones for our application are discussed below:

(a) Note that the solution for $\underline{u}(t,\ \underline{x})$ and P(t) are <u>independent</u> of $\underline{x}_i$ or $\underline{x}_f$. This is extremely important because it means that the problem need be solved only once (off-line) and this solution will be valid for all initial and final conditions. This was not the case for the nonlinear theory.

(b) $\underline{u}(t,\ \underline{x})$ is a function of the system state $\underline{x}(t)$. The fact that $\underline{u}(t,\ \underline{x})$ is a <u>feedback</u> control law means that it is less sensitive to noise, external disturbances, and modeling errors. Such a property is called <u>robustness</u> in the literature.

(c) Let $\underline{u}(t,\ \underline{x}) = K(t)\underline{x}(t)$, where $K(t) = -R^{-1}(t)G^T(t)P(t,\ t_f)$. K(t) is called the <u>control gain</u>. All the information needed to determine K(t) can be computed off-line and stored in a missile computer. Furthermore, if F, G, Q, and R are constant and $t_f \to \infty$, K becomes a constant. However, as the true missile system is not linear, if we use the on-line linearization technique discussed previously, we must compute a new K for each new value of F, G, Q, and R.

Besides the general disadvantages already noted at the beginning of this section for linear theory, there are two others which deserve mentioning. First the solution depends on a good choice for $t_f$. At first one might argue that $t_f$ is a "free" parameter, subject to the designers selection. In theory this is true, but in practice $t_f$ really determines how good our solution is. A review of eq (6-2) reveals that the choice of $t_f$ not only affects the minimum value of PI but also drives the optimal trajectory solution and the final state $\underline{x}_f$. In the air-to-air missile problem, selecting a given value of $t_f$ in effect determines the terminal miss distance for a given launch condition. If the true objective is to minimize terminal miss distance, then the problem now becomes one of selecting the "optimal" $t_f$ which results in the minimum miss distance! In effect, we have the freedom to <u>select</u> the missile time of flight from launch to intercept. The problem now becomes one of selecting <u>both</u> the $\underline{u}(t,\ \underline{x})$ <u>and</u> the $t_f$ which will result in the smallest value of PI.

The other disadvantage of the linear theory is the requirement for a real-time knowledge of $\underline{x}(t)$, the relative target/missile state. Since our missile model is only a crude linear approximation and since we have no definite knowledge of future target manuevers, $\underline{x}(t)$ must be determined on-board the missile. Current sensors provide an estimate of only a few missile states. To increase the quantity and quality of our missile sensors would also add significant cost.

An alternative approach is to use optimal estimation theory to extract the mathematically observable states from the limited measurement data. But before we examine this topic, let us focus further on the advantages and disadvantages of Linear Quadratic Theory by discussing a simple example.

## 7. ILLUSTRATIVE EXAMPLE

More detailed information concerning the application of LQ theory to tactical missiles can be found in references 6 and 8. The example which I will use here was selected primarily for its tutorial merits. Consider the engagement scenario and terminology outlined in figure 7-1. Let M be the missile and let $\underline{r}_m$, $\underline{v}_m$, and $\underline{a}_m$ be the missile's position, velocity, and acceleration vectors relative to an inertial reference frame. Let T be the target and let $\underline{r}_T$, $\underline{v}_T$, and $\underline{a}_T$ be the target's position, velocity, and acceleration vectors relative to the same inertial reference frame. Now define the following quantities:

$x_1 \triangleq$ the target/missile relative position in the y direction $(x_1=r_{T_y}-r_{M_y})$

$x_2 \triangleq$ the target/missile relative position in the z direction $(x_2=r_{T_z}-r_{M_z})$

$x_3 \triangleq$ the target/missile relative velocity in the y direction

$x_4 \triangleq$ the target/missile relative velocity in the z direction

$\dot{x}_3 \triangleq$ the target/missile relative acceleration in the y direction $(\dot{x}_3=a_{T_Y}-a_{M_Y})$

$\dot{x}_4 \triangleq$ the target/missile relative acceleration in the z direction $(\dot{x}_4=a_{T_z}-a_{M_z})$

$\sigma \triangleq$ LOS (line-of-sight) angle relative to the y axis of the inertial reference frame

$R \triangleq$ scalar distance from missile to target (the line-of-sight).

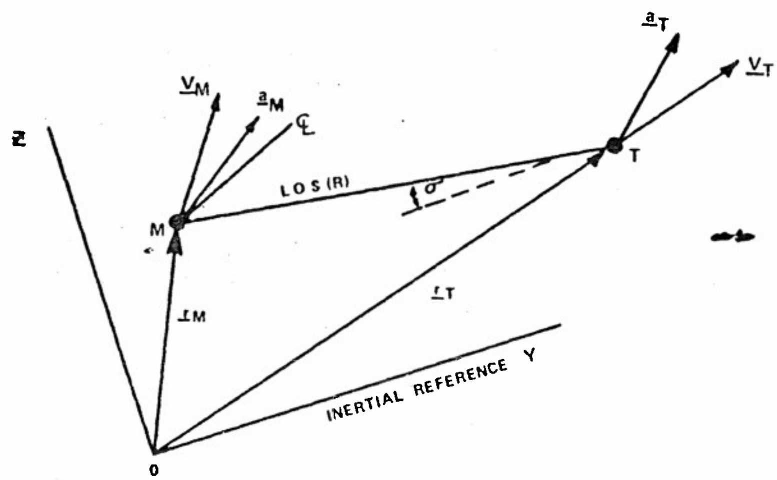


FIGURE 7-1 SIMPLIFIED PLANAR AIR-TO-AIR ENGAGEMENT

Straight-forward kinematic relationships give us the following linear model for our dynamical system:

$$
\begin{aligned}
\dot{x}_1 &= x_3 \\
\dot{x}_2 &= x_4 \\
\dot{x}_3 &= a_{T_Y}-a_{M_Y} \\
\dot{x}_4 &= a_{T_z}-a_{M_z}
\end{aligned}
\qquad (7\text{-}1)
$$

Throughout this example I will be making numerous assumptions, not necessarily because they must be made in order to solve the problem, but rather because I want to terminate this derivation with a guidance law that resembles proportional navigation. In doing so, I will have explicitly stated all the assumptions which are involved in claiming that proportional navigation is the optimal guidance law which minimizes terminal miss distance.

Assumption #1: The engagement takes place in one plane. This assumption is made solely to simplify our derivation.

Assumption #2: Let $\underline{a}_T = \underline{0}$. This implies that the target has constant velocity (both magnitude and direction). Of course, this is far from true in air-to-air missile engagements.

Assumption #3: Let the control vector be the missile's inertial acceleration vector ($\underline{u} \triangleq \underline{a}_M$). This assumption has far reaching consequences. It effectly says that we have complete and immediate control over all three acceleration components of the missile. Since low cost throttable short range air-to-air missile engines have not yet been developed, we actually have little control over the missile's acceleration component in the direction of its centerline. The acceleration in this direction equals the thrust of the missile minus its axial drag. The thrust is usually designed to maximize missile velocity early in flight so that the time for target evasive maneuvers is minimized. The drag is usually uncontrollable once the missile has been designed.

We can control the missile acceleration components perpendicular to its centerline. However, this cannot be accomplished instantaneously as required by our model. For aerodynamic control, the airframe must undergo rotations to produce the proper angle-of-attack which in turn results in normal forces, the magnitude of which are controlled through a feedback loop using accelerometers which measure the actual normal accelerations. In effect, our model has neglected the rotational and translational inertial properties of the missile and have assumed a perfect control loop. It should be mentioned, however, that recent classical techniques have resulted in some very effective and efficient control loops for autopilot design. This is one major reason we have chosen to decouple our guidance law studies from autopilot considerations (reference section 2).

We can now write eqs (7-1) as

$$\begin{aligned} \dot{x}_1 &= x_3 \\ \dot{x}_2 &= x_4 \\ \dot{x}_3 &= u_1 = -a_{M_y} \\ \dot{x}_4 &= u_2 = -a_{M_z} \end{aligned} \qquad (7\text{-}2)$$

Or, in vector notation,

$$\begin{aligned} \dot{\underline{x}} &= F\underline{x} + G\underline{u} \\ &= \left[\begin{array}{c|c} 0 & I \\ \hline 0 & 0 \end{array}\right] \underline{x} + \left[\begin{array}{c} 0 \\ \hline I \end{array}\right] \underline{u}, \end{aligned} \qquad (7\text{-}3)$$

where $I \triangleq$ a 2x2 identity matrix.

Let our performance index (see section 6) be

$$PI = \underline{x}_f^T A \underline{x}_f + 1/2 \int_{t_i}^{t_f} \underline{u}^T R \underline{u}\, dt \qquad (7\text{-}4)$$

where $A = \left[\begin{array}{c|c} I & 0 \\ \hline 0 & 0 \end{array}\right]$ and $R = \begin{bmatrix} b & 0 \\ 0 & b \end{bmatrix}$.

This reduces to the relationship

$$PI = x_1^2(t_f) + x_2^2(t_f) + 1/2b \int_{t_i}^{t_f} (u_1^2 + u_2^2)dt \qquad (7\text{-}5)$$

OBSERVATION: Our performance index is designed to minimize terminal miss distance and the integral of the control effort over the flight duration. If b is small, we are willing to expend whatever acceleration is required to minimize terminal miss. (Of course, our real missile must be capable of producing and sustaining such accelerations.) If b is large, we will in effect limit the magnitude of acceleration available to achieve small miss distances. In other words, we are free to choose how much we wish to "pay" for terminal accuracy, since acceleration capability is related to monetary cost.

From section 6 we have

$$\underline{u}(t,\underline{x}) = -R^{-1}G^T P(t,T)\underline{x}(t) = -1/b \left[\begin{array}{c|c} 0 & I \end{array}\right] P(t,T)\underline{x}(t) \text{ and} \qquad (7\text{-}6)$$

$$-\dot{P} = PF + F^T P - PGR^{-1}G^T P + Q = P\left[\begin{array}{c|c} 0 & I \\ \hline 0 & 0 \end{array}\right] + \left[\begin{array}{c|c} 0 & 0 \\ \hline I & 0 \end{array}\right] P - P \left[\begin{array}{c} 0 \\ \hline I \end{array}\right] {}^1/b \left[\begin{array}{c|c} 0 & I \end{array}\right] P \ . \qquad (7\text{-}7)$$

The above equations can be solved analytically to produce

$$\underline{u}(t) = \begin{bmatrix} -\dfrac{3t_g}{3b+t_g^3} & 0 & -\dfrac{3t_g^2}{3b+t_g^3} & 0 \\ 0 & -\dfrac{3t_g}{3b+t_g^3} & 0 & -\dfrac{3t_g^2}{3b+t_g^3} \end{bmatrix} \qquad (7\text{-}8)$$

where $t_g \triangleq t_f - t$.

Assumption #4: Let $b = 0$. This means that we have a missile that can exert unlimited control (if necessary) and we are willing to pay the cost.

Equation (7-8) now becomes

$$u_1(t) = -\frac{3}{t_g^2} x_1 - \frac{3}{t_g} x_3 \qquad (7\text{-}9)$$

$$u_2(t) = -\frac{3}{t_g^2} x_2 - \frac{3}{t_g} x_4 \qquad (7\text{-}10)$$

Reverting back to figure 7-1 we can establish the following relationships:

$$\begin{aligned} x_1 &= R\cos\sigma \\ x_2 &= R\sin\sigma \\ x_3 &= \dot{x}_1 = -R\dot{\sigma}\sin\sigma + \dot{R}\cos\sigma \\ x_4 &= \dot{x}_2 = R\dot{\sigma}\cos\sigma + \dot{R}\sin\sigma \end{aligned} \qquad (7\text{-}11)$$

Assumption #5: First, choose the inertial y-direction such that $\sigma$ remains small over the entire trajectory. Note that, once the inertial coordinate system has been selected, we cannot change its orientation during flight, since this would violate our inertial requirement. Hence, our assumption is based on the hope that the engagement scenario will not result in large changes in $\sigma$. Our small angle approximation becomes quite questionable if $\sigma$ varies more that a total of $30^\circ$ throughout the engagement. ($\cos 15^\circ = .97$ and $\sin 15^\circ = 0.26$.) This assumption is seldom (if ever) stated in similar derivations and has led to much confusion and erroneous conclusions.

Equations (7-11) now become

$$\begin{aligned} x_1 &= R \\ x_2 &= R\sigma \\ x_3 &= -R\dot{\sigma}\sigma + \dot{R} \\ x_4 &= R\dot{\sigma} + \dot{R}\sigma \end{aligned} \qquad (7\text{-}12)$$

Assumption #6: Let $t_g = -R/\dot{R}$. This is the least understood assumption among applied researchers. The rationale ususally given for it is based on the additional assumption that the relative velocity component along the LOS remains constant. This is because $\dot{R} = \dfrac{-R}{t_f - t}$ for all t if and only if $\dot{R}$ is constant. Remember, $t_g$ must be accurate for all t if our solution is truly optimal. However, as I have stated earlier, $t_f$ (and thus $t_g$) is actually a design parameter! The time history of $\dot{R}$ depends on $t_f$; $t_f$ should not be chosen based upon the assumption that $\dot{R}$ is constant! If it is, then the opitmal solution will simply force the missile to fly a trajectory such that $x_f$ is minimized when $t = t_f$, but we have no guarantee that this is the best $t_f$ to result in the smallest $x_f$.

Assumption #7: Let $u_1 = 0$. Note that $u_1$ is the control along the y direction. If $\sigma$ is small, and if the missile thrust vector is directed primarily along the LOS, we may as well assume $u_1 = 0$ since we can't control the thrust anyway. However, we should realize that this is a poor assumption for two reasons. First, air-to-air engagement scenarios do not usually result in thrust directed primarily along the LOS, especially if we have a large initial boresight angle. Secondly, the optimal control law requires that we control acceleration in this direction. If we don't, optimality will suffer.

Substituting $t_g = -R/\dot{R}$ and eqs (7-12) into (7-10) results in

$$u_2(t) = -3\left(\frac{\dot{R}}{R}\right)^2 R\sigma + 3\left(\frac{\dot{R}}{R}\right)(R\dot{\sigma} + \dot{R}\sigma) = 3\dot{R}\dot{\sigma}$$

Our final guidance law now reduces to

$$\underline{u}(t) = \begin{bmatrix} u_1(t) \\ u_2(t) \end{bmatrix} = \begin{bmatrix} 0 \\ 3\dot{R}\dot{\sigma} \end{bmatrix} \text{ or} \qquad (7\text{-}13)$$

$$a_{M_z} = -3\dot{R}\dot{\sigma}$$

When we compare this result to the description of proportional navigation in section 3, we find that it is identical to proportional navigation with an effective navigation gain of 3, provided that we assume $a_{\underline{M}}$ is the missile acceleration perpendicular to $\underline{v}_M$. When Assumptions 5 and 7 are valid this is approximately true. In practice, navigation gains of 4 to 5 are commonly used based upon classical control theory analysis.

Figures 7-2 and 7-3 show the effect that assumptions 5 and 6 have on the missile's inner launch boundary. These figures represent the inner launch boundary for a highly maneuverable missile using three different guidance laws. Figure 7-2 is for a $0^{\circ}$ boresight launch and figure 7-3 is for a $40^{\circ}$ off-boresight launch. Guidance law # 1 (G1) uses the simplified calculation for time-to-go (assumption 6). The time-to-go calculation used in G2 is based on an approximation of the relative missile/target acceleration along the line-of-sight. Although this quantity may be difficult to estimate during an actual engagement, these plots do show that significant performance improvements can be achieved by carefully selecting $t_f$, especially in large off-boresight launches. Note also that assumption 5 (small $\sigma$) has little effect for $0^{\circ}$ boresight launches but can significantly degrade performance in large off-boresight launches. This is primarily due to the fact that the assumption is fairly valid for boresight launches at very short ranges.

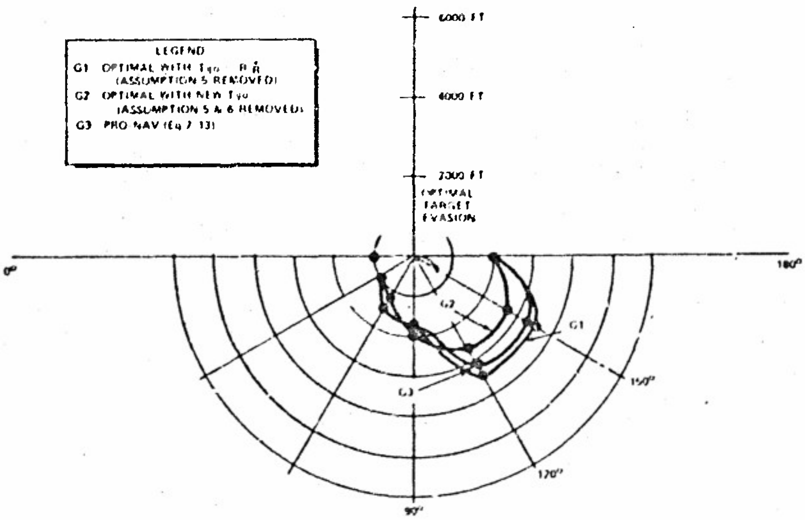


FIGURE 7-2 INNER LAUNCH BOUNDARY FOR THREE GUIDANCE LAWS ($0^{\circ}$ BORESIGHT)

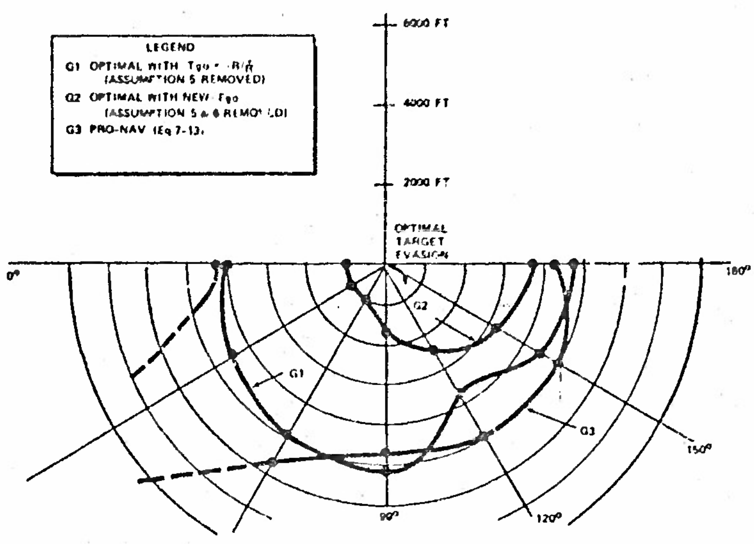


FIGURE 7-3 INNER LAUNCH BOUNDARY FOR THREE GUIDANCE LAWS ($40^{\circ}$ OFF-BORESIGHT)

We have stated all the assumptions in mathematical terms - let's now see what type of engagement scenario is required for them to be valid. Assumptions 2 and 5 require that the target have relatively constant velocity and the missile is launched in a direction close to the line of sight. Assumption 3 requires that the missile have a very good autopilot, and assumption 4 requires that the missile have sufficient manuverability to accomplish all required accelerations. Assumption 6 imposes the additional requirement that the missile's velocity is relatively constant. Assumption 7 is difficult to interpret because it results from events over which we have no control. It is interesting to note that the scenario we have just described is typically the kind encountered in air-to-surface engagements. Thus, for such engagements, proportional navigaton is close to being the best guidance law for minimizing terminal miss distance. Missile designers have realized for years that pro nav works very well in these situations. However everyone of these assumptions is significantly violated in a typical air-to-air engagement.

It should also be mentioned that, in cases where these assumptions are invalid, we should not conclude that the missile will drastically miss the target or go unstable. It is still possible to get acceptable performance in many of these situations. What we can conclude is that, under these conditions, pro nav is not <u>the</u> best guidance law for minimizing terminal miss. We could do better, perhaps much better.

In concluding this section, I would like to mention that a portion of our research program is devoted to alleviating the need to make many of these assumptions. Assumptions 1 and 4 did not have to be made - they were done solely for convenience and the purpose of illustrating proportional navigation. Assumption 2 was not required for mathematical tractability but rather for practical considerations. If we had not assumed that the target acceleration was zero, an additional term would have appeared in our guidance law requiring a continuous knowledge of target acceleration. Since the target is being observed by the missile, and since the missile is not an inertial reference frame, it is impossible to measure this quantity directly. However, we are studying optimal estimation techniques which could allow us to estimate this quantity based upon inputs from other measurements.

Assumption 3 has prompted us to design better autopilots. As I have mentioned before, we are making excellent progress in this area.

Assumption 4 was not strictly necessary, but allowing b>0 would have increased the minimum value of $\underline{x}_f$ since the use of missile acceleration would have increased the performance index. This has led us to design more manueverable airframes which can produce large normal accelerations with little penalty in drag or cost.

Assumption 5 actually involves a linearization technique. Our research is looking at alternative methods of linearization, including such things as making some missile/target relative states the independent variable rather than time.

Assumption 6 is also a subject of our research. It is possible that we should consider a dual coupled optimization problem, one that finds <u>both</u> the optimal $\underline{u}(t)$ <u>and</u> optimal $t_f$ which together minimize $\underline{x}_f$.

Assumption 7 has prompted other Air Force Laboratories to study the design of low cost throttable rocket engines. We ourselves are treating the problem by looking at mathematical techniques which will represent the lack of thrust control as a mathematical constraint in the problem formulation while still allowing us to use LQ theory.

Finally, we are examining the structure of the performance index itself, both in terms of linear and nonlinear theory. Evidence from previous efforts indicate that the minimization of terminal miss distance $(x_1^2 + x_2^2)$ may not be the best approach to achieve the most improvement in missile performance. This is partially due to the effect of other phenomena which aren't being modelled and also a re-examination of our mission objectives. It is difficult to translate subjective mission goals into mathematical equations. There might very well be other states which are a better indicator of terminal miss distance or which would produce better overall results. Certain of these quantities could be included in the integral term with time varying weighting matrices.

## 8. MISSILE AND TARGET STATE ESTIMATION

In discussing the optimal control problem, I briefly mentioned two potential drawbacks associated with applying this theory to tactical missiles. One drawback was the need to have an accurate and current knowledge of the system models. This is true whether we are using the nonlinear or linear theory. Secondly, the linear quadratic theory results in a feedback solution for $\underline{u}(t, \underline{x})$ requiring a complete knowledge of <u>all</u> the states in our system model. Additional assumptions and approximations could reduce this requirement, but the statement is true in general.

Completely accurate system models are never possible, even if we are using the nonlinear theory. The aerodynamic properties of a missile can only be approximated, even if we supplement our experience with extensive wind tunnel and free flight testing. Many of the missile subsystems include unknown nonlinearities and noise characteristics which, at best, can only be modelled by stochastic processes. Even our six-degree-of-freedom equations of motion often include simplifications made for practical considerations. If we choose to linearize our system model in order to apply the linear theory, the model becomes even more inaccurate and could require periodic updating throughout the missile trajectory.

In a small low cost tactical missile, few of the relative target/missile states which are required for a feedback guidance law are directly measurable. Typical sensors on-board such missiles consist of two rate gyros (pitch and yaw), two normal accelerometers, and a roll gyro. Sometimes pitch and yaw attitude gyros and a roll rate gyro are also included, either as additions or replacements for the other sensors. All of these sensors have been used in the past for autopilot rather than guidance law implementation. They also require their own models, including appropriate stochastic models for noise.

Additional state information, of course, is provided by the seeker. This sensor has been the principle source for guidance law information in the past. The primary quantity measured by the seeker is inertial line-of-sight rate; a radar seeker could also provide range rate and range. The seeker is also a dynamical system, and it must be deterministically and stochastically modelled in the same manner as the other sensors. The seeker gimbal angles (angles between the seeker axes and the missile axes) can also usually be measured for little additional cost, but they have seldom been used in the past for guidance law or autopilot implementation. Our current research has shown that these angles contain much valuable information, since they provide an approximation of the missile/target boresignt angle. Recent studies have also indicated that including the target inertial acceleration in our model can also significantly increase performance (see equation 7-1), but currently there are no missile sensors which can directly measure this quantity.

A review of the simple example in section 7 will reveal that one effect of our many assumptions was to reduce the requirement for relative missile/target state information. At the time we claimed such assumptions were made to obtain a closed-form analytical solution, but an equally important motive of applied researchers has been that, without such assumptions, the guidance law would require state information that is simply not measurable. For example, our final solution (pro nav) requires that we measure only $\dot{\sigma}$ and $\dot{R}$, the first of which is always available and the second of which is available from either an active or semi-active RF seeker. However, if we had not made the other assumptions, we would have also required a knowledge of $(\underline{V}_M - \underline{V}_T)$, R, $\sigma$, and $a_T$. This is true even <u>after</u> the initial assumptions that the dynamics are linear, the autopilot is perfect, and all motion occurs in one plane.

Clearly, the gap between required state information and measured state information creates a significant problem if we are to apply modern control theory to develop advanced guidance laws. The additional requirement that our models be accurate for both linear and nonlinear formulations and in the presence of stochastic processes presents additional challenges. For this reason, an important part of our research program is devoted to the study of advanced optimal estimation techniques for tactical missile application. Figure 8-1 is a simplified functional description of the use of optimal estimation in tactical missiles. The objective is to provide accurate estimates of all states and model parameters required for the advanced guidance law <u>without</u> significantly increasing the sensor requirements (and therefore cost) for <u>future</u> tactical missiles. As we will see, the computational requirements for such algorithms are similar to those for the optimal control algorithms. However, the one important difference is that the estimation algorithms always require repeated solution on-board the missile in real time. This is primarily due to the fact that they are continually processing measurement data to update the estimates for the constantly changing states and model parameters. If it were not for the microprocessor revolution, we would have little hope of being able to use such theory in the tactical missile scenario. The next section will briefly discuss the general theory and cite some of the advantages and disadvantages of its use in tactical missile applications.

## 9. OPTIMAL ESTIMATION THEORY

There are many different optimal estimation techniques currently undergoing research, some of which we are investigating in our own research program (refer back to figure 4-2). There is no way the scope of this paper will allow me to even mention all these techniques, much less discuss them in any detail. References 10 an 11 contain some information on each of these techniques, plus a compilation of many more references. My approach will be similar to that used in the previous sections on optimal control theory: I will briefly outline a particular theory and then describe some of its major advantages and disadvantages for tactical missile application. See reference 9 for a much more complete treatment of both theory and application. Just as before, the theory I have chosen to outline is one of the simplest but yet most eloquent - the Kalman Filter. Most other techniques under study involve some type of theoretical or practical extension of Kalman filtering theory.

In our previous discussion of optimal control theory, we assumed a <u>continuous</u> model of our system. However, because of the large and repetitive calculation requirements for the Kalman filter, we must assume that the updated state estimates will be provided only at discrete points in time. This leads us to formulate a discrete model of our system. (This also requires us to re-evaluate our optimal control solution, since it requires a <u>continuous</u> knowledge of the system state vector.) In addition, the Kalman filter not only requires a linear dynamical system model but also a linear measurement model. A discrete linear system satisfying the above requirements may be represented by

$$\underline{x}_{K+1} = \phi_K \underline{x}_K + \Gamma_K \underline{u}_K + \theta_K \underline{w}_K \tag{9-1}$$

$$\underline{y}_K = H_K \underline{x}_K + \underline{v}_K \tag{9-2}$$

where (9-1) is a first order Markov process model for our system dynamics and (9-2) is a zero order Markov process model for our measurements. The following definitions are in order:

$\underline{x}_K \triangleq$ the system state vector at time $t = t_k$

$\underline{y}_K \triangleq$ the system measurement vector at time $t = t_k$

$\underline{u}_K \triangleq$ the system control vector at time $t = t_k$

$\underline{w}_K \triangleq$ the system process noise vector at time $t = t_k$

$\underline{v}_k \triangleq$ the system measurement noise vector at time $t = t_k$

$\phi_K \triangleq$ a matrix relating the state at $t = t_{k+1}$ to the state at $t = t_k$

$\Gamma_K \triangleq$ a matrix relating the state at $t = t_{k+1}$ to the control at $t = t_k$

$\theta_K \triangleq$ a matrix relating the state at $t = t_{k+1}$ to the process noise at $t = t_k$

$H_K \triangleq$ a matrix relating the measurement vector at time $t = t_k$ to the state vector at $t = t_K$.

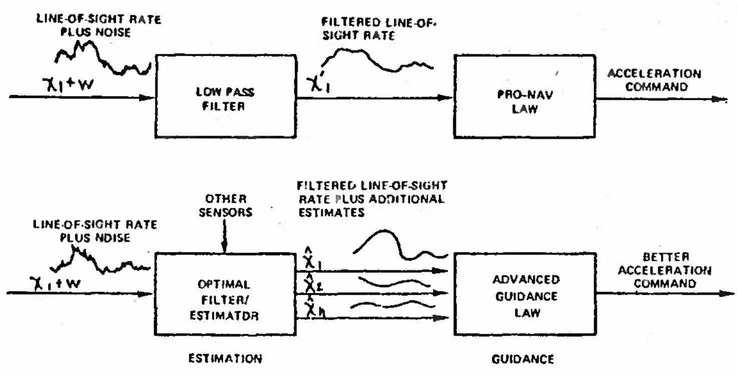


FIGURE 8-1 ILLUSTRATING THE USE OF OPTIMAL ESTIMATION FOR TACTICAL MISSILES (CLASSICAL VS MODERN APPROACH)

To describe the Kalman filter for the system, the following notation is required:

$\bar{\underline{x}}_k = E[\underline{x}_k]$ = Statistical expectation of $\underline{x}_k$.

$\delta_{kj} = 1$ if $i=j$ or $0$ if $i \neq j$ (Kronecker delta).

$\mathrm{Cov}(\underline{w}_K, \underline{x}_j) = E\left[(\underline{w}_K - \bar{\underline{w}}_K)(\underline{x}_j - \bar{\underline{x}}_j)^T\right]$ (Covariance).

$\hat{\underline{x}}_{k/k}$ = Estimate of $\underline{x}_k$ given $\underline{y}_k, \underline{y}_{k-1}, \cdots \underline{y}_o$.

$\hat{\underline{x}}_{k/k-1}$ = Estimate of $\underline{x}_k$ given $\underline{y}_{k-1}, \underline{y}_{k-2}, \cdots, \underline{y}_o$.

$\tilde{\underline{x}}_{k/k} \triangleq \underline{x}_k - \hat{\underline{x}}_{k/k}$

$M_k \triangleq \mathrm{Cov}(\tilde{\underline{x}}_{k/k-1}, \tilde{\underline{x}}_{k/k-1})$

$P_k \triangleq \mathrm{Cov}(\tilde{\underline{x}}_{k/k}, \tilde{\underline{x}}_{k/k})$

If now a linear system model is assumed, and it is further assumed

1) Cov $(\underline{v}_k, \underline{w}_j) = 0$, all k,j;

2) $\underline{w}_k$ is a white noise process with cov $(\underline{w}_k, \underline{w}_j) = K_{w_k}\delta_{jk}$;

3) $\underline{v}_k$ is a white noise process with cov $(\underline{v}_k, \underline{v}_j) = R_k \delta_{jk}$;
then it can be shown that the minimum mean-square error linear estimate of $\underline{x}_{k+1}$ given $\underline{y}_0, \underline{y}_1, \ldots, \underline{y}_k$ is given by:

$$\hat{\underline{x}}_{k+1/k} = \Phi_k\hat{\underline{x}}_{k/k-1} + \Gamma_k\underline{u}_k + \Theta_k\overline{\underline{w}}_k + K_k\,(\underline{y}_k - H_k\hat{\underline{x}}_{k/k-1} - \overline{\underline{v}}_k). \qquad (9\text{-}3)$$

Here the gain $K_k$ is given by

$$K_k = \Phi_k M_k H_k^T\,(H_k M_k H_k^T + R_k)^{-1} \qquad (9\text{-}4)$$

and $M_k$ and $P_k$ are given by

$$M_k = \Phi_{k-1}P_{k-1}\Phi_{k-1}^T + \Theta_{k-1}K_{w_{k-1}}\Theta_{k-1}^T \qquad (9\text{-}5)$$

$$P_k = M_k - M_k H_k^T(H_k M_k H_k^T + R_k)^{-1}H_k M_k \qquad (9\text{-}6)$$

To implement this, the system matrices $\Phi_k$, $\Gamma_k$, $\Theta_k$, $H_k$ and the noise covariances $K_{w_k}$ and $R_k$ and the noise expectations $\overline{\underline{v}}_k$ and $\overline{\underline{w}}_k$ must be supplied, along with an initial state estimate $\overline{\underline{x}}_{0/-1} = E[\underline{x}_0]$ and its covariance $M_0 = \text{Cov}\,[\underline{x}_0, \underline{x}_0]$.

For clarity, the filter can be broken up into a predictor, which takes the state estimate $\hat{\underline{x}}_{k/k}$ and extrapolates it to K+1, yielding $\hat{\underline{x}}_{k+1/k}$, and a corrector, which takes the new data vector $\underline{y}_{k+1}$ and uses it to improve this estimate, yielding $\hat{\underline{x}}_{k+1/k+1}$. These equations are:

Predictor- $$\hat{\underline{x}}_{k+1/k} = \Phi_k\hat{\underline{x}}_{k/k} + \Gamma_k\underline{u}_k + \Theta_k\overline{\underline{w}}_k \qquad (9\text{-}7)$$

Corrector- $$\hat{\underline{x}}_{k+1/k+1} = \hat{\underline{x}}_{k+1/k} + M_{k+1}H_{k+1}^T\,(H_{k+1}M_{k+1}H_{k+1}^T + R_k)^{-1}(\underline{y}_{k+1} - H_{k+1}\hat{\underline{x}}_{k+1/k} - \overline{\underline{v}}_k) \qquad (9\text{-}8)$$

The Kalman filter is the optimal filter for state estimation if:

(1) The criterion of optimality is minimum weighted mean square error; i.e., the minimization of $J = E\left[(\underline{x}-\hat{\underline{x}})^T A(\underline{x}-\hat{\underline{x}})\right]$ where A is a symmetric, positive definite matrix.

(2) The system is linear.

(3) All information required is available $(\Phi_k, \Gamma_k, \Theta_k, H_k, K_{w_k}, R_k, E[\underline{x}_0]$, Cov$[\underline{x}_0, \underline{x}_0], \overline{\underline{w}}_k, \overline{\underline{v}}_k)$.

(4) Either

a) $\underline{x}_0, \underline{w}_j, \underline{v}_j$ (j=0, ...,k) are independent jointly Gaussian random vectors meeting the previous assumptions, or

b) The estimator is constrained to be a linear function of $\underline{x}_0$ and $\underline{y}_j$ (j=0, 1,...,k).

It may be noted that for the missile problem, requirements 2, 3, and 4 are not met. This means that the Kalman filter will probably not be optimal for this problem. However, a suboptimal filter based on a variation of it may yield desirable results.

Despite its theoretical optimality for the mathematical problems posed above, there are difficulties to be overcome in the application to practical problems. These include divergence, degradation of control performance, and large computational requirements.

Divergence can occur due to inaccuracies in the system model (including unaccounted-for nonlinearities or simple errors in the selection of coefficients for the system matrices), inaccuracies (including state dependence) in the statistical models of the system and observation noise processes ($\underline{w}_k$ and $\underline{v}_k$), and simple computational truncation and roundoff errors. Even small errors of these kinds can produce divergences in which the state estimate errors either converge to finite values other than zero or become infinite. The intuitive explanation for this is that the errors cause the covariance matrices $M_k$ and $P_k$ to take on unrealistically small values, thus causing the filter to essentially ignore the incoming data (See reference 9). This problem is commonly overcome by either limiting the memory of the filter or using a variation on the filter to explicitly account for model errors.

Computational inaccuracies may even result in calculated covariances which are not positive semidefinite, with disastrous consequences. The use of "square root" algorithms avoids this difficulty. (See reference 9).

It has been shown that an optimal state-feedback controller (based on LQ theory), with a Kalman filter estimate of the state substituted for a direct measurement of the state, will always have degraded performance, even if the models are perfect. However, it has also been shown that this is the optimal solution to the combined linear control/estimation problem if the optimization criteria is to minimize $J = E\left[PI\right]$, where PI is given by equation 6.2. But, in the nonlinear case, the estimation and control problems are not in general separ ble. This means that a controller which would be optimal if perfect state information were available may no longer be the best controller if only estimates of the state can be used.

Finally, the computational requirements of a Kalman filter can become very large, especially if the number of measurements or number of states is large. For real-time processing, this may force a simplification of the filter algorithm or system model, and will at least require the use of very efficient algorithms and programs. This will of course be true of any filter selected for this problem.

Techniques have been devised to permit consideration of non-white and cross-correlated measurement and system noise. These essentially amount to ways to restructure the system model to permit direct application of the Kalman filter, which remains optimal and conceptionally unmodified.

Of more present concern are problems which force a modification of the filter. These problems include:

1) Nonlinear state and/or measurement equations.

2) State-dependent noise processes.

3) Uncertainty in the system model.

4) Uncertainty in the statistical properties of the noise processes.

The missile problem suffers from all of these difficulties, although it may be possible or advisable (if established by more specific analysis) to gloss-over or ignore some of them without excessive penalty.

## 10. SUMMARY

In this lecture I have tried to describe the functions of major tactical missile subsystems, review classical guidance laws, and summarize some of the features of modern control and estimation theory. In doing so, I have attempted to exhibit some of the major advantages and disadvantages of each technique, both in general and in relationship to short range air-to-air missile applications. Because time and space have not permitted me to either discuss all the techniques or treat in much detail the ones I did discuss, I have given adequate references which will allow the interested reader to pursue this subject matter in whatever detail he chooses. I hope this presentation has conveyed some appreciation for the exciting future that optimal control and estimation theory will have in solving our common defense problems in an economical manner.

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# GUIDANCE SIMULATION TECHNIQUES

by

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## SUMMARY

Early guidance simulations were extensions of flight control simulations. Since the guidance techniques were inertial, it was easily shown that the dynamics of the two loops could be decoupled. Thus, most simulations of the flight control system were performed on analog computers which grew to hybrid digital systems as the non-linear effects became better known. The inertial systems were modeled on a digital computer because of the small parameters which were important in predicting their performance. Inner loops were frequently simulated by analog computers or simply designed by conventional servo analysis techniques.

This began to change with the introduction of terminal guidance, (initially radar for ground-to-air and air-to-air missiles, and then television for air-to-surface weapons). These technologies did not lend themselves to simulation, thus resulting in many flight test failures. The failures provided the justification for more complex hardware in-the-loop simulations and additional technology evaluation and math modeling - the basis for simulations in use today.

This paper examines the economic and political constraints which allow, justify, and require simulations. Factors such as range time availability and cost increases are shown to drive the designer toward providing 100 percent assurance of each test. A six-degree-of-freedom hybrid simulation will be described, and equipment plus operating costs will be identified and contrasted with range costs.

The simulation described is an accurate aerodynamic simulation model used in a six-degree-of-freedom (6-DOF) simulation program for an air defense missile. The program provides a 6-DOF simulation of the missile system, including detailed non-linear models of the airframe and associated aerodynamics, the autopilot, inertial reference unit, control actuation system, and gyro and accelerometer sensors. Implemented on both digital and hybrid simulations, the 6-DOF simulation program serves as the primary tool for flight test planning, postflight data analyses, and preflight predictions of missile performance characteristics.

Extensive wind-tunnel test programs provided input for development of the basic aerodynamic model. The simulation program was used during control test vehicle flights and showed close agreement between preflight predictions and flight results. Observations of minor differences between actual and predicted characteristics served as indicators to further refine the aerodynamic model.

Aerodynamic model evaluation required development of two simulation programs. One simulation utilizes all pertinent telemetry and radar measurements of flight data and solves the appropriate equations to yield the uncertain aerodynamic characteristics. The other simulation employs a complete linear derivative model that utilizes an aerodynamic derivative model rather than a conventional aerodynamic coefficient model. This linear model determines the correct coupling flight transients by matching flight test responses, and these results are used to update the conventional aerodynamic model.

TABLE I

NOMENCLATURE, SYMBOLS, AND ABBREVIATIONS
USED IN FIGURES AND TEXT

| | |
|---|---|
| ACC | Acceleration |
| A/D | Analog-to-digital converter |
| A/P | Autopilot |
| ATT | Attitude |
| [B] | Matrix for transformation from inertial axes to body coordinates |
| CLOBBER | Selective input routine for updating input data |
| CASPRE | EAI diagnosis routine |
| CTV | Control test vehicle |
| CMD | Command |
| DCU | Digital coefficient unit (see Note 1) |
| EAI | Electronic Associates, Inc. |
| g | Acceleration due to gravity |
| HOI | Hytran operations interpreter |
| HSL | Hytran simulation language |

| | |
|---|---|
| I/O | Input-output |
| IRU | Inertial reference unit |
| MDAC | Multiplying digital-to-analog converter |
| MPXR | Multiplexer |
| MTI | Multi-time interrupt |
| MACH | Mach number |
| PRI | Priority interrupt |
| $q$ | Dynamic pressure |
| 6-DOF | 6-degrees-of-freedom |
| μs | Microseconds |
| $1/s$ | Integrator |
| $\lambda_i$ | Angle between projection of missile velocity, velocity vector and referenced control fin |
| $\phi'$ | Aerodynamic wind angle |
| $\alpha$ | Total angle of attack |
| $\delta_i$ | Control fin $i$ deflection angle |
| $\theta_{pc}$ | IRU pitch attitude command |
| $\theta_{yc}$ | IRU yaw attitude command |
| $\phi_{sc}$ | IRU roll attitude command |
| $QG_c$ | IRU yaw rate command |
| $RG_c$ | IRU pitch rate command |
| $X$ | Launcher-frame downrange position |
| $Y$ | Launcher-frame vertical position |
| $Z$ | Launcher-frame crossrange position |
| $\dot{X}$ | Launcher-frame downrange velocity |
| $\dot{Y}$ | Launcher-frame vertical velocity |
| $\dot{Z}$ | Launcher-frame crossrange velocity |
| $U_\omega$ | Component of missile velocity vectory along X-axis with respect to wind |
| $V_\omega$ | Component of missile velocity vectory along Y-axis with respect to wind |
| $W_\omega$ | Component of missile velocity vectory along Z-axis with respect to wind |
| $V_{TW}$ | Total missile velocity relative to wind |
| $V_{TW}$ | Total inertial velocity |
| $U$ | Component of missile velocity vector along X-axis |
| $V$ | Component of missile velocity vector along Y-axis |
| $W$ | Component of missile velocity vector along Z-axis |
| $\dot{U}$ | Acceleration along X-axis |
| $\dot{V}$ | Acceleration along Y-axis |
| $\dot{W}$ | Acceleration along Z-axis |
| $N_A$ | Acceleration component along roll axis |
| $N_{LP}$ | Acceleration component along pitch axis |
| $N_{LY}$ | Acceleration component along yaw axis |
| $NCP$ | Pitch acceleration command |
| $NCY$ | Yaw acceleration command |
| $\theta_{PE}$ | IRU pitch gimbal signal to autopilot |
| $\theta_{YE}$ | IRU yaw gimbal signal to autopilot |
| $\epsilon_R$ | IRU output roll command to autopilot |
| $P$ | Angular roll rate about missile X-axis |
| $Q$ | Angular yaw rate about missile Y-axis |
| $R$ | Angular pitch rate about missile Z-axis |
| $\psi$ | Euler yaw angle |
| $\theta$ | Euler pitch angle |
| $\phi$ | Euler roll angle |
| $\dot{\psi}$ | Euler yaw angle rate of change |
| $\dot{\theta}$ | Euler pitch angle rate of change |
| $\dot{\phi}$ | Euler roll angle rate of change |

Note 1 A DCU is a digital coefficient unit and it replaces the older servo potentiometers.

Note 2 The four languages are Fortran IV, Assembly, Hytran operations interpreter, and Hytran simulation language.

Note 3 ADC's are used to output data to the disk while normal calculations are being performed.

THE WEAPON SYSTEM DEVELOPMENT ENVIRONMENT

Anyone who has followed the course of weapon system research and development is aware that funding is becoming scarce, projects are more expensive, and poor performance is frequently used to terminate projects. In this environment, guidance simulation has prospered. There is no longer any question of whether simulations should be made or not. Prior knowledge of the missile flight-path is needed to define the safe use of test ranges, to plan test instrumentation, and to establish an estimate of the missile system behavior to compare post-flight and pre-flight data.

In today's funding environment, the buying agencies have established careful and methodical approaches to evaluate each phase of weapon development and ensure that a sound basis of technology is available. Thus, programs proceed from advanced development through full-scale development and then production over periods as long as eight or ten years. With the stretch-out of development time, there has also been a tendency to start more programs than following-year funding will allow to be continued. Therefore, projects are competing for funding. This places a heavy emphasis on successful flight demonstration for public relations purposes. Simulation is therefore used to evaluate even more conditions than required from a design viewpoint in order to furnish confidence of successful system operation. A recent complex missile system which used extensive simulations has experienced a success rate of 78 out of 80 firings.

If one considers the economics of simulation versus testing, it is easy to understand the increased use of simulation. The most expensive hybrid simulations cost approximately $1,200.00 per day to operate. Doubling this amount to support the analysis of the data develops a cost of $300,000 for six months of simulation. This much simulation will evaluate every conceivable combination of system parameters. By contrast, actual flight tests cost from $30,000 to $60,000 per week of range time with an additional expense of $100,000 to $2,000,000 for the expendable weapon. The non-recurring costs of the simulation are not a factor since they occur even if little parametric simulation is done.

ANALOG/HYBRID FACILITY

The Orlando Division of Martin Marietta Corporation owns and operates one of the few multiple hybrid computing installations in the world. The facilities include Electronic Associates Inc. (EAI) 231R-V, 8812, and 781 analog computers, EAI 8400 and Adage Ambilog A200 digital computers, EAI 8900 and Martin Marietta 5200 hybrid computers, and other scientific computing elements. Telephone linkages have been developed such that these hybrid computers can be operated by an ordinary telephone line from remote locations. (The computers have been operated from both the Netherlands and Germany.) The user has control over the simulation and receives both analog strip plots and digital output data.

The analog/hybrid computer facility is used by engineering personnel for missile-control and guidance-design studies, design tradeoffs, and flight-test evaluation. The facility, by tying directly into a system test laboratory, allows preflight checkout of missile hardware both before and after final assembly. It is large enough to handle four complete large-scale 6-degree-of-freedom simulations simultaneously, since it contains four hybrid computing systems configured as shown in Figure 1.

Each of the two EAI 8900 hybrid computing systems consists of three 8812 analog computers, one 8400 digital computer, and one 8930 data interface. The 8812 units represent the state-of-the-art in solid state, 100-volt, general purpose analog computers that have a computing bandwidth of 50 kHz, extensive parallel logic, electronic mode control, servo-set potentiometers of highest accuracy, electronic resolvers and multipliers, card-programmed function generators, high-speed data logging printers, and repetitive operation capability. The main features of the 8400 digital computer and 8930 hybrid data interface are shown in Figure 1.

The 5200 system is composed of all-electronic (no moving parts) EAI 231R-V analog computers, one hybrid data interface, and one Adage Ambilog 200 digital computer. The analog computers are equipped much the same as the 8812 units previously described, differing only in bandwidth (10 kHz for the 231R-V), and in a lesser amount of parallel logic. A unique feature of this facility is the size of the data interface, i.e., 238 multiplying digital-to-analog converters (MDAC), and 4 analog-to-digital (A/D) converters with 32-channel multiplexer capability (128 channels of A/D conversion) and 408 sense and control (logic-level) lines each.

The 790 system contains the latest state-of-the-art EAI 781 analog computers that have a 100 kHz bandwidth and digitally set coefficients, enabling problems to be set up in seconds.

Each of the two EAI 8400 digital computing systems (subsystems of the 8900 hybrid computing systems) brings to the programmer a completely integrated and unique combination of capabilities for scientific real time and batch computation. These advanced, medium-scale systems, in conjunction with an advanced operating system, effectively handle real-time applications associated with simulation, hybrid

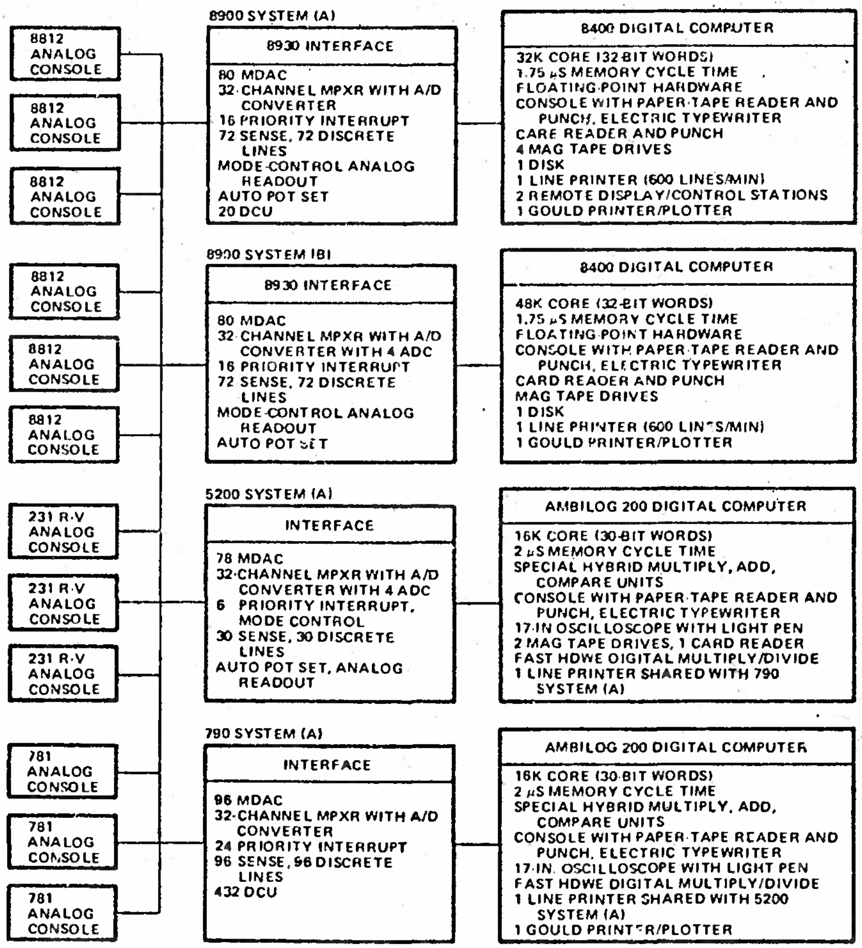



Figure 1. Features of 8900, 5200, and 790 hybrid systems

computation, scientific on-line monitoring and control, and general-purpose scientific computation.

The EAI 8400 digital computer hardware consists of the following:

Central processor

With arithmetic and control logic hardware for a wide range of operations, this unit includes high-speed, floating-point arithmetic with both 32- and 56-bit precision; byte manipulation in 16-, 8-, 4-, 2-, or 1-bit bytes; extensive test and branch operations; special register functions; and other arithmetic and control operations.

Magnetic core memory

The capability of this unit includes 32,000 or 48,000 words of high-speed (1.75 microseconds) core memory (with a memory word consisting of 32 data or instruction bits, 2 special executive bits, and 2 parity bits).

Multilevel interrupt structure

This unit includes extensive interrupt control (32 levels) with masking for dynamic priority assignment.

Automatic data channels

These elements contain two data channels, each with autonomous automatic data channel processors.

Peripheral input/output complement

This complement includes four magnetic tape drives (52/60 kHz at 556/800 bpi) and one disk, one high-speed card reader (800 cpm) and punch (100 cpm), one line printer (600 lpm), one paper-tape reader console with register displays and on-line typewriter.

The EAI 8400 operating system incorporates both digital and hybrid programming features in a single integrated software system that provides efficient operation by ensuring maximum system utilization with a minimum of manual intervention. This system operates under the control of a resident monitor that accommodates a variety of language processors, including the macro-assembler, the Fortran IV Hytran operations interpreter (HOI), and the Hytran simulation language (HSL). The processors (except HOI, which is interpretive) generate object code with a common format that enables them to be combined with programs from the subroutine library by the linking loader. The user can prepare programs in a mixture of languages, calling on library programs, linkage routines, and input/output and control facilities of the monitor. Software provided with the system includes a monitor, processors for four distinct languages, a linking loader, an input/output control system (IOCS), a hybrid run-time library, debugging and utility routines, system diagnostics, and a complete mathematical library.

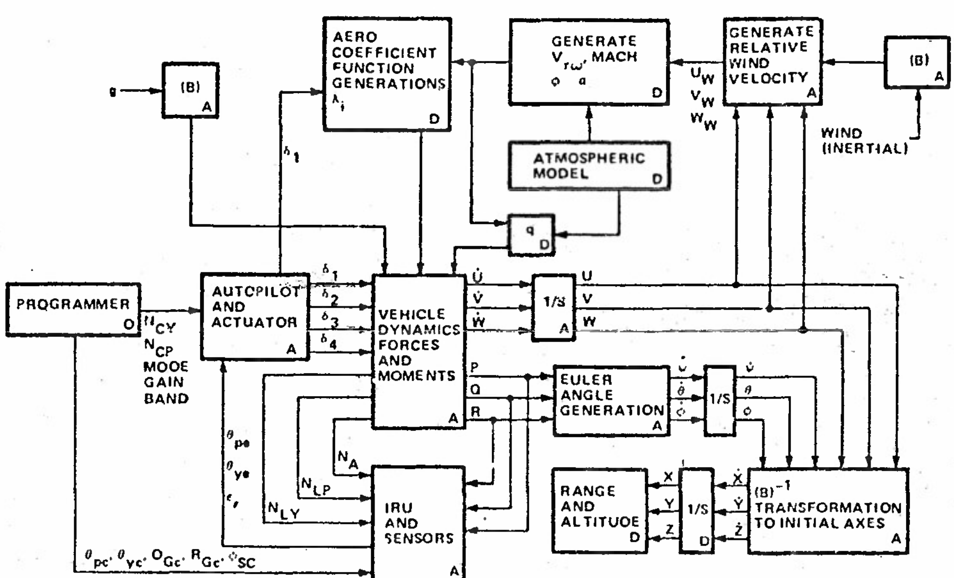
## TACTICAL MISSILE SIMULATION

The hybrid facility has been used to simulate many military missile systems during design and development stages. In general, these simulations normally include the simulation of aerodynamics, autopilot, actuators, and, when applicable, inertial reference units (IRUs). Aerodynamic data, obtained from wind-tunnel tests, are usually programmed on function generators that may be either card-programmed, hybrid, or digital function look-up routines. The autopilot and actuators are generally programmed on the analog portion of the simulation. Tie-in with hardware is usually included as an option, with the actual missile autopilot substituted for the simulated autopilot.

Programs of 2-, 3-, and 5-degrees of freedom are used when single-plane or fixed-velocity studies are sufficient for evaluating missile performance changes due to autopilot compensation changes, actuator changes, or other parameter or module modifications not requiring a complete 6-DOF study.

## AIR DEFENSE 6-DOF SIMULATION MODEL

The design criteria for an air defense hybrid simulation model are determined primarily by the complexity of the non-linear aerodynamic functions, the frequency response requirements of the missile's pitch, yaw, and roll control loops, and the large quantity of required simulated error sources. The simulation represents the six degrees of freedom of the vehicle, with comprehensive modeling of the aerodynamics, rocket motors, on-board programmer, inertial reference unit (IRU), autopilot, body-mounted sensors (three linear accelerometers and three rate gyroscopes), control surface actuators, winds, atmospheres, and error sources. The hybrid computer simulation flow diagram is presented in Figure 2.




NOTES:
A – DENOTES ANALOG COMPUTER IMPLEMENTATION.
D – DENOTES DIGITAL COMPUTER IMPLEMENTATION

Figure 2. Hybrid 6-degree-of-freedom (6-DOF) simulation flow diagram

Representation of the missile's control hardware contains all the major non-linearities required for evalution of stability and performance. The simulation must be maintained in real time in order to permit insertion of actual missile hardware for some phases of the program. The IRU model has pitch and yaw degree of freedom, with simulation of the pitch, yaw, and roll integrating rate gyroscopes, gimbal-angle limits, torque limits of the pitch and yaw platform-drive motors, and electronic-amplifier dynamics and compensation. Equations of motion used are based on a spherical, nonrotating earth with gravity.

The initial-turn flight phase provides a desired change in azimuth and elevation angles after the missile has cleared the launcher. The missile is steered along a precalculated trajectory by use of an inertially stabilized attitude reference system IRU aligned to the desired heading angle in pitch and yaw. This closed-loop body-attitude control system provides the pitch/yaw control loop with a constant input attitude command signal.

For attitude control guidance, the same pitch-yaw autopilot model is used, but it receives error signals resulting from IRU rate commands as well as gain changes. The roll loop maintains the required roll attitude, receiving similar gain and attitude commands, together with other commands that switch in an integral compensation network. In the acceleration guidance mode, the autopilot pitch and yaw control loop accepted acceleration commands by switching in the body-mounted accelerometers.

The rocket-motor model consists of a highly detailed thrust profile, selectable from a group of several stored in memory, with the capability of expanding and contracting either or both amplitude and time duration of any selected profile, such that either thrust or total impulse, or both, could be varied as desired. Thrust ignition also could be time shifted with respect to related events. Uncorrected thrust is corrected for altitude, noting ground-level pressure of the location at which the profile was measured. Propellant weight, missile weight, remaining impulse, missile center of gravity, and inertias are all derived from the generation of thrust. Thrust amplitude and duration are subject to both random and temperature effects. Thrust misalignment as an error source is provided in three dimensions.

The body-coordinate-axis system was selected for this simulation to avoid noise contamination of coordinate second-derivative terms, and to avoid, in the forward and inverse transformation resolvers, mismatch inherent in the inertial axis system mechanization. In addition, a slight equipment edge favored the body-axis system. The existence of a potential scaling problem in the cross-product terms of the U, V, W equation mechanization in the body-axis system could reduce resolution at lower levels.

The guidance programmer model duplicates time and amplitude granularity of the attitude, rate, and acceleration comands, and issues correct mode and gain band switching to the autopilot. The control-actuator model contains servo-valve dynamics and all major non-linearities pertinent to performance, such as current limits, hinge moment load, pressure feedback, and fin deflection limits, and provides histories of coil current, differential pressure, hinge moments, fin rates, integrated fin travel, battery capacity, and other factors. Accelerometer models are provided with bias, cross-axis acceleration error, and angular misalignment. Gyro models include bias, g and g-squared sensitivity errors, and angular misalignments. The autopilot model is equipped with mode and gain-band switching, correct dynamic ranges, a number of monitor points with a fixed ratio between hardware and simulation levels for all monitor points, and such error sources as biases, gains, and limit errors.

Interface with actual missile hardware (autopilot, fin actuators, programmer, and inertial platform) provides coupling at matched signal levels with impedance match and loading isolation, when needed. Switching logic under digital control selects all or part of the necessary hardware components. and monitors 56 points, which are run-to-run selectable. Immediate hardware/all-simulation comparisons are used to check the validity of the models and to detect hardware abnormalities.

AERODYNAMIC MODEL

Required to predict the aerodynamic nature of the missile for any Mach number, aerodynamic roll angle, attack angle, or combination of fin deflections within the operational envelope, the aerodynamic model task can be quite complex. This model is carefully built up through iteration between wind tunnel and flight test data. The final model may be defined by 36,000 pertinent data points packed into 18,000 core locations. Final mechanization of a typical model includes 20 functions of four arguments, 10 functions of three arguments, 5 functions of two arguments, and 40 functions of one argument. Interpolation of all arguments except for the high-speed fin motions is performed digitally, with fin interpolation accomplished by analog circuitry in a pseudo-hybrid fashion, to ensure adequate dynamic response to fin deflection in real time. Control, table access, interpolation, and handling routines for function generation occupies as many as 2500 core locations; mechanization of the aerodynamic model requires 20,000 memory locations, including data tables and generating routines.

The format of the aerodynamic model, especially the fin coefficients, is ideal for simulation. The undeflected fin characteristics are extracted from the total fin coefficients and are combined into the missile total body coefficients, leaving a set of incremental fin coefficients that vanish as the fins approach zero. This results in symmetry with respect to fin deflection and requires the use of data for only one fin polarity, with coefficient polarity switching handled by switching logic. Since incremental coefficients vanish as the fins approach zero, polarity switching is smooth and without discontinuity.

The incremental fin coefficients are based on single-panel data. The effects of interaction of adjacent fin deflections are modeled by means of multipanel correction factors applied to the single-panel fin coefficients. Aerodynamic generation requires approximately 17 of the 25 ms fast-frame time.

## SIMULATION EQUIPMENT AND PROGRAMMING

### Hardware requirements

The 6-DOF hybrid simulation completely fills one EAI 8900 hybrid system, using all 708 amplifiers, 498 auto-set potentiometers, 9 electronic revolvers, and 144 multipliers in three EAI 8812 analog consoles. All 80 multiplying D/A converters and 32 multiplexed A/D converters of the interface are required. Although the digital program was divided into three overlaying sections (each stored on disk for instant recall into memory as needed), one overlay (together with root segment, common, and resident monitor) required 43,840 locations of the 49,000 in the core memory of one EAI 8400 digital computer. During normal running, the model is supported by three magnetic tape drives, one card reader, one line printer, seven 8-channel strip charts, three X-Y plotters, one remote text display and control station, and one disk drive. When flight hardware is brought into the system, a fourth 8812 analog console is required for interfacing. The simulation normally captures 80 data values every 100 ms (missile time) and dumps real time to disk for a non-real-time dump of 160,000 values to magnetic tape at the end of the run. Special runs require data grabs every 20 ms (missile time) for a maximum of 800,000 total values dumped from disk to tape at the end of the run. These tapes dumps are in format, ready for off-line data reduction. The following is a summary listing of the simulation's hardware requirements:

One 8400 EAI digital system, of which the following were required: 49,000 magnetic-core memory locations (32-bit words), 1.75 microseconds cycle time, CPU with high-speed floating-point hardware, byte manipulation, multilevel interrupt structure (32 levels, with masking), console register, 16-bit manual/programmable control, and register displays.

Three 8812 EAI analog consoles, of which the following were required: 708 amplifiers, 144 multipliers, 498 autoset potentiometers, a quantity of 10-card memory diode function generators, 9 electronic resolvers, 65 servo-set limiters, 63 comparators, 135 AND gates, 45 D/A switches, and 51 operational relays.

One EAI 8930 interface, of which the following were required: 32 A/D multiplex converters, 80 D/A multiplying converters, 16 external interrupts, 24 A/D sense lines, 24 D/A control lines, 3 16-bit input data words, 3 16-bit output data words, and 1 automatic data channel processor.

Peripheral gear, of which the following were required: 1 line printer (600 lpm), 3 magnetic tape drives (7 track), 1 disk drive, 1 terminal, 1 remote display and control station, 1 card reader (800 cpm), 1 card punch, and 7 8-channel strip charts.

### Software requirements

The hybrid simulations require a particularly versatile resident monitor that can handle several types of processors, permit the use of on-line debugging routines, and provide timer control, external priority structured interrupts, easy control of input-output, parallel automatic data channel operation, and other services, retrievable from software system in disk storage, as needed. The monitor occupies 5352 core locations.

The 6-DOF simulation has the following multiframe timing requirements: 25 ms for function generation, integration, A/D, D/A, and all computations; 100 ms for data capture to disk, A/D, D/A, and slower computations; and 200 ms for issue of programmer commands to autopilot.

An executive program is used to simplify the handling of the multiframe timing and scheduling, including program initialization, A/D and D/A conversion, mode and program sequencing, real time data capture, and interrupt servicing. The program uses 1536 core locations, sharing memory with the simulation.

To provide rapid but thorough automatic setup and checkout of the analog mechanization, the EAI hybrid operating system (HOI) was used, together with a program written in HOI language, to compute potentiometer settings, set potentiometers and MDAC values, and conduct static checks based on both model equations and on analog wiring. The HOI program and the interpreter HOI do not share memory with the simulation. Capture of large amounts of data in real time is accomplished by means of the real time disk I/O software routines. On-line debugging of the digital program requires the essential EAI CASPRE routing, which shares memory with the program.

The EAI function table processor and function dump are used in preparation of digital function-generation routines and tables (off-line). Extensive off-line data processing of the run data tapes is performed by the T100 data-processing program to provide statistical analyses and automatic data plotting for evaluation.

Data acquisition and output

Run-history data samples of 80 pertinent variables are captured every 0.1 second and dumped into disk storage in real time, using real time I/O software, and are then dumped to magnetic tape at the end of the run in T100 data-reduction format. For special cases, the model can be slowed down by time scaling. Sampling could be done as frequently as every 20 ms (missile time) for finer granularity of output. A disk access-time limitation of approximately 60 ms (maximum) precludes increasing the real time sampling frequency without resorting to multi-buffering. As many as 800,000 values can be captured, stored, and dumped per run. While the dump to tape is occurring, 44 selected parameter values are dumped to the line printer at 1-second or 5-second intervals, with an option of finer granularity (down to 0.1 second) for any selected 10-second interval. Along with the run history, a description of each run and the status of the model are outputted to monitor and to provide a permanent record of status at the time each run is made. Analog tracings of 56 variables are recorded on strip charts throughout each run, while three parameters of special interest can be recorded on X-Y plotters.

Extensive off-line data reduction can be performed by a digital data analysis program that uses the 6-DOF output magnetic tape as input. Being a comprehensive analysis package, this analysis program provides a wide spectrum of analytical capabilities. These capabilities range from simple digital listing and plots to composite overlay plots and tabulations of comprehensive statistical analyses.

Run control

Automated but flexible control of the program is accomplished by means of instructions, parameter change, and option selection input by typewriter, card reader, and console register switches. As many as 280 error sources and parameters can be varied about their nominal values and automatically snapped back to nominal after the run. Nominals can be changed as easily, but are not changed at the end of the run. Run control also permits the user to select any one of 10 atmosphere models, 6 wind profiles, and 7 thrust profiles. The thrust profile can then be shifted and reshaped to model hot or cold motors with or without a change in total impulse. In addition, to increase the effectiveness of the simulation, run control provides some 34 options, some of which are: hardware mode No. 1 (hardware autopilot and control actuation system only); hardware mode No. 2 (hardware programmer, inertial platform, autopilot, and control actuation system); halt at end of run; verify but do not set potentiometers; initialize for new output tape (MTI); terminate run in hold mode; digital start interrupt; output run history to line printer; automatic selection of gain band; initialize new run series; print list of input commands; do not set or verify potentiometers; fine printout option; output run history to tape; change time scale; put coded run number of strip charts; change strip chart speed; change strip chart spacing; use roll-coupling filter; get CASPRE debugging routine; change normal potentiometer setting tolerance; change critical potentiometer setting tolerance; read input commands in CLOBBER mode; suppress multipanel fin correction; and commands from magnetic tape.

Improved potentiometer setting routine

The potentiometer-setting routine included in the EAI system library was greatly improved by the use of an iterative technique that forced the potentiometers to be set to much higher accuracy than that attained with the standard routine. This results in significant improvement in quality and efficiency of both the setup and the run control of this simulation.

Pseudohybrid function generation

Because of the amount of digital work done in the fastest time frame, particularly function generation, the shortest realizable frame time is 25 ms. The resulting update frequency is much too low for interpolating on the relatively fast-moving fins. Consequently, a compromise technique for multivariant generation of fin coefficients was developed whereby interpolation on all arguments except fin deflection is done with analog circuitry. In this way, instantaneous injection of fin effects on missile kinematics are realized, with no update errors so long as fin deflections are within $\pm$ 10 degrees. When fins are deflecting at a high rate and through large angles, update errors equivalent to as much as 20 degrees of deflection could exist for as long as 50 ms. In generation of an erratic function such as hinge moment, such update delay errors become appreciable and the model has to be slowed down to obtain the required sampled frequency.

Setup and checkout of program for operation

Setting up the 6-DOF hybrid program consists of installing analog patchboards on three consoles, inserting function cards into generators, mounting a 6-DOF disk pack, bringing in the H01 program from the disk, setting potentiometers for static test, performing the static test (wire test), setting potentiometers for a dynamic check run, and performing the dynamic check run.

Initially, the setup required approximately 60 minutes, including the time required to manually trim a large number of potentiometers (which failed to set within tolerance) for the dynamic check run. The improvement in the potentiometer-setting routine previously mentioned drastically reduced the number of potentiometers that were set out of tolerance. This, together with the insertion of noise-suppressing capacitors around critical components during static test, resulted in a reduction of checkout time to approximately 30 minutes.

Originally, in order to allow insertion of a complete section of the HOI program, a modified programming technique was developed that, while packing the program into core more efficiently, resulted in awkwardness and inefficiency in the process of making changes to the program. The HOI program was then coded in a straightforward manner that resulted in higher efficiency. To achieve this, however, it was necessary to redivide the HOI program into its functional sections and store each section as a file on disk.

### Hardware-in-the-loop

The 6-DOF hybrid hardware-in-the-loop capability allows the simulation to be interfaced with the autopilot, control actuation system hardware, and simulated-flight missile. This capability allows the guidance programmer, autopilot, and control actuation system to be checked out in normal operation modes.

## HYBRID SIMULATION AND THE FLIGHT-TEST PROGRAMS

The 6-DOF hybrid simulation has been the primary analysis tool for several test programs. The simulation allowed for extensive real time statistical, stability, performance analysis, and hardware-in-the-loop studies that could not be done by digital simulation.

Some 280 system error sources and controllable parameters could be varied about their nominal values and automatically returned to nominal at the end of a run on the 6-DOF hybrid program. Of the 280 parameters, 36 were initial condition values and any one of seven sets of 36 initial condition values was selectable, together with the option that any of the 280 elements, either inside or outside the selected initial condition matrix, could be changed simultaneously. This could be accomplished with the same ease and in the same manner that any controllable parameter could be changed, thus providing the capability for complete analysis of all possible toleranced conditions that would ensure a successful flight.

A key step in validating the simulation was the successful accomplishment of hardware integration tests. Major subsystems, such as the autopilot and control actuation system, were integrated with the simulations and were then tested under extreme conditions to ensure that the simulation models accurately represented the hardware. These tests were accomplished as soon as breadboard or prototypes became available. Matching of hardware and simulation data in a benign environment is a prerequisite to successful matching of hardware and simulation in a flight environment, so the ultimate test was the integration of a complete missile with the 6-DOF hybrid. All flight trajectories were then flown with the missile, which had an active programmer, inertial reference unit, autopilot, and control actuation system. Comparison plots between the simulated system and the hardware system were made for all key system performance parameters to ensure proper operation for the entire flight.

Structural bending data and flexible-body analytical models were verified by stability and frequency tests performed on a ground-vibration survey missile. The purpose of this series of tests was to verify the stability of the missile, determine various frequency responses of the missile and subsystems, and validate analytical models at the flexible-body frequencies. The results were used to synthesize the autopilot filters required to assure stable operation in flight under both nominal and toleranced conditions. During the tests, the missile was suspended in low-frequency slings, and both launch and burnout flight conditions were tested. An active inertial sensor assembly (gyro and accelerometers), inertial reference unit, control actuation system, and autopilot were used in the tests, with all but the autopilot contained in the missile. Other important subassemblies not functionally required for the test were simulated by equivalent mass models to obtain the designed weight distribution throughout the missile. The autopilot breadboard was designed to operate in several different modes so that both open-loop and closed-loop tests could be performed with and without the autopilot filters.

### Flight-test planning

Missile flight-test planning, like many other design activities associated with large systems, is an iterative procedure, since it is outside the capabilities of existing mathematical techniques to set up the system as an analytical optimization problem subject to a set of goals and constraints. Generally, this iterative procedure involves simulating the system, observing the system response to a set of inputs, and then providing a new set of inputs that hopefully will more nearly provide the desired response. The weakest portion of this procedure is the data analysis, i.e., the observation of system reaction to inputs in sufficient depth to provide the basis for perceptive decisions on future inputs.

Digital and hyorid computer techniques, along with the statistical methods, greatly simplify the flight planning task, and result in shorter transition times from system definition to final analysis and to a clearer, more profound understanding of the missile system. The 6-DOF hybrid simulation makes it possible to change both the missile system and its environment in many different ways to provide vast amounts of information on the effects of large numbers of independent error sources on geometrical and missile-performance variables. These data, used as inputs to the digital analysis program, provide a wide spectrum of analysis capabilities. The progress of flight-test planning is illustrated in Figure 3.

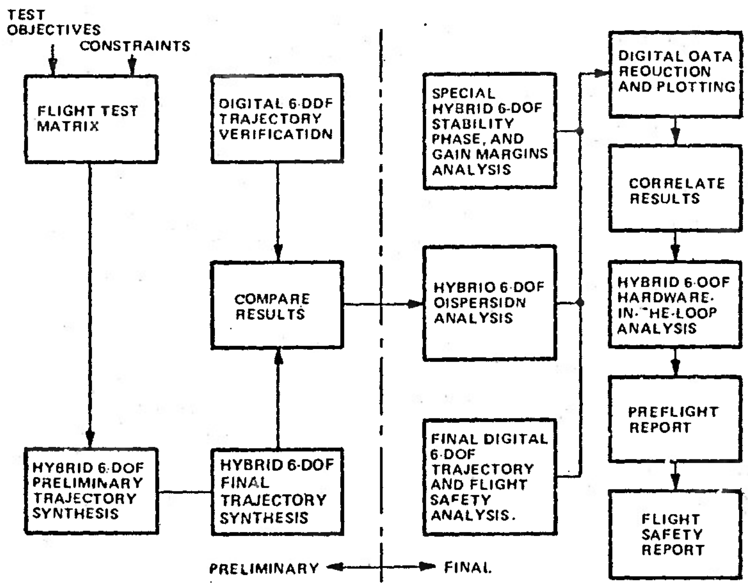



Figure 3. Progress of flight-test planning

The test vehicle trajectories are synthesized on-line on the 6-DOF hybrid simulation. The 6-DOF digital simulation is used to check and verify the final nominal trajectory predictions before dispersion studies are performed on the 6-DOF hybrid simulation. The individual parameter-variation dispersion study is performed to establish statistical predictions of expected 3-sigma dispersions for each missile. 238 independent random dispersion contributors, including propulsion, atmospheric environment, electronic scale errors and biases can be investigated. For each dispersion run, performance information on 63 parameters can be stored on magnetic tape. The data analysis program will process this stored date to obtain the 3-sigma dispersion variations associated with geometrical position of the missile, on-board guidance and control system parameters, Mach-altitude, angle of attack, and load factor. The results of the dispersion study are used to ensure that the predicted dispersion characteristics of the test missile are well within the rated operational capabilities of the missile on a 3-sigma basis for all test commands.

Time histories of missile performance characteristics and their 3-sigma dispersion values can be automatically plotted by a CalComp plottr. For a deeper insight into the dynamics of the missile, error-source ranking tables can be generated. These error-source ranking tables are summarized for each output variable by plotting the time history of each error-source dispersion. This type of plot clearly shows the interplay between the error sources and is also extremely useful during the post-flight reconstruction of flight characteristics when predictions of the inflight errors must often be incorporated into the simulation in search of better predicted flight characteristics. Figure 4 is a typical plot showing the sensitivities of each subsystem and major subsystem error as a function of altitude. The information presented in this figure shows that the major contributor of altitude dispersion is the autopilot electronics subsystem.

Traditionally, dispersions are predicted only on trajectory parameters such as downrange position, crossrange, and altitude. In some cases, angle of attack and total normal load factor may be added. To verify the simulation, dispersion predictions were made for all key performance parameters in order to evaluate flight results on a statistical basis. Statistical dispersions were predicted for geometrical position, angle of attack, total normal load factor, IRU pitch-and-yaw-gimbal angles, pitch-and-yaw-rate gyro output, and pitch-and-yaw-accelerometer output. These predictions were made for the total flight. Key portions of the flight, where test commands were introduced to check missile response, were examined on an expanded

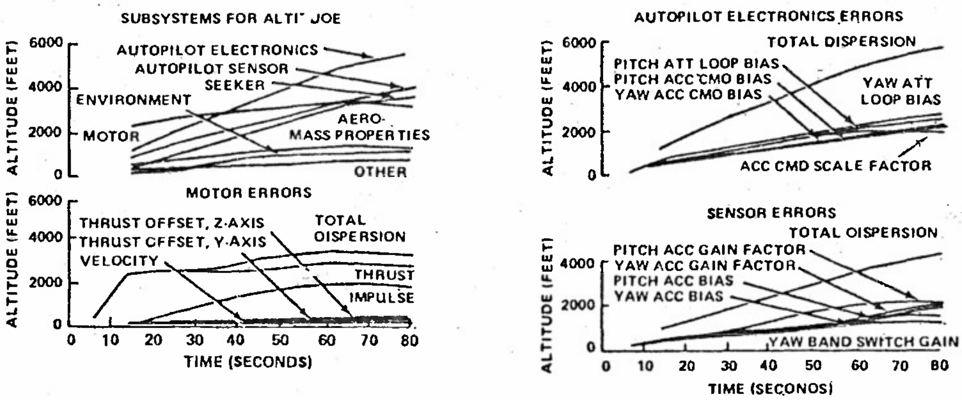



Figure 4. Altitude sensitivities of each subsystem and major subsystem errors at fixed times

time scale in order that motions of the parameter could be evaluated. Figure 5 is a typical preflight prediction expanded-time plot for body pitch-rate gyro output. A typical overlay comparison plot between flight and preflight predictions is presented in Figure 6.

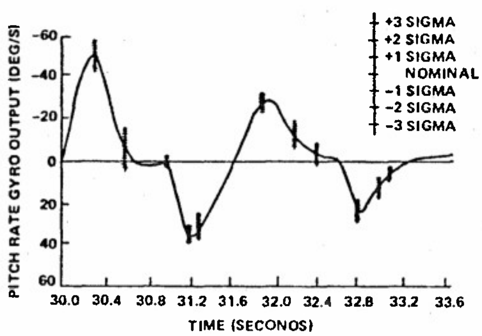



Figure 5. Pitch-rate-gyro output, nominal with 3-sigma dispersions

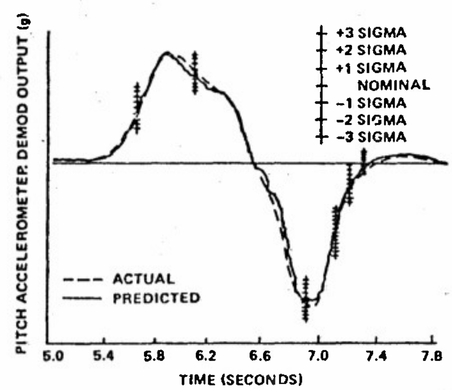



Figure 6. Pitch accelerometer output (transverse acceleration in pitch plane), flight and predicted, with 3-sigma dispersions

In addition to the statistical predictions, overlay comparison plots between flight and preflight predictions were made for 25 performance parameters which include inertial reference unit roll attitude command, IRU roll gyro output, IRU roll integrator output, IRU autopilot command, autopilot roll command, body roll-rate gyro output, pitch-, yaw-, and roll-fin command, control fins 1 through 4 positions, control fins 1 through 4 hinge moments, longitudinal acceleration, downrange, cross-range, and vertical velocity, total velocity, total normal load, and total angular rate.

A prime consideration of the flights was the relative stability (phase and gain margin) available in all control loops during flight. A special stability analysis performed on the 6-DOF hybrid provided a realistic and accurate check of stability characteristics. After the nominal trajectory was obtained, a series of off-nominal trajectories, off-nominal autopilot gains, and aerodynamic tolerance flights was performed. The off-nominal trajectories were configured to approximate the ± 3-sigma trajectories, and the autopilot gains and aerodynamic fin effectiveness were changed to approximate the expected ± 3-sigma change in low frequency and high frequency gain margins.

The advantage of this approach was that the actual angle of attack and aerodynamic roll angle were considered. The use of the hybrid offered many advantages, the most important being that all system non-linearities were modeled, and therefore more accurate results were obtained. However, the results were not a true 3-sigma dispersed value, since the combination of trajectory and autopilot tolerances used to obtain these data was not a random occurrence. The maximum values obtained for the important trajectory, autopilot, navigation system, and control actuation system variables were reviewed as worst-case values representing a larger than 3-sigma dispersion.

The results of this special study were used to ensure that the predicted dispersion characteristics of the test missile were within its rated operational capabilities, and also to determine if the particular missile would complete its flight before dispersions became sufficiently large to result in very low stability.

Flight-test results

Ten flights were completed representing a nine month test program conducted at White Sands Missile Range. The first missile performed exactly as predicted until an error in the range safety system prematurely aborted the flight. The remaining nine missiles were successfully flown according to plan, and all program test objectives were realized. The success of the test flights allowed analysis to be focused on the comparison of predicted versus measured flight values, with refinement of the simulation models as the projected goal.

Accomplishments in this successful flight-test program were many. Missile integrity was demonstrated in 17 maximum acceleration tests and in 24 maximum angle-of-attack tests; 79 pitch/yaw coupling tests were performed to provide aerodynamic and stability characteristics data; and 978 pitch/yaw acceleration commands and 117 roll commands were executed in the flight program to test the missile's response characteristics in all autopilot gain bands.

The test series not only proved the design of the missile's control system, structure, and aerodynamics, but also validated the 6-DOF hybrid simulation model. In all cases, actual flight dynamic responses matched preflight predictions generated 6 to 12 months earlier. Comparison plots, showing that the 6-DOF predicted simulation data were virtually identical to flight data, indicated not only a competent missile system design but also a high degree of sophistication in simulation techniques. The use of the hybrid simulation techniques eliminated the need for a large number of flight-test missiles to verify missile design and performance capabilities, and thus saved time and money.

The flight-test program was designed to maximize the coverage of the specified performance. The objective was to fly trajectories that would test the missile system, satisfy all flight-test objectives, and be compatible with all aerodynamic, structural, control system, and range safety constraints.

From the voluminous quantity of data acquired, four illustrations representative of predicted and flight data were chosen for inclusion here. Typical results of the excellent agreement between predicted and flight data are shown in Figure 7, which presents the time history of yaw-rate gyro, with 3-sigma dispersion bars. Data representative of the yaw accelerometer are presented in Figure 8. Figure 9 illustrates the excellent agreement between predicted and flight data for a command sequence. This figure presents the expanded time-history of the body pitch-rate gyro for a control actuation system duty-cycle sequence. Data representation of the close agreement between predicted and flight data for a control actuation system duty-cycle are presented in Figure 10. Illustrated is the expanded time-history of the No. 3 fin position. The data presented in this figure illustrate the accuracy of the prediction capability of the hybrid simulation, since control-fin position for a given acceleration command sequence is extremely sensitive to variations in missile velocity and altitude.

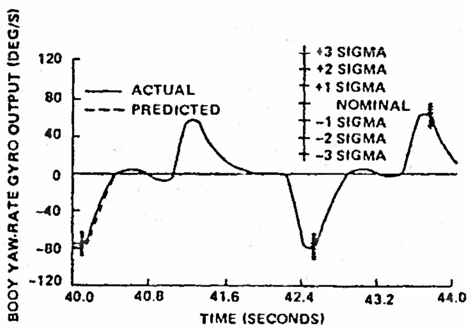



Figure 7. Body yaw-rate gyro output, flight and predicted with 3-sigma dispersions

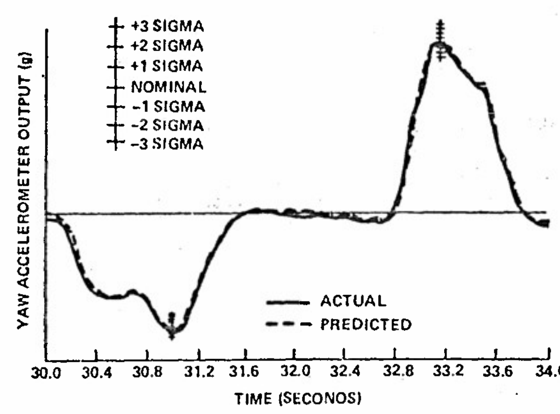



Figure 8. Yaw accelerometer output (transverse acceleration in yaw plane), flight and predicted with 3-sigma dispersions

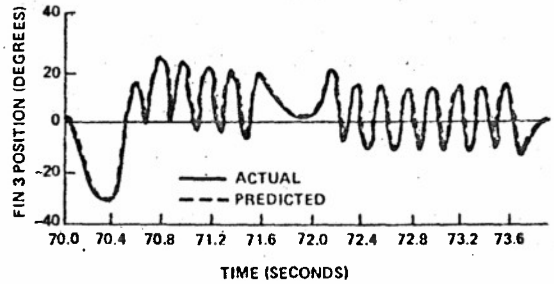



Figure 9. Body-pitch rate-gyro output flight and predicted

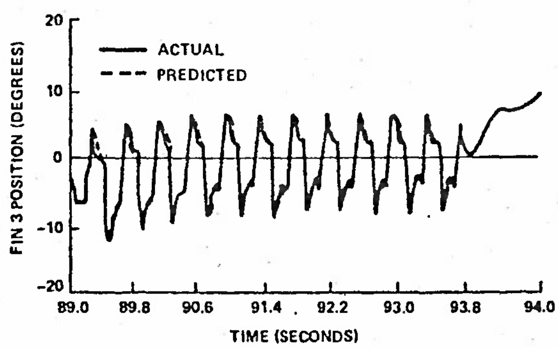



Figure 10. Control actuation system duty-cycle, fin 3 position, flight and predicted

## CONCLUSIONS

A comprehensive hybrid simulation of missile flight dynamics and subsystem operating characteristics can be used for effective prediction of missile flight performance. For a representative missile, this required three analog-computer consoles and one digital-computer console with a 48,000-word core memory and an interface unit. This simulation can also be used effectively for statistical analysis of effects of random variables and for hardware-in-the-loop studies in real time.

Comprehensive simulations must be supported by extensive wind-tunnel and subsystem test programs to provide input data compatible with the degree of sophistication of the analytical simulation.

The flight results to verify the simulation and design were in close agreement with the preflight predictions made 6 to 12 months in advance of the flight.

# MISSILE GUIDANCE TECHNIQUES

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SUMMARY

A unifying approach is proposed for the systematic investigation of missile guidance techniques in guidance law and information processing design. It is based on the consequent distinction between the well-modelled kinematic world and the fuzzy real world together with the separation of the overall guidance problem in a steering problem and a feedback problem. The approach provides for insight in guidance law structures and information requirements as well as in the necessity and the potential benefit of applying modern control theory. Modern filtering and controller design techniques are reviewed for this purpose.

The exemplified investigation of the two-point and three-point guidance principles shows the kinematic guidance law character of extended proportional navigation and of line-of-sight guidance and the possible improvement of system behaviour by the application of modern control techniques.

## 1. INTRODUCTION

The *guidance law* as part of the guidance loop (fig. 1.1) represents an essential component in the design of guided missile systems [1]. The information, which is needed to perform the guidance task of missile-target intercept, determines basically the configuration of necessary *sensors* and *information processing*. The demands on the missile *accelerating capability* as an important system parameter depend strongly on the kind of guidance law. Concerning with the proportional navigation, constant bearing and line-of-sight guidance laws [2] these relations are illustrated in fig. 1.2.

A unifiying approach is proposed to investigate *missile guidance techniques* in the areas of guidance law design and information processing with the advantage of a systematic design and analysis procedure of missile guidance loops. Furthermore insight can be gained about the necessity and potential benefit of applying modern control theory [3].

The approach is based on a consequent distinction between the *"well-modelled kinematic world"* and "the *fuzzy real world*" (fig. 1.3). This provides for a rather clear insight into the necessary guidance law structure and allows to associate guidance law components systematically with distinct guidance properties. Furthermore the overall guidance problem is separated in a "*steering problem*" and a "*feedback problem*" motivated by the insight in the information structure of control problems offered by control theory. This concept admits the application of mathematical tools to the nonlinear steering problem and of control theory to the linear (linearized) feedback problem. As a consequence guidance laws can be structured in *feedforward and feedback control terms* [4, 5].

The different steps of the solution approach are as follows:

(i) A *guidance principle* is formulated as a reasonable idea for missile-target intercept. The principles of two-point guidance (fig. 1.4a) and three-point guidance (fig. 1.4b) are investigated in chapter 2. From the viewpoint of these guidance principles proportional navigation and constant bearing guidance belong to the first guidance class, whereas line-of-sight guidance can be associated with the latter.

(ii) The solution of the overall guidance problem, i.e. how to guide the missile to satisfy the guidance principle, is first treated under the well-defined *kinematic relations of motion*. The concept of *the separation into a steering and a feedback problem* involves the determination of nominal conditions for *exact* satisfaction of the guidance principle and of stabilizing measures for an *asymptotic* satisfaction of the guidance principle in the presence of deviations from the guidance conditions.
The solution of the steering problem provides for initial conditions of the missile motion and the acceleration of the missile along the nominal course. The relation for the necessary acceleration is denoted as steering law. A stability analysis shows that the kinematic deviation behaviour is not asymptotically stable, i.e. errors in the initial conditions and the construction of the steering law will cause unsatisfactory deviations from the nominal course. To stabilize the kinematic deviation behaviour a feedback of the deviations is necessary. Hence the resulting *kinematic guidance law* consists of the *steering law* and the *stabilizing feedback*. It illustrates the *fixed* structure and information needs of the steering law and *possible* information requirements, structures and parameter sets of the feedback law.
Proportional navigation and line-of-sight guidance represents stabilized kinematic guidance laws for two-point guidance and three-point guidance respectively.

(iii) The guided missile while operating in the *real world* is subject to disturbing conditions e.g. limited and noisy information, missile and sensor dynamics, constraints on missile acceleration etc. Since real world effects cannot be comprehended completely and described perfectly, the real world guidance problem is a *fuzzy problem*. It has to be solved by an *iterative step-by-step procedure* to find an "optimal" solution in the sense of keeping the missile motion "sufficiently close" to the nominal course of kinematic guidance with regard to essential real world effects such as dynamic delays and noisy measurements. Hence real world affects the design of the feedback portion of the guidance law and necessitates information processing.
The step-by-step procedure is characterized by the *analysis* of essential effects and their influence on the system performance, by the *synthesis* of suitable information processing and guidance law algorithms and by the *simulation* of the guidance loop to evaluate the system performance.

Design considerations using *frequency domain control techniques* usually involve *parameter determination* in the kinematic guidance laws and in noise suppressing filters. The Wiener filter approach [6, 7] is limited to single-input-single-output systems with time-independent system parameters and noise statistics, assumptions which are in general violated in missile guidance.
To include real world properties more systematically in the solution of the missile guidance problem modern, *time domain control techniques* offer attractive advantages in this case of a multi-input-multi-output system with time-varying system and noise parameters. Optimal (nonlinear or linear) *filtering techniques* [8, 9] provide for noise suppression and additional information processing about guidance loop states from the noisy measurements. Optimal *control law design techniques* [10, 11, 12] lead to extended feedback control structures and parameter determination algorithms to compensate the disturbing influence of missile and sensor dynamics. A brief review of modern control techniques is given in chapter 3. Finally some examples demonstrate the benefit of applying modern control theory in the case of proportional navigation and line-of sight guidance (chapter 4).

## 2. Two-point and Three-point Guidance

### 2.1 Review of Proportional Navigation Derivation

The well-known approach to proportional navigation [2, 5] is reviewed to demonstrate the different viewpoint of the proposed concept for the investigation of missile guidance law and information processing design.

*Choice of the guidance law structure:*
The choice of the guidance law structure for proportional navigation can be motivated by the attempt to avoid the *disadvantages of the pursuit guidance law*, i.e. the kinematically unfavourable missile course with high demands on the missile normal acceleration [2]. Pursuit guidance means that the missile velocity vector $\underline{v}_m(t)$ has always to be directed to the target. Using the notation of fig. 2.1 this can be realized by a missile turning rate $\dot{\Theta}_m(t)$ equal to the line-of-sight angle rate $\dot{\sigma}(t)$. To circumvent the disadvantages of the pursuit course the modified guidance law

$$\dot{\Theta}_m(t) = K(t)\,\dot{\sigma}(t) \quad ; \quad K(t) > 1 \tag{2.1}$$

seems to be reasonable since it provides for a missile lead angle $\varphi_m(t)$ as a basis for a more suitable guidance course. The guidance law structure (e.g. 2.1) is the basic *structure of proportional navigation*.

*Guidance law paramter determination by an analysis of the kinematical behaviour:*
The condition $\dot{\sigma}(t) \equiv 0$ characterizes the collision course in the case of constant speed missile and non-manoeuvring targets [2]. Hence it is interesting to analyse proportional navigation with regard to this condition. For this purpose a relation for the line-of-sight angle acceleration $\ddot{\sigma}(t)$ will be derived from the general kinematic relations (fig. 2.1)

$$\begin{aligned} r \cdot \dot{\sigma} &= v_t \cdot \sin\varphi_t - v_m \sin\varphi_m \; ; \; \sigma(t_o) = \sigma_o \\ \dot{r} &= v_t \cdot \cos\varphi_t - v_m \cos\varphi_m \; ; \; r(t_o) = r_o \end{aligned} \tag{2.2}$$

by differentiation and substitution:

$$\begin{aligned} r\ddot{\sigma} + 2\dot{r}\dot{\sigma} = &- v_m \cdot \cos\varphi_m \dot{\Theta}_m - \dot{v}_m \sin\varphi_m + \\ &+ v_t \cdot \cos\varphi_t \dot{\Theta}_t + \dot{v}_t \sin\varphi_t ; \; \dot{\sigma}(t_o) = \dot{\sigma}_o \end{aligned} \tag{2.3}$$

This is a general differential equation which governs the line-of-sight angle $\sigma(t)$ as a function of the missile turning rate $\dot{\Theta}_m(t)$. Substituting the proportional navigation (eq. 2.1) yields:

$$r\ddot{\sigma} + (2\dot{r} + v_m \cos\varphi_m K)\dot{\sigma} = - \dot{v}_m \sin\varphi_m + v_t \cos\varphi_t \dot{\Theta}_t + \dot{v}_t \sin\varphi_t ; \; \dot{\sigma}(t_o) = \dot{\sigma}_o \tag{2.4}$$

If the *missile velocity is constant* and the *target is not manoeuvring*, the righthand side of eq. 2.4 vanishes. The behaviour of $\dot{\sigma}(t)$ is governed by a homogeneous differential equation. If the gain factor K(t) of proportional navigation (eq. 2.1) is chosen as

$$K = -\Lambda_n \frac{\dot{r}}{v_m \cdot \cos\varphi_m} \quad ; \quad \Lambda_n > 2 \tag{2.5a}$$

and substituted into eq. 2.4

$$r\ddot{\sigma} + \dot{r}(\Lambda_n - 2)\dot{\sigma} = 0 \; ; \quad \dot{\sigma}(t_o) = \dot{\sigma}_o \, , \tag{2.5b}$$

the solution of this differential equation tends to zero for the missile-target-closing, i.e. $\dot{r}(t) < 0$. For a detailed analysis an analytical solution for eq. 2.5b can be obtained as follows [13]: Assuming $\dot{r}(t) \neq 0$ the substitution of

$$\rho = -\ln \frac{r}{r_o} \quad ; \quad 1 \geq \frac{r}{r_o} \geq 0 \quad ; \quad 0 \leq \rho \leq \infty \tag{2.6a}$$

and the application of the differential relations

$$\frac{d}{dt} = \dot{r}\frac{d}{dr} \quad ; \quad \frac{1}{d\rho} = -r \cdot \frac{1}{dr} \tag{2.6b}$$

yields the first-order differential equation with a constant eoefficient

$$\frac{d}{d\rho}\dot{\sigma} + (\Lambda_n - 2)\dot{\sigma} = 0 \tag{2.7a}$$

This equation can easily be solved. Eliminating the substitution the solution of eq. 2.5b is given by

$$\dot{\sigma} = \dot{\sigma}_o \left(\frac{r}{r_o}\right)^{\Lambda_n - 2} \quad . \tag{2.7b}$$

The graphical illustration (fig. 2.2) shows that the angular rate $\dot{\sigma}(t)$ decreases linearly for $\Lambda_n = 3$ and approaches the zero-line asymptotically for $\Lambda_n > 3$. The eollision course condition of $\dot{\sigma}(t) = 0$ is satisfied exaetly at the final point $r = 0$ with a vanishing turning rate $\dot{\theta}_m = 0$. Henee the disadvantages of pursuit guidance are avoided by the proportional navigation guidance law (eq. 2.1) with the gain faetor $K(t)$ given by eq. 2.5.

Furthermore fig. 2.2 shows that the line-of-sight rate $\dot{\sigma}(t)$ of the proportional navigation tends to the collision eourse condition $\dot{\sigma}(t) = 0$ for increasing values of the navigation constant $\Lambda_n$. Therefore the latter is viewed as a speeial ease of proportional navigation in this approach, which is reached in the limiting case of $\Lambda_n \to \infty$.

*Extension for varying missile speed and target manoeuvres:*
The proceeding discussion of proportional navigation has assumed constant missile speed and a non-manoeuvring target. Removing this assumption the differential equation (eq. 2.4) is no longer homogeneous. The solution will not tend to zero except the guidance law of proportional navigation is extended to eompensate the driving functions of eq. 2.4:

$$\dot{\theta}_m = K \cdot \dot{\sigma} + [-\dot{v}_m \cdot \sin\varphi_m + \dot{v}_t \sin\varphi_t + v_t \cos\varphi_t \, \dot{\theta}_t]/(v_m \cdot \cos\varphi_m) \tag{2.8}$$

This type of guidance law is denoted by *extended proportional navigation*. The first term in the braekets represents the well-known *drag compensation*, whereas the other terms provide for *target manoeuvre compentation*.

## 2.2 Two-point Guidance Priniciple and Proportional Navigation Guidance Law

*The two-point guidance principle:*
The principle of two-point guidance is based on relations between the target T and the missile M without any external reference point. *To guarantee missile-target intercept* there is a reasonable idea:

> The missile M will intercept the target T, if it is guided such that an observer in the missile states the target to be approaching on the line to the target's initial position (fig. 1.4a).

This corresponds to the requirement of a constant line-of-sight angle $\sigma(t) = \sigma_o = \text{const}$ with respect to an inertially fixed direction (fig. 2.1). Given the initial line-of-sight angle $\sigma(t_o) = \sigma_o$ the two-point guidance principle can be formulated by the *kinematic guidance conditions*:

$$\bar{\sigma}(t) - \sigma_o = 0 \tag{2.9a}$$

$$\dot{\bar{\sigma}}(t) = 0 \qquad t_o \leq t \leq t_f \tag{2.9b}$$

$$\ddot{\bar{\sigma}}(t) = 0 \tag{2.9c}$$

$$\vdots$$

$$[\dot{\bar{r}}(t) < 0]^{+)} . \tag{2.9d}$$

It is emphasized that the approach of two-point guidance applies the underlying eoncept of constant bearing or collision eourse guidance [2] as initial point-of-view.

+) The closing condition is not analysed. It induces inequalities on the velocity ratio $\kappa_v = v_m / v_t$ [14].

*Kinematic model of motion:*
The kinematic model is based on the *equations of relative kinematics* (eq. 2.2) *and of the missile kinematics*. A normal missile acceleration $a_{mn}(t)$ provides for a proportional turning rate $\dot{\theta}_m(t)$ of the velocity vector $v_m(t)$ and a tangential missile acceleration $a_{mt}(t)$ causes a proportional variation of the missile speed $v_m(t)$. The accelerations are considered to contain control commands as well as disturbing effects.
The target velocity $v_t(t)$ and flight direction $\theta_t(t)$ are the external (driving) functions of the relative kinematics. Using the lead angle relations

$$\varphi_m = \theta_m - \sigma \quad ; \qquad \varphi_t = \theta_t - \sigma \tag{2.10}$$

the kinematic model of motions is given by (fig. 2.1):

$$\dot{\sigma} = r^{-1} \cdot [v_t \cdot \sin\varphi_t - v_m \cdot \sin\varphi_m] \quad ; \ \sigma(t_o) = \sigma_o \tag{2.11a}$$

$$\dot{r} = v_t \cdot \cos\varphi_t - v_m \cdot \cos\varphi_m \quad ; \ r(t_o) = r_o \tag{2.11b}$$

$$\dot{\theta}_m = v_m^{-1} \cdot a_{mn} \quad ; \ \theta_m(t_o) = \theta_{mo} \tag{2.11c}$$

$$\dot{v}_m = a_{mt} \quad ; \ v_m(t_o) = v_{mo} \ . \tag{2.11d}$$

Differentiation and substitution in eq. 2.11 yields a relation for the line-of-sight angle acceleration $\ddot{\sigma}(t)$ needed in the further considerations:

$$\ddot{\sigma} = r^{-1} \cdot [-2 \cdot \dot{r} \cdot \dot{\sigma} - \cos\varphi_m \cdot a_{mn} - \sin\varphi_m \cdot a_{mt} + v_t \cdot \cos\varphi_t \cdot \dot{\theta}_t + \sin\varphi_t \cdot \dot{v}_t]; \ \dot{\sigma}(t_o) = \dot{\sigma}_o \tag{2.12}$$

*Solution of the steering problem:*
The steering problem involves the *determination of the free parameters of the kinematic model* to satisfy the kinematic guidance conditions (eq. 2.9) exactly. The missile normal and tangential acceleration $a_{mn}(t)$ and $a_{mt}(t)$ can be chosen (eq. 2.12) to satisfy the condition (eq. 2.9c) of a vanishing angle acceleration $\ddot{\sigma}(t) = 0$ for $t_o \leq t \leq t_f$. This implies a constant angle rate $\dot{\sigma}(t) = \dot{\sigma}_o$. By a suitable choice of the initial lead angle $\varphi_{mo}$ (or the initial speed $v_{mo}$) in eq. 2.11a a vanishing value of $\dot{\sigma}_o = 0$ can be obtained to satisfy the condition (eq. 2.9b) of a vanishing angle rate $\dot{\sigma}(t) = 0$, $t_o \leq t \leq t_f$. The satisfaction of the angle condition (eq. 2.9a) follows directly. Summarizing the solution of the steering problem consists of relations for

- the initial lead angle $\bar{\varphi}_{mo}$:

$$\bar{\varphi}_{mo} = \arcsin\{\bar{\kappa}_v^{-1} \cdot \sin\varphi_{to}\} \tag{2.13a}$$

with the speed ratio $\bar{\kappa}_v$

$$\bar{\kappa}_v = \bar{v}_m / v_t \quad ; \quad \bar{\kappa}_v \neq 0 \tag{2.13b}$$

- the steering law for the missile accelerations:

$$\bar{a}_{mn} + \tan\bar{\varphi}_m \, \bar{a}_{mt} = v_t \frac{\cos\varphi_t}{\cos\varphi_m} \dot{\theta}_t + \frac{\sin\varphi_t}{\cos\varphi_m} \dot{v}_t \tag{2.14}$$

Since the angle rate condition (eq. 2.9b) is satisfied for $t_o \leq t \leq t_f$, the lead angle relation (eq. 2.13a) describes the lead angle $\bar{\varphi}_m(t)$ for $t_o \leq t \leq t_f$.

*Stability analysis:*
Loosely spoken, stability characterizes the system property of returning to nominal conditions under the influence of initial errors and of moving "close" to the nominal conditions unter the influence of "small" external disturbances. Hence the *stability of the kinematic guidance conditions* (eq. 2.9) has to be investigated to ensure the satisfaction of the guidance conditions in the presence of initial lead angle errors and of steering law construction errors.

The stability analysis applies the "linearized" point of view due to the assumption of only "small" deviations from the kinematic guidance conditions which is finally ensured by a "well-designed" guidance law. To set up a *kinematic deviation model* the actual behaviour of the guidance conditions is linearized about the nominal behaviour of the guidance conditions (eq. 2.9 or equivalently eqs. 2.13, 14) by a first-order Taylor-expansion. The deviation variables $x_1(t)$ and $x_2(t)$ are defined:

$$\begin{aligned} x_1 &= \delta\sigma = \sigma - \bar{\sigma} = \sigma - \sigma_o \\ x_2 &= \delta\dot{\sigma} = \dot{\sigma} - \dot{\bar{\sigma}} = \dot{\sigma} \ . \end{aligned} \tag{2.15a}$$

In view of the following feedback problem the input variables $u_1(t)$ and $u_2(t)$ as additional components of the normal and tangential missile acceleration commands are introduced:

$$u_1 = \delta a_{mn} = a_{mn} - \bar{a}_{mn} \qquad (2.15b)$$
$$u_2 = \delta a_{mt} = a_{mt} - \bar{a}_{mt} .$$

Applying the rule of Taylor-series expansion [10] to the relation of the line-of-sight angle acceleration (eq. 2.12) the kinematic deviation model can be derived:

$$\dot{x}_1 = x_2 \qquad ; \quad x_1(t_o) = 0$$
$$\dot{x}_2 = \bar{r}^{-1} [- 2\cdot\dot{\bar{r}}\cdot x_2 - \cos\bar{\varphi}_m\, u_1 - \sin\bar{\varphi}_m\, u_2] + \qquad (2.16)$$
$$+ \text{ steering law construction errors} \qquad ; \quad x_2(t_o) = x_{2o}$$

representing a system of linear, time-varying differential equations of first order. An *approximate stability analysis* [10] with the assumption of piecewise constant or "frozen" coefficients of the deviation model is performed using the "characteristic polynomial" p(s) of the deviation model

$$p(s) = s^2 + 2 \cdot \dot{\bar{r}} / \bar{r} \cdot s \qquad (2.17)$$

Since the (necessary and sufficient) conditions for the stability of a second-order system are violated, i.e. not all coefficients of the characteristic polynomial are positive, the stability of the deviation behaviour is not guaranteed.

*Solution of the stabilizing feedback problem:*
The input variables $u_1(t)$ and $u_2(t)$, i.e. additional acceleration commands, can be used to modify the dynamical behaviour of the kinematic deviation model by the feedback of the deviation variables $x_1(t)$ and $x_2(t)$, e.g. by the *linear feedback law*

$$u_1 + \tan \bar{\varphi}_m \cdot u_2 = k_1 \cdot x_1 + k_2 \cdot x_2 \qquad (2.18)$$

To determine parameter sets for the feedback coefficients $k_1(t)$ and $k_2(t)$, which stabilize the deviation behaviour, the "characteristic polynomial" $p_c(s)$ of the closed-loop deviation model is analysed

$$p_c(s) = s^2 + \bar{r}^{-1}\cdot(2\dot{\bar{r}} + \cos\bar{\varphi}_m\cdot k_2)\ s + \bar{r}^{-1} \cdot \cos\bar{\varphi}_m\, k_1 \quad . \qquad (2.19)$$

The previously mentioned conditions for stability of a second-order system are satisfied by the choice of the feedback coefficients

$$k_1 > 0 \qquad (2.20a)$$

$$k_2 = - \Lambda_n \cdot \frac{\dot{\bar{r}}}{\cos\bar{\varphi}_m} \quad ; \quad \Lambda_n > 2 \qquad (2.20b)$$

By a suitable choice of $\Lambda_n >> 2$ an arbitrarily fast approach to the kinematic guidance conditions can be achieved which guarantees the missile-target intercept under kinematic world conditions.

*The kinematic guidance law for two-point guidance:*
Summarizing the kinematic guidance law consists of the steering law (eq. 2.14) and the stabilizing feedback law (eqs. 2.18, 20) for the normal and tangential missile accelerations:

$$a_{mn} + \tan \bar{\varphi}_m \cdot a_{mt} = v_t \cdot \frac{\cos\varphi_t}{\cos\bar{\varphi}_m} \cdot \dot{\theta}_t + \frac{\sin\varphi_t}{\cos\bar{\varphi}_m} \cdot \dot{v}_t -$$
$$- k_1 \cdot \delta\sigma - \Lambda_n \cdot \frac{\dot{\bar{r}}}{\cos\bar{\varphi}_m} \cdot \delta\dot{\sigma} \quad . \qquad (2.21)$$

The kinematic guidance law (eq. 2.21) provides for the asymptotic stable satisfaction of the kinematic guidance conditions (eq. 2.9) in the presence of initial lead angle errors and steering law construction errors. For exact implementation target information $[\theta_t(t)\ ,\ \dot{\theta}_t(t),\ v_t(t),\ \dot{v}_t(t)]$, kinematic information $[\sigma(t),\ \dot{\sigma}(t),\ \dot{r}(t)]$ and missile information $[\theta_m(t),\ v_m(t)]$ would be needed.

*Requirements and properties of the kinematic guidance law:*
A discussion of the kinematic guidance law provides for insight into some basic requirements and properties of two-point guidance, especially a different interpretation of the propertional navigation guidance law.

1. Implications of a constant-speed missile and non-manoeuvring targets:
   In the case of a constant-speed missile ($\dot{v}_m = 0$) and non-manoeuvring targets ($\dot{\theta}_t$, $\dot{v}_t = 0$) the steering law (eq. 2.14) implies a vanishing normal missile acceleration

$$\bar{a}_{mn} = 0 \ , \qquad (2.22)$$

i.e. the kinematic guidance law (eq. 2.21) consists of the stabilizing feedback law only.
The nominal two-point guidance course is a straight line between the initial missile position and the collision point and is identical to the well-known collision course [2]. The nominal missile flight direction is determined by eq. 2.13. The closing velocity $\bar{v}_c = -\dot{\bar{r}}$ is constant.

2. Demands on the missile acceleration in case of target manoeuvres:
The demands on the missile acceleration $\bar{a}_m(t)$ along the nominal two-point guidance course

$$\bar{a}_m = \bar{a}_{mn} + \tan\bar{\varphi}_m\,\bar{a}_{mt} \quad , \tag{2.23}$$

which are caused by target manoeuvres, can be derived by substitution of the lead angle relation (eq. 2.13) into the steering law (eq. 2.14). The acceleration ratios $\kappa_{an}$ and $\kappa_{at}$ with respect to normal and tangential target manoeuvres

$$\kappa_{an} = \frac{|\bar{a}_m|}{|\dot{\theta}_t\ v_t|} = \kappa_v \cdot \frac{\cos\varphi_t}{\sqrt{\kappa_v^2 - \sin^2\varphi_t}} \ ; \ 0^\circ \le \varphi_t \le 180^\circ \tag{2.24a}$$

$$\kappa_{at} = \frac{|a_m|}{|\dot{v}_t|} = \kappa_v \cdot \frac{\sin\varphi_t}{\sqrt{\kappa_v^2 - \sin^2\varphi_t}} \tag{2.24b}$$

are illustrated in fig. 2.3 as a function of the target lead angle $\varphi_t(t) = \theta_t(t) - \sigma_o$ and with the speed ratio $\kappa_v(t) = v_m(t) / v_t(t)$ as parameter. The demands on missile acceleration $\bar{a}_m(t)$ exceeds the driving target acceleration only in the case of a tangential manoeuvre near the turning point. Since this kind of manoeuvre is typically negligible the two-point guidance principle is favourable from the view-point on system demands.

3. Implications of a constrained missile lead angle $\bar{\varphi}_m(t)$:
If the missile lead angle $\bar{\varphi}_m(t)$ is limited to a maximum admissible value $\bar{\varphi}_{m\,max}$

$$|\bar{\varphi}_m(t)| \le \bar{\varphi}_{m\,max} \tag{2.25}$$

e.g. due to a constrained field-of-view of a target tracking onboard sensor, a minimum condition on the velocity ratio $\kappa_v$ can be derived from eq. 2.13:

$$\kappa_v \ge \sin\varphi_t \ / \ \sin\bar{\varphi}_{m\,max} \tag{2.26}$$

If the velocity condition is satisfied along the nominal course, the technical constraint (eq. 2.25) is not violated. For technical reasonable values of $\bar{\varphi}_{m\,max} \ge 30^\circ$ the minimum condition (eq. 2.26) is satisfied by a speed ratio $\kappa_v > 2$.

4. Normal and tangential missile acceleration efficiency:
The kinematic guidance law (eq. 2.21) indicates that normal as well as tangential missile acceleration can be applied to satisfy the guidance principle. But their efficiency depends on the missile lead angle $\bar{\varphi}_m(t)$: For lead angles $\bar{\varphi}_m(t) > 45^\circ$ the necessary tangential acceleration $a_{mt}(t)$ is less than the necessary normal acceleration $a_{mn}(t)$.

5. Interpretation of proportional navigation in terms of a two-point guidance principle:
A comparison between the extended proportional navigation guidance law (eq. 2.8) and the kinematic guidance law based on the two-point guidance principle (eq. 2.21) shows two essential differences:

The kinematic guidance law (eq. 2.21) relates the lead angle values $\bar{\varphi}(t)$ to the initial line-of-sight angle $\sigma_o$ due to the relation $\bar{\varphi} = \bar{\theta} - \bar{\sigma} = \bar{\theta} - \sigma_o$.

It contains a feedback term proportional to the line-of-sight deviation $\delta\sigma(t)$ from the initial line-of-sight angle $\sigma_o$ due to the relation $\delta\sigma = \sigma - \bar{\sigma} = \sigma - \sigma_o$.

Hence the two-point guidance condition on the target to approach the missile under the initial line-of-sight angle $\sigma_o$ seems to be the reason for these differences. This condition can be weakened, since it is not necessary that the target approaches under the initial line-of-sight angle $\sigma_o$, but under the given instantaneous line-of-sight angle $\sigma(t)$. In this case of an *instantaneous two-point guidance principle* the kinematic guidance conditions (eq. 2.9) reduce to

$$\begin{aligned}
\bar{\sigma}(t) - \sigma(t) &= 0 \qquad t \le t_f \\
\dot{\bar{\sigma}}(t) &= 0 \\
\ddot{\bar{\sigma}}(t) &= 0 \\
&\ \vdots \\
[\dot{\bar{r}}(t) &< 0] \ .
\end{aligned} \tag{2.27}$$

Repeating the previously shown derivation steps the *kinematic guidance law*

$$a_{mn} + \tan \bar{\varphi}_m \cdot a_{mt} = v_t \cdot \frac{\cos\varphi_t}{\cos \bar{\varphi}_m} \cdot \dot{\theta}_t + \frac{\sin\varphi_t}{\cos\bar{\varphi}_m} \dot{v}_t - \Lambda_n \cdot \frac{\dot{\bar{r}}}{\cos\bar{\varphi}_m} \delta \dot{\sigma} \qquad (2.28)$$

can be derived to satisfy the instantaneous two-point guidance conditions (eq.2.27). Essentially the lead angles $\bar{\varphi}_m(t)$ and $\varphi_t(t)$ are related to the *instantaneous line-of-sight angle* $\sigma(t)$. Furthermore the stability analysis is only to be performed for the deviations of the line-of-sight angle rate, since the angle condition is satisfied by definition. A comparison with the extended proportional navigation guidance law (eq. 2.8) shows the identity of both guidance laws. Especially it turns out that the *proportional navigation term* (eq. 2.1) can be interpreted as the *stabilizing feedback term* of the kinematic guidance law (eq. 2.28).

*Analysis of real world effects:*
Real world effects violate the assumptions of the kinematic guidance law. Hence *their influence on the deviation behaviour from the kinematic guidance conditions* has to be analysed. As a typical example the *influence of missile dynamics on proportional navigation* is discussed. Recalling this is the kinematic guidance law for the instantaneous two-point guidance principle assuming a constant-speed missile and nonmanoeuvring targets. It is referred to [5, 14, 15] for the analysis of target manoeuvres, sensor dynamics and measurement noise.
To approximately describe the real world effects with regard to missile dynamics a second-order, linear model for the missile dynamics is inserted in the guidance loop model (fig. 2.4). Assuming "frozen" parameter of the guidance loop model the characteristic polynomial p(s) can be evaluated:

$$p(s) = s^3 + 2\left(\frac{\dot{\bar{r}}}{\bar{r}} + \xi_m \omega_m\right)s^2 + \omega_m\left(4\frac{\dot{\bar{r}}}{\bar{r}}\xi_m + \omega_m\right)s - \omega_m^2 \frac{\dot{\bar{r}}}{\bar{r}}(\Lambda_n - 2) \qquad (2.29)$$

The necessary stability conditions of positive polynomial coefficients lead to

- the *lower limit* on the navigation constant $\Lambda_n > 2$ known also from the kinematic stability analysis (eq. 2.20b) and
- a *feedback invariant instability* for "small" distances $\bar{r}(t)$, which arises if the relations

$$\bar{r}_m < \frac{|\dot{\bar{r}}|}{\xi_m \omega_m} \quad \text{or} \quad \bar{r}_m < \frac{4\xi_m}{\omega_m}|\dot{\bar{r}}| \qquad (2.30)$$

hold. Assuming for example the values $\omega_m = 5\ s^{-1}$ and $\xi_m = 1.0$ the instability occurs for $r \leq 0{,}8\ |\dot{\bar{r}}|$. In a head-on fight situation with high closing velocities $v_c = |\dot{\bar{r}}|$ e.g. 800 m $s^{-1}$ the guidance design engineer has to give due regard to this effect.

Together with the necessary conditions the following condition

$$2\left(\frac{\dot{\bar{r}}}{\bar{r}} + \xi_m \omega_m\right)\left(4\frac{\dot{\bar{r}}}{\bar{r}}\xi_m + \omega_m\right) + \omega_m \frac{\dot{\bar{r}}}{\bar{r}}(\Lambda_n - 2) > 0 \qquad (2.31)$$

is sufficient for stability which leads to

an *upper bound* on the navigation constant $\Lambda_n$:

$$\Lambda_n < 2\left[1 + \frac{\bar{r}}{|\dot{\bar{r}}|}\,\frac{\left(\xi_m\omega_m - \frac{|\dot{\bar{r}}|}{\bar{r}}\right)\left(\omega_m - 4\xi_m\frac{|\dot{\bar{r}}|}{\bar{r}}\right)}{\omega_m}\right] \qquad (2.32)$$

Substituting the above-mentioned numerical values the admissible region for stabilizing navigation constants $\Lambda_n$ is shown in fig. 2.5.

The stability analysis including the missile dynamics makes evident that the kinematic guidance conditions (eq. 2.27) cannot be satisfied within the final region by the proportional navigation guidance law. Therefore it is necessary to investigate the terminal miss distance behaviour with respect to missile dynamics. The results of [15] are illustrated in fig. 2.6.

*Guidance law synthesis under real world conditions:*
The synthesis of the guidance law together with the necessary information processing using available measurements is first considered from *the "conventional" viewpoint.* Continuing the example of proportional navigation as realization of the instantaneous two-point guidance principle under the above-mentioned assumptions there are typically two design steps:

Guidance law synthesis by parameter determination:
Applying the structure of proportional navigation (eqs. 2.1, 4) as the guidance law to be realized it remains to determine the value of the navigation constant $\Lambda_n^*$ (eq. 2.4) providing for a "sufficient compromise" of the deviation behaviour under real world conditions. The influence of the factor $1/\cos\bar{\varphi}_m$ can be included in the consideration.

Information processing by noise suppressing filters:
The realization of proportional navigation fig. 2.7 requires information about the closing velocity $v_c = |\dot{r}|$ and the line-of-sight angle rate $\dot{\sigma}(t)$, which can be obtained from a missile homing radar sensor by a Doppler-frequency measurement and by the differentiating property of a target tracking unit, obviously to be seen from the transfer-function of the line-of-sight tracking unit

$$F_T(s) = \frac{s}{1+\frac{s}{K_{TT}}} \quad . \tag{2.33}$$

Since the measurements are corrupted by noise e.g. radar glint noise suppressing filters are implemented. Frequency domain techniques e.g. Wiener filtering are applied for a suitable design.

If the prescribed specifications on guided missile behaviour cannot be satisfied by the conventional solution the above-shown approach indicates systematically *two areas for potential improvement* of the guided missile properties:

Inclusion of the steering law into the guidance law design (eq. 2.28):
Due to the omitted steering law in conventional design *target manoeuvres and missile velocity variations* act as external disturbances on the deviation behaviour from the kinematic guidance conditions (eq. 2.27) which cannot be compensated sufficiently by the stabilizing feedback law of proportional navigation.

Extensions of the stabilizing feedback law:
To stabilize the deviation behaviour from the kinematic idance conditions with regard to *delaying elements in the guidance loop* it is n. essary to extend the feedback law to include additional deviation variables. In the case of missile dynamics (fig. 2.5) the extended feedback law

$$\delta a_{mn} = -\Lambda_n \frac{\dot{\bar{r}}}{\cos\varphi_m} \delta\dot{\sigma} - k_1 \delta a - k_2 \delta\dot{a} \tag{2.34}$$

leads to the closed-loop characteristic polynomial $p_c(s)$

$$\begin{aligned} p_c(s) &= s^3 + (2\dot{\bar{r}}/\bar{r} + 2\xi_m \omega_m + \omega_m^2 k_2)\, s^2 + \\ &\quad + [2\dot{\bar{r}}/\bar{r}\,(2\xi_m \omega_m + \omega_m^2 k_2) + \omega_m^2 (1 + k_1)]\, s + \\ &\quad + \omega_m^2\, \dot{\bar{r}}/\bar{r}\, [2(1 + k_1) - \Lambda_n] \end{aligned} \tag{2.35}$$

The necessary *conditions for stability can be satisfied* in this situation by the choice of feedback gains

$$\begin{aligned} k_2 &> -2\,[\dot{\bar{r}}/\bar{r} + \xi_m \omega_m]/\omega_m^2 \\ k_1 &> -2\,\dot{\bar{r}}/\bar{r}\,(2\,\xi_m \omega_m + \omega_m^2 k_2)/\omega_m^2 - 1 \\ \Lambda_n &> 2\,(1 + k_1) \end{aligned} \tag{2.36}$$

The necessary parameter regions for the above-mentioned numerical values ($\xi_m = 1$, $\omega_m = 5$, $|\dot{r}| = 800$) are illustrated in fig. 2.8.
Both measures will in general require the availability of additional information which can be obtained from additional sensors e.g. missile accelerometers or partially can be drawn by processing on the present measurements. Hence a *third area* has to be considered:

Information processing for *noise suppression* and *information reconstruction from noisy measurements.*

*To solve the design problems* related to the three areas of possible improvement of guided missile properties *modern control theory* offers solution techniques

to estimate complete information from noisy measurements by *optimal filtering theory* and

to determine suitable structures and parameter sets of the extended feedback law by *optimal control theory.*

Both areas of modern control theory are roughly reviewed in chapter 3.

## 2.3 Three-point Guidance and the Line-of-sight Guidance Law

Due to the unifying property of the proposed approach to guidance law design this section is identically structured as the previous section. Hence the application to distinct guidance principles e.g. in midcourse guidance is straight forward. For reason of the limited space the presentation of this section is kept briefly.

*The three-point guidance principle:*
The guidance principle of three point guidance is based on relations between the target T and the missile M relative to a reference point O. The *reasonable idea for missile-target intercept* can be formulated:

The missile M will intercept the target T if it approaches the target on the line between the reference point O and the target T.
Using the missile and target line-of-sight angles $\varepsilon_m(t)$ and $\varepsilon_t(t)$ relative to the reference point O (fig. 2.9) the three-point guidance principle can be expressed by the *kinematic guidance conditions*:

$$\bar{\varepsilon}_m(t) - \varepsilon_t(t) = 0 \tag{2.37a}$$

$$\dot{\bar{\varepsilon}}_m(t) - \dot{\varepsilon}_t(t) = 0 \tag{2.37b}$$

$$\ddot{\bar{\varepsilon}}_m(t) - \ddot{\varepsilon}_t(t) = 0 \tag{2.37c}$$

$$\vdots$$

$$(\dot{\bar{r}}_t - \dot{r}_m < 0) . \tag{2.37d}$$

*Kinematic model of absolute motion:*
The kinematic model of absolute motion consists of the missile motion model (section 2.2) and the relations of the absolute kinematics to be derived from fig. 2.9:

$$\dot{\varepsilon}_m = v_m \cdot \sin\varphi_m / r_m \quad ; \quad \varepsilon_m(t_o) = \varepsilon_{mo} \tag{2.38a}$$

$$\dot{r}_m = v_m \quad \cos\varphi_m \quad ; \quad r_m(t_o) = r_{mo} \tag{2.38b}$$

$$\dot{\theta}_m = 1/\, v_m \quad a_{mn} \quad ; \quad \theta_m(t_o) = \theta_{mo} \tag{2.38c}$$

$$\dot{v}_m = a_{mt} \quad ; \quad v_m(t_o) = v_{mo} \tag{2.38d}$$

From these equations the relation for the line-of-sight angle acceleration $\ddot{\varepsilon}_m(t)$ can be derived:

$$\ddot{\varepsilon}_m = r_m^{-1} \cdot [- 2v_m \cos\varphi_m \cdot \dot{\varepsilon}_m + \cos\varphi_m\, a_{mn} + \sin\varphi_m\, a_{mt}] \; ; \; \dot{\varepsilon}_m(t_o) = \dot{\varepsilon}_{mo} \tag{2.39}$$

*Solution of the steering problem:*
Applying an analogous derivation as in chapter 2.2 the kinematic guidance conditions of three-point guidance (eq. 2.37) are exactly satisfied by the following conditions:

The initial missile line-of-sight angle $\varepsilon_{mo}$ and initial missile lead angle $\varphi_{mo} = \theta_{mo} - \varepsilon_{mo}$

$$\bar{\varepsilon}_{mo} = \varepsilon_t(t_o) \tag{2.40a}$$

$$\bar{\varphi}_{mo} = \arcsin\,(\dot{\varepsilon}_t(t_o) \cdot \bar{r}_{mo} / \bar{v}_{mo}) . \tag{2.40b}$$

The missile acceleration by the steering law

$$\bar{a}_{mn} + \tan\bar{\varphi}_m\, \bar{a}_{mt} = \bar{r}_m/\cos\bar{\varphi}_m \cdot \ddot{\varepsilon}_t + 2\bar{v}_m\, \dot{\varepsilon}_t \tag{2.41}$$

Since the angle rate condition (eq. 2.37b) is satisfied for $t_o \leq t \leq t_f$, the lead angle behaviour $\varphi_m(t)$ is described by eq. 2.40b also for $t_o \leq t \leq t_f$.

*Stability analysis:*
The *kinematic deviation model* to describe the actual deviation behaviour from the kinematic guidance conditions (eq. 2.37) due to errors in the nomial conditions (eqs. 2.40, 41) contains the deviation variables

$$\begin{aligned} x_1 &= \delta\, \varepsilon_m = \varepsilon_m - \bar{\varepsilon}_m \\ x_2 &= \delta\, \dot{\varepsilon}_m = \dot{\varepsilon}_m - \dot{\bar{\varepsilon}}_m \\ u_1 &= \delta\, a_{mn} = a_{mn} - \bar{a}_{mn} \\ u_2 &= \delta\, a_{mt} = a_{mt} - \bar{a}_{mt} . \end{aligned} \tag{2.42}$$

Linearization of eq. 2.39 by a first-order Taylor series yields the kinematic deviation model for three-point guidance

$$\begin{aligned} \dot{x}_1 &= x_2 \qquad\qquad ; \; x_1(t_o) = x_{1o} \\ \dot{x}_2 &= \bar{r}_m^{-1} \cdot [- 2\bar{v}_m \cdot \cos\bar{\varphi}_m \cdot x_2 + \cos\bar{\varphi}_m \cdot u_1 + \sin\bar{\varphi}_m \cdot u_2]; \; x_2(t_o) = x_{2o} \end{aligned} \tag{2.43}$$

Under the assumption of "frozen" parameters the examination of the characteristic polynomial p(s)

$$p(s) = s^2 + 2\, \bar{v}_m \cdot \cos\bar{\varphi}_m / \bar{r}_m \cdot s \tag{2.44}$$

indicates a deviation behaviour, which is not asymptotically stable since the coefficient of $s^0$ is zero

*Solution of the stabilizing feedback problem:*
To modify the dynamical behaviour of the deviation model the linear feedback law

$$u_1 + \tan\bar{\varphi}_m\, u_2 = -k_1\, x_1 - k_2\, x_2 \tag{2.45}$$

can be applied. Examining the characteristic polynomial $p_c(s)$ of the closed loop deviation model

$$p_c(s) = s^2 + \bar{r}_m^{-1} \cdot (2\, \bar{v}_m \cos\bar{\varphi}_m + \cos\bar{\varphi}_m \cdot k_2) \cdot s + \bar{r}_m^{-1} \cdot \cos\bar{\varphi}_m \cdot k_1 \tag{2.46}$$

stabilizing parameter sets of the feedback law (eq. 2.45) are given by

$$k_1 = \beta \cdot \bar{r}_m / \cos\bar{\varphi}_m \quad ; \beta > 0 \tag{2.47a}$$

$$k_2 \geq -2\, \bar{v}_m . \tag{2.47b}$$

*The kinematic guidance law for three-point guidance:*
Summarizing the kinematic guidance law for three-point guidance includes the steering law (eq. 2.41) and the stabilizing feedback law (eqs. 2.45, 47):

$$a_{mn} + \tan\bar{\varphi}_m\, a_{mt} = \bar{r}_m/\cos\bar{\varphi}_m \cdot \ddot{\varepsilon}_t + 2\bar{v}_m\, \dot{\varepsilon}_t - \beta \cdot \bar{r}_m/\cos\bar{\varphi}_m \cdot \delta\, \varepsilon_m - k_2 \delta\dot{\varepsilon}_m \tag{2.48}$$

It provides for asymptotic satisfaction of the kinematic guidance conditions (eqs. 2.37, 38). In the case of $k_2 = 0$ the kinematic guidance (eq. 2.48) corresponds to the well-known line-of-sight guidance law [5].

*Properties and requirements of the kinematic guidance law:*

1. Implications of a constant-speed missile and non-manoeuvring targets:
Examining the steering law of three-point guidance it is to be seen that the assumptions $\dot{v}_m$, $\dot{\theta}_t$, $\dot{v}_m = 0$ do not cause the steering law acceleration (eq. 2.41) to vanish - a basic difference to two-point guidance.

2. Implications of a constrained missile accelerating capability:
Investigating the influence of the above mentioned property of three-point guidance for the case of a constant-speed missile with a maximum admissible normal acceleration $\bar{a}_{m\,max}$

$$|a_{mn}(t)| \leq \bar{a}_{m\,max} \quad , \tag{2.49}$$

shows a further disadvantage. There are regions around the reference point, where the kinematic guidance conditions of three-point guidance (eq. 2.37) cannot be satisfied without violating the system constraint (eq. 2.49). The bounds of these regions, where the limitation is just violated at interception, are derived in [16]. They are illustrated in fig. 2.10 in a normalized (x*, y*)-plane with the velocity ratio $\varkappa_v = v_m / v_t$ as parameter.

3. Implications by constrained missile lead angle:
Assuming the admissible lead angle $\varphi_m(t)$ to be limited to a maximum value $\varphi_{m\,max}$

$$|\varphi_m(t)| \leq \varphi_{m\,max} \tag{2.50}$$

for technical reasons as limited beamwidth of onboard beacons or retroreflectors, a minimum condition on the velocity ratio $\varkappa_v$ can be derived.
Substitution the relation for the target line-of-sight angle rate $\dot{\varepsilon}_t(t)$ according to eq. 2.38a into the nominal lead angle condition (eq. 2.40b) it follows that the maximum value of $\bar{\varphi}_m(t)$ for a fixed target lead angle $\varphi_t(t)$ is obtained at the collision point $r_m(t_f) = r_t(t_f)$. After reordering eq. 2.40b under this condition, an inequality for the velocity ratio $\varkappa_v$ is derived

$$\varkappa_v \geq \sin\varphi_t \Big|_{r_m = r_t} / \sin\bar{\varphi}_{m\,max} \tag{2.51}$$

which is equivalent to the condition (eq. 2.26) of lead angle limitation in two-point guidance.

For reason of similarity to the results of section 2.2 about the anaysis of real world effects [17] and to the considerations about the synthesis under real world conditions these steps of approach are omitted.

## 3. Review of Modern Control Theory With Regard to Missile Guidance Demands +)

### 3.1 Information Processing by Optimal Filtering Techniques

Information processing represents a *substantial link between the information needs of the guidance law and the possible information offer of the feasible sensor equipment* of a guided missile system. Especially the considerations about extended guidance law design are influenced by two features: On one hand information can be obtained from noisy measurements only; on the other hand direct information sensing cannot be performed for each signal by physical and/or economical reasons.

Filtering theory provides for *tools of information processing on noisy measurements*. It is based on the reasonable idea to separate the measurement signals in *time-correlated signals* and *time-uncorrelated disturbances*. The later do not possess any information about the past which may be useful in the future: They are purely random. Therefore filtering techniques aim at *estimation of the complete time-correlated information*.

The correlated portion of measurement signals includes the information signals as well as time-correlated disturbances, i.e. coloured noise. To describe their dynamical behaviour mathematically differential equations can be used (concept of shaping filters [18]). From the physical point of view uncorrelated disturbances represent noise with negligible time-correlation relative to the correlated signals. Mathematically they can be modelled by white noise [8]. Restricting the review to the linear, Gaussian case, filtering theory is based on the *mathematical (real world) model*:

- measurement model:

$$z = H x + v \tag{3.1a}$$

$z(t)$: m-dimensional measurement vector;

$v(t)$: m-dimensional measurement noise vector with white, Gaussian noise $v(t) \sim N(0,R(t))$;

$x(t)$: n-dimensional state vector for correlated signal modelling

- state space model:

$$\dot{x} = F x + G w + D u \;; \quad x(t_o) \sim N(x_o, P_o) \tag{3.1b}$$

$w(t)$: s-dimensional input noise vector with white, Gaussian noise $w(t) \sim N(0, Q(t))$;

$u(t)$: r-dimensional deterministic input vector

The matrices $F(t)$, $G(t)$, $D(t)$ and $H(t)$ are of appropriate dimensions. Since the state vector $x(t)$ contains all useful information the design aim of filtering theory consists of developing *algorithms to produce a state estimate* $\hat{x}(t)$ using the available measurements $z(\tau)$, $t_o \leq \tau \leq t$. In the case of *high quality demands on the estimation performance* it is advantageous to formulate the estimation problem as an *optimal filtering problem* with regard to the estimation-error variances as performance measure:

Given measurements $z(\tau)$, $t_o \leq \tau \leq t$ based on a state vector model (eq. 3.1). Find a state estimate $\hat{x}(t)$ of the actual state $x(t)$ such that a quadratic performance criterion J on the error-covariance matrix $P(t) = E\{\tilde{x}(t)\,\tilde{x}^T(t)\}$ with the estimation error vector $\tilde{x}(t) = x(t) - \hat{x}(t)$ is minimized:

$$J = \text{trace } P(t) \rightarrow \min . \tag{3.2}$$

The solution of the optimal filtering problem is given by the well-known *Kalman-Bucy filter* [8], which consists of

- a linear vector differential equation for the state estimate $\hat{x}(t)$:

$$\dot{\hat{x}} = F\hat{x} + K_f(z - H\hat{x}) + D u \;; \qquad \hat{x}(t_o) = x_o \tag{3.3a}$$

- a nonlinear matrix (Riccati) differential equation for the error-covariance matrix $P(t)$ to be integrated forward in time:

$$\dot{P} = F P + P F^T - P H^T R^{-1} H P + G Q G^T \;; \qquad P(t_o) = P_o \tag{3.3b}$$

- and a computational rule for the filter feedback matrix $K_f(t)$:

$$K_f = P H^T R^{-1} \quad . \tag{3.3c}$$

The block diagramm (fig. 3.1) shows the solution structure of information processing by optimal filtering.

+) Though realization aspects of digital (onboard-) computers would require a discrete-time presentation of the review of control and filtering techniques [8, 10], the continuous-time discription is used for reason of basic understanding.

The solution of the filtering problem by the time domain approach of Kalman-Bucy filtering offers *essential advantages* as against the frequency domain approach of Wiener filtering [6]:

- The cases of multi-noise-inputs and multi-sensors-configuration can be treated within this framework.
- The real world model (eq. 3.1) is formulated to include time-varying system coefficients and statistic parameters.
- There are numerically efficient algorithms to solve the matrix Riccati equation by means of a digital computer [10, 19].
- If real world and real world model coincide, the estimation accuracy of information processing can directly be obtained from the diagonal elements of the error-covariance matrix P(t). Else it has to be determined by sensitivity analysis [8] or simulation [21].

### 3.2 Controller Design by Optimal Control Techniques

The discussion about *stabilizing feedback laws in guidance law design* (chapter 2) has emphasized the need for controller design techniques which provide for *extended structures* and for *parameter determination algorithms* to satisfy stability and system performance requirements especially with regard to real world effects.

To review *optimal control techniques based on the time-domain approach* (state-space approach [10]) the distinction between optimal control techniques with free design parameters and specification-oriented optimal control techniques is advantageous. This classification of control techniques gives insight in the meaning of "optimality" and in the necessary design procedure of distinct design techniques.

(i) *Optimal Control Techniques with Free Design Parameter Determination:*

To satisfy prescribed system specifications in a (more or less systematic) step-by-step procedure a controller design technique needs free parameters to be determined iteratively until satisfying system behaviour is achieved. For this purpose an optimality criterion can be formulated which includes free design parameters. The minimization of the optimality criterion provides for a unique solution which is continuous in the free design parameters: small variations of the design parameter will cause small variations of system properties under observation. Hence optimaility is of secondary importance for this class of control techniques. The Gaussian, quadratic optimal design procedure as typical example is reviewed in the following [18]:

- The design procedure is based on a real world model according to eq. 3.1. An optimality criterion J is formulated as follows:

$$J = E\{x^T(t_f)\ S_f\ x(t_f) + \int_{t_o}^{t_f} (x^T\ L\ x + u^T\ M\ u)\ dt\} \quad ; \qquad (3.4)$$

  It assumes a fixed control intervall $t_o \le t \le t_f$. The symmetric weighting matrices $S_f$, $L(t) \geq 0$ and $M(t) > 0$ represents the free parameters of the design procedure. By variation of the coefficients of the weighting matrices the final and transition behaviour of the state vector x(t) as well as the control input behaviour u(t) can be influenced.

- The derive a control design technique a functional optimization problem is formulated:

  Given the batch of measurements $z(\tau)$, $t_o \le \tau \le t$, of a dynamical system governed by (eq. 3.1). Find the control input vector $\bar{u} = u(z(\tau), t)$ such that the optimality criterion (eq. 3.4) is minimized.

- The well-known solution of this optimization problem separates in two parts according to the certainty - equivalence - principle [18]:

  - The linear control law with state-vector feedback:

$$u = K_c\ \hat{x}\ ; \qquad \hat{x}(t) \mathrel{\hat{=}} \text{optimal state estimate}\ . \qquad (3.5a)$$

  The optimal feedback matrix $K_c$

$$K_c = M^{-1}\ D^T\ S \qquad (3.5b)$$

  can be calculated by integration of a nonlinear matrix Riccati differential equation for the (n,n) matrix S(t) backward in time:

$$\dot{S} = -\ S\ F - F^T\ S + S\ D\ M^{-1}\ D^T\ S - L\ , \qquad S(t_f) = S_f\ . \qquad (3.5c)$$

  - The optimal state estimator or Kalman-Bucy filter (eq. 3.3) to provide for the optimal state estimate $\hat{x}(t)$ using the instantaneous measurements z(t).

  The structure of the resulting closed-loop system is illustrated in fig. 3.2.

- Some *essential properties of Gaussian quadratic control* are summarized:

A1: The functional optimization problem leads to a *desirable feedback control of the instantaneous measurements* $z(t)$.

A2: The *stability of the closed-loop system* can be proven examining system properties as observability and controllability [10].

A3: The overall design procedure separates in *two independent design steps* corresponding to the considerations about feedback laws and information processing in chapter 2.

A4: If real world and real world model coincide the *average behaviour of the closed loop system* can be examined by the state covariance matrix $X(t) = E\{x(t)\ x^T(t)\}$ and the control covariance matrix $U(t) = E\{u(t)\ u^T(t)\}$. For this purpose the estimation error covariance matrix $P(t)$ of eq. 3.3b and the estimation state covariance matrix $\hat{X}(t) = E\{\hat{x}(t)\ \hat{x}^T(t)\}$ as solution of

$$\dot{\hat{X}} = (F - D\ K_c)\ \hat{X} + \hat{X}(F - D\ K_c)^T + K_f\ R\ K_f^T \quad ; \qquad \hat{X}(t_o) = 0 \tag{3.6}$$

are necessary to evaluate the relations for $X(t)$ and $U(t)$:

$$X = \hat{X} + P \ ; \qquad U = K_c\ \hat{X}\ K_c^T \ . \tag{3.7}$$

If the modelling assumption is not satisfied the system performance is examined by sensitivity analysis and/or simulation.

D1: In most cases the design of the control law (eq. 3.5) requires *extensive computational work* to find a proper set of free design parameters $S_f$, $L(t)$ and $M(t)$.

D2: The solution of the optimal control problem strongly depends on the value of *the final time* $t_f$ which is not exactly known in missile guidance. This may lead to the necessity of on-line computation of the optimal controller.

D3: *On-line computation* of the optimal parameter sets $K_c(t)$ and $K_f(t)$ requires the solution of two Riccati equations (eq. 3.5c, eq. 3.6c), each of which represents $n\cdot(n+1)/2$ nonlinear differential equations.

D4: The *realization of the controller* (eq. 3.3, eq. 3.5) requires the implementation of a full-order state estimator and a full-order feedback law. In case of a sophisticated real world model this may cause problems by *realization effort and reliability*. Hence the controller design is usually based on *lower-order approximate real world models*.

(ii) Specification-oriented Optimal Control Techniques:

To treat the guidance design problem *more systematically with respect to the guidance specifications of most accurate system performance and of low realization effort* it is advantageous

- to use a physically meaningful performance measure in terms of the variances of the state variables and/or
- to put constraints on the structure of admissible solutions for the guidance problem.

Two examples for these types of optimal control techniques are given below.

The *proportional feedback, mean square optimal control problem* is formulated as follows: The real world model is given by eq. 3.1. Find the parameters of the proportional feedback law as most simple realization

$$u = K_p\ z \ , \tag{3.8}$$

such that the instantaneous performance measure J on the variances of the state variables

$$J = E\{x^T(t)\ \Theta(t)\ x(t)\} \quad ; \quad \Theta(t) > 0 \tag{3.9}$$

is minimized. The symmetric weighting matrix $\Theta(t)$ serves to express the distinct accuracy demands on the states and in time.

The optimal solution for the feedback matrix $K_p^*(t)$ can be computed from

$$K_p^* = -\ [D^T\ \Theta\ D]^{-1} \cdot D^T \cdot \Theta \cdot X \cdot H^T \cdot R^{-1} \tag{3.10a}$$

where the state covariance matrix $X(t) = E\{x(t)x^T(t)\}$ is solution of the matrix differential equation

$$\dot{X} = F_c \cdot X + X \cdot F_c^T + D \cdot K_p^* \cdot R \cdot K_p^{*T} \cdot D^T + G \cdot Q \cdot G^T \ ; \qquad X(t_o) = P_o \tag{3.10b}$$

$$F_c = F + D\ K_p^*\ H \ .$$

This control design technique suffers of the *general disadvantage of proportional output feedback:* stability of the closed-loop system cannot be guaranteed in general [10].

The *controller design technique of variance optimal disturbance compensation* [20] is motivated by the distinction between time-correlated disturbances, acting on the system dynamics and measurements, and the physical states, to be controlled most accurately. In this case a performance measure is meaningful which contains the variances of the physical states as well as of the disturbance estimation errors. The minimization of such performance measure provides for a most accurate physical state control by a proper disturbance compensation.

The design techniques assume the *separation of the real world model in a disturbance model and a physical state model* (fig. 3.3). During the derivation it turns out to be necessary that all physical states can be influenced separately by the control inputs corresponding to an invertible control input matrix D(t). This situation is satisfied in proportional navigation or in line-of-sight guidance by the application of regular or pseudo-inverse matrix calculation rules [21]. Using the notation of fig. 3.3 *a performance measure J including the physical states* $x_1(t)$ and the estimation errors $\tilde{x}_2(t) = x_2(t) - \hat{x}_2(t))$ can be formulated as follows:

$$J = E\{[x_1^T(t) \quad \tilde{x}_2^T(t)]\Theta(t) \begin{bmatrix} x_1(t) \\ \tilde{x}_2(t) \end{bmatrix}\}; \qquad \Theta(t) > 0 \quad . \tag{3.11}$$

The compensation control problem is formulated as a functional optimization problem:
The real world model is given by eq. 3.1 with an invertible input matrix $D_1(t)$. Based on a batch of measurements $z(\tau)$, $t_o \leq \tau \leq t$, find the control input $u = u(z(\tau), t)$ such that the performance measure J in eq. 3.11 is minimized.

The solution structure (fig. 3.4) is governed by a proportional feedback of the instantaneous measurements z(t) and a feedback of the disturbance estimates $\hat{x}_2(t)$

$$u = - K_1 z + K_3 \hat{x}_2 \tag{3.12a}$$

$$\dot{\hat{x}}_2 = (F_{22} - K_2 H_2) \hat{x}_2 + K_2 z \; ; \qquad \hat{x}_2(t_o) = x_{20} \quad . \tag{3.12b}$$

To determine the optimal parameter sets for the gain matrices $K_1(t)$, $K_2(t)$ and $K_3(t)$ the subsequent equations have to be solved:

$$K = \begin{bmatrix} K_1 \\ \hline K_2 \end{bmatrix} = D_1^{-1} P H^T R^{-1} \tag{3.13a}$$

with the covariance matrix P(t)

$$P = E\{\begin{bmatrix} x_1 \\ \tilde{x}_2 \end{bmatrix} [x_1^T \quad \tilde{x}_2^T]\} \tag{3.13b}$$

which is solution of the matrix Riccati differential equation

$$\dot{P} = F P + P F^T - P H^T R^{-1} H P + G Q G^T \; ; \qquad P(t_o) = P_o \; . \tag{3.13c}$$

The gain matrix $K_3(t)$ can be computed from

$$K_3 = - D_1^{-1} F_{12} + K_1 H_2 \; . \tag{3.13d}$$

The solution of the optimal disturbance compensation problem offers some *interesting properties:*

- There is a direct *equivalence* between the optimal compensation and optimal filtering problem.
- The *stability of the closed-loop system* can be proven in terms of controllability and observability.
- The solution requires the information of the *instantaneous measurements* z(t) only.
- The solution of the compensation problem *depends on system parameters* only. It can be computed in a one-design-step procedure by forward integration of (eq. 3.13c).
- The *realization* requires a state estimator of order $n_2$ only. From this point of view the controller makes a compromise between full-order and purely proportional feedback requirements.

## 4.1 Application of Optimal Parameter Determination Techniques to Line-of-sight Guidance Law Design

Guidance law design for line-of-sight guided missiles can advantageously be based on the structure of the kinematic guidance law (eq. 2.48) to realize the three-point guidance principle. An exemplified investigation for a modelwise surface-to-air guided missile illustrates

- the influence of an approximation of the primarily necessary coefficients in the kinematic guidance law and
- the performance improvement by applying optimal control techniques to the determination of a variable feedback gain factor $\beta(t)$ for a given feedback structure in the line-of-sight guidance law.

For the purpose of comparison an initial or conventional solution of the line-of-sight guidance problem is derived assuming approximate steering law coefficients and a constant feedback gain factor $\beta$. The essential design steps are as follows:

(i) Structural considerations about the guidance system design:

The structure of the conventional line-of-sight guidance system is governed by the available sensor equipement:

The reference for the line-of-sight motion is given by a target-tracking radar unit.

The missile deviation angle from the line-of-sight $\delta\varepsilon_m(t)$ is measured by a radar-mounted angle measuring device (e.g. a goniometer) with the relation $z_1 = z_1(\delta\varepsilon_m(t), \text{noise})$.

The line-of-sight angular rate $\dot{\varepsilon}_t(t)$ is measured by a radar-mounted angular velocity measuring device (e.g. a rate gyro) producing $z_2(t) = z_2(\dot{\varepsilon}_t, \text{noise})$.

To obtain necessary information about the line-of-sight angular acceleration $\ddot{\varepsilon}_t(t)$ information processing is performed in a differentiating network driven by the $\dot{\varepsilon}_t(t)$ - measurement $z_2(t)$. The output is the signal $z_3 = z_3(\ddot{\varepsilon}_t(t), \text{noise})$.

According to the structure of the kinematic guidance law (eq. 2.48) the commanded missile normal acceleration $a_{mn}(t)$ is a linear combination of the available information (assumption: $a_{mn}(t) \equiv 0$, $v_m = \text{const}$):

$$a_{mn} = \varkappa_1 \cdot z_3 + \varkappa_2 \cdot z_2 - k_2 \cdot z_1 \quad . \tag{4.1}$$

The primarily necessary coefficients in eq. 2.48 are approximated using a "mean missile range function" for the missile range: $\bar{r}_m(t) \approx \bar{r}_{mo} + \bar{v}_m \cdot (t-t_o)$, and the related assumption of only small lead angles $\bar{\varphi}_m(t)$: $\cos\bar{\varphi}_m(t) \approx 1$. Applying these approximations the guidance law coefficients are given by

$$\varkappa_1 = \bar{r}_{mo} + \bar{v}_m \cdot (t - t_o) \; ; \qquad \varkappa_2 = 2\,\bar{v}_m \tag{4.2a}$$

$$k_2 = \beta \cdot [\bar{r}_{mo} - \bar{v}_m \cdot (t - t_o)] \; ; \qquad \beta = \text{const} \quad . \tag{4.2b}$$

The constant gain factor $\beta$ of the feedback portion is determined in a subsequent design step.

(ii) A real world simulation model:

To evaluate the system performance it is necessary to set up a simulation model as a "sufficient" image of the real world. In this case the following real world effects are included in the real world simulation model (fig. 4.1):

- Normally distributed initial conditions of missile motion to describe the uncertainties of launch- and boost-phase.
- A second-order model to approximate the missile dynamics.
- A limitation on the commanded missile normal acceleration.
- Coloured and white noise modelling with regard to sensor noise: glint noise $\varepsilon_{gl}(t)$, thermal noise $\varepsilon_{th}(t)$, random bias terms $b_\varepsilon(t)$, $b_{wk}(t) = \text{const}$ and broad-band gyro noise $\eta_{wk}(t)$.
- Different target engaging conditions (fig. 4.2).

The simulation model represents a system of nonlinear differential equations (of order 13) which is driven by white Gaussian noise. Since the distribution of the terminal miss distance $d_m(t_f)$

$$d_m(t_f) = \delta\varepsilon_m(t_f) \cdot \bar{r}_m(t_f) \tag{4.3}$$

is (at least approximately) Gaussian distributed, the stochastic linearization approach [21] offers an attractive tool for an analytical solution of the nonlinear stochastic analysis problem on a digital computer. It provides for the mean value $\bar{d}_m(t_f) = E\{d_m(t_f)\}$ and the standard deviation $\sigma_d(t_f) = E\{[d_m(t_f) - \bar{d}_m(t_f)]^2\}$ to characterize the distribution of the terminal miss distance.

(iii) Definition of a system performance measure:

The simulation of a guidance law with regard to different target engaging situations denoted by the index i produces several numbers of distribution characteristics $\bar{d}_m^i(t_f)$ and $\sigma_d^i(t_f)$. To compare the system performance for distinct guidance laws it is useful to introduce a system performance measure $\Pi = \Pi(\bar{d}_m^i(t_f), \sigma_d^i(t_f))$ which transforms the performance information to one performance value. A meaningful performance measure $\Pi$ is given by

$$\Pi = \frac{1}{N} \sum_{i=1}^{N} [|\bar{d}_m^i(t_f)| + \sigma_d^i(t_f)]/L_t \qquad (4.4)$$

N : number of engaging situations;

$L_T$ : target expansion measure, e.g. mean visible length.

(iv) Constant guidance law parameter determination:

A parametric simulation analysis with respect to the feedback parameter $\beta$ indicates the system performance behaviour $\Pi(\beta)$ illustrated in fig. 4.3. The parameter $\beta = \beta_{opt}$ is chosen as guidance law parameter to obtain "optimality in the sense of the performance measure $\Pi$" under the assumption of a constant feedback parameter $\Pi$ = const. The distribution parameters of the terminal miss distance are shown in table 4.1, case 1.

To improve the system performance a considerable design step is to remove the assumption of a constant parameter $\beta$. Hence it is necessary to apply a suitable control design technique producing a time-varying parameter $\beta = \beta(t)$. To solve this problem of proportional feedback parameter determination, the (optimal) proportional feedback technique in section 3.2 can be used. Since it provides for "optimality in the sense of minimum state variances" it can be viewed as a systematic design techniques in the sense of the performance measure $\Pi$ in eq. 4.4. The design procedure consists of the following design steps:

(i) Design model considerations:

The output feedback design techniques (eqs. 3.8 ÷ 3.10) is based on a real world model (eq. 3.1). Since the posed guidance parameter design problem is concerned with the feedback portion of the guidance law, the real world model corresponds to a linearized deviation model from the kinematic conditions of the three-point guidance principle (eqs. 2.37). To obtain optimality in the sense of the mean square error performance measure (eq. 3.9) a sophisticated design model is necessary inducing high computational on-line realization effort. To satisfy the given design specification of low on-line realization effort a "simple, but sufficient design model" of the real world is to be set up providing for a suboptimal feedback parameter $\beta(t)$ with satisfying system behaviour.

A sensitivity analysis of the system performance with respect to the real world effects of initial condition errors, missile dynamics and sensor noise leads to the following design model:

- It contains the state variables of the lead angle deviation $x_1(t) = \delta\varphi_m(t)$ and the missile line-of-sight deviation angle $x_2(t) = \delta\varepsilon_m(t)$. To derive a deviation model for these state variables the associated nonlinear differential equations for $\varphi_m(t) = \theta_m(t) - \varepsilon_m(t)$ and $\varepsilon_m(t)$ are set up from eq. 2.38:

$$\begin{aligned} \dot{\varphi}_m &= - v_m \cdot r_m^{-1} \sin\varphi_m + v_m^{-1} a_{mn} \\ \dot{\varepsilon}_m &= v_m \cdot r_m^{-1} \sin\varphi_m \quad . \end{aligned} \qquad (4.5)$$

Linearization by a first-order Taylor expansion yields the deviation model in terms of the previously introduced state variables $x_1(t)$ and $x_2(t)$. Because of formal consistency between the control law: $u(t) = K(t)\, z(t)$ in eq. 3.8 and the feedback guidance law $\delta a_{mn} = [\bar{r}_m(t)/\cos\bar{\varphi}_m(t)] \cdot \beta(t) \cdot z_1(t)$ the part [.] is associated with the control input matrix. Hence the control input is defined by $u(t) = [.]^{-1} \cdot \delta a_{mn}(t)$. The deviation model is given by

$$\begin{bmatrix} \dot{x}_1 \\ \dot{x}_2 \end{bmatrix} = \begin{bmatrix} -\bar{v}_m \cdot \bar{r}_m^{-1} \cdot \cos\bar{\varphi}_m & 0 \\ \bar{v}_m \cdot \quad \cdot \cos\bar{\varphi}_m & 0 \end{bmatrix} \begin{bmatrix} x_1 \\ x_2 \end{bmatrix} + \begin{bmatrix} \bar{r}_m \cdot [\bar{v}_m \cdot \cos\bar{\varphi}_m]^{-1} \\ 0 \end{bmatrix} u + \begin{bmatrix} \text{system noise} \\ 0 \end{bmatrix} \qquad (4.6a)$$

$$z_1 = x_2 + \text{measurement noise} \; . \qquad (4.6b)$$

In this notation the parameter determination problem of the guidance feedback law is equivalent to the output feedback control problem

$$u = \beta \cdot z_1 \quad , \qquad (4.6c)$$

such that the system matrices of the control technique can be obtained by comparison of eq. 4.6 and eqs. 3.1, 3.8.

- The deviation model is not extended to include missile dynamics or coloured noise shaping filters by reason of low realization effort.
- The noise terms in the deviation model (eq. 4.6) give regard to disturbances in the steering law measurements ($z_2$ and $z_3$) and to disturbances in the feedback law measurement $z_1$. They are modelled approximately by white Gaussian noise. The spectral densities of glint and thermal noise include the target range dependency, i.e. the design model depends on the different engaging situations.
- The weighting matrix $\Theta(t)$ provides for free parameters of the design procedure. They are weighted by increasing functions of the time to express the increasing demand on system accuracy during missile-target approach.

Summarizing the design model consists of the system matrices in eq. 4.6, approximately modelled noise statistics and a free parameter containing weighting matrix.

(ii) Guidance solution based on an approximated design model:

For a first guidance law design using the control technique (eqs. 3.8 ÷ 3.10) the design model is based on the approximations $\bar{r}_m(t) \approx \bar{r}_{mo} + v_m \cdot (t-t_o)$ and $\cos\bar{\varphi}_m(t) \approx 1$. Hence only the target range $r_t(t)$ is needed as additional information for target-depending modelling of the noise statistics. Due to the principal character of the investigation target range $r_t(t)$ is assumed to be measured noise-free.

To find suitable free design parameters in the weighting matrix $\Theta(t)$ an iterative design procedure leads to feedback parameter functions $\beta(t)$ which are illustrated in <u>fig. 4.4, case 2</u> for the three engaging situations under investigation. The associated <u>distribution parameters</u> of terminal miss distance are given in <u>table 4.1, case 2</u>. Essentially it can be stated that a performance improvement <u>of about 23% is achieved</u> by

- calculating a time-dependent parameter $\beta(t)$ by an on-line algorithm with low realization effort (second-order design model) and
- the target range $r_t(t)$ as additional information.

(iii) Guidance solution based on a lead angle adapted design model:

A further improvement of system performance can be achieved if the functions $\bar{r}_m(t)$ and $\cos\bar{\varphi}_m(t)$ can be realized. This enables

- a lead angle adapted design model (eq. 4.6) and
- lead angle adapted guidance law parameter according to the kinematic guidance law (eq. 2.48):

$$\varkappa_1 = \bar{r}_m(t)/\cos\bar{\varphi}_m(t) \quad ; \qquad \varkappa_2 = 2\,\bar{v}_m \tag{4.7a}$$

$$k_2 = \beta(t) \cdot \bar{r}_m(t)/\cos\bar{\varphi}_m(t) \; . \tag{4.7b}$$

The functions $\bar{r}_m(t)$ and $\bar{\varphi}_m(t)$ can be evaluated e.g. by an integrating function generator

$$\dot{\bar{\varphi}}_m = -\bar{v}_m\,\bar{r}_m^{-1}\cos\bar{\varphi}_m + \bar{v}_m^{-1}\,\bar{a}_{mn} \quad ; \qquad \bar{\varphi}_m(t_o) = \bar{\varphi}_{mo} \tag{4.8a}$$

$$\dot{\bar{r}} = \bar{v}_m \cdot \cos\bar{\varphi}_m \quad ; \qquad \bar{r}_m(t_o) = \bar{r}_{mo} \tag{4.8b}$$

$$\bar{\varphi}_{mo} = \arcsin\{\dot{\varepsilon}_t(t_o)\,\bar{r}_{mo}\,\bar{v}_m^{-1}\} \quad . \tag{4.8c}$$

The initial line-of-sight angular rate $\dot{\varepsilon}_t(t_o)$ can be obtained from the initial $z_2(t_o)$-measurement. The term $\bar{a}_{mn}(t)$ representing the steering law acceleration in eq. 2.48 can be approximated by the feedforward term in the guidance law: $\bar{a}_{mn}(t) \approx \varkappa_1(t) \cdot z_3(t) + \varkappa_2 \cdot z_2(t)$. Hence this solution approach does not require additional information but additional realization effort.

Repeating the iterative design procedure a different set of free parameters in the weighting matrix $\Theta(t)$ produces the parameter function $\beta(t)$ in <u>fig. 4.4</u>, <u>case 3</u> and the distribution parameters in <u>table 4.1</u>, <u>case 3</u>. The further performance improvement of about 16% to total 35% can <u>be explained by</u> the following:

- The mean terminal miss is reduced due to the adapted steering law.
- The improved steering behaviour necessitates a less feedback gain (fig. 4.4) such that the standard deviation of the terminal miss distance decreases.

This result corresponds to the familiar control design experience of unburdening the feedback control by a proper feedforward control to obtain improved system performance.

### 4.2 Application of Optimal Filtering and Control Techniques to Proportional Navigation Guidance Law Design

As shown in chapter 2.2 proportional navigation can be treated within the framework of the kinematic guidance law of the two-point guidance principle. Furthermore it was pointed out that the feedback part of the guidance law can be extended with regard to real world effects. In this section an exemplified investigation illustrates the possible performance improvement by applying the Gaussian, quadratic optimal control technique (chapter 3.2) with special regard to missile dynamics, stochastic noice and target manoeuvres. In addition the problems of final-time dependency and realization effort are emphasized.

To simulate different guidance loops the *real world model* as shown in fig. 4.5 is used. Essentially it is characterized by the nonlinear kinematics, which are driven by the states of target and missile motion. Two target engaging situation are considered:

case I: $r(t_o) = 3000$ m ; $\sigma(t_o) = 1.9^o$

$v_m(t_o) = v_m(t) = 750\ ms^{-1}$ ; $a_m(t_o) = 0$ g ; $\Theta_m(t_o) = \sigma(t_o)$ ;

$v_t(t_o) = v_t(t) = 250\ ms^{-1}$ ; $a_t(t_o) = a_t(t) = 0$ g ; $\Theta_m(t_o) = 180^o$ ;

case II: $r(t_o) = 2000$ m ; $\sigma(t_o) = 0^o$ ;

$v_m(t_c) = v_m(t) = 750\ ms^{-1}$ ; $a_m(t_o) = 0\ m/s^2$ ; $\Theta_m(t_o) = \sigma(t_o)$ ;

$v_t(t_o) = v_t(t) = 250\ ms^{-1}$ ; $a_t(t_o) = 5$ g ; $\Theta_m(t_o) = 180^o$ .

The missile dynamics are characterized by a linear second-order model with a constraint on the commanded acceleration input. The sensor equipment (fig. 2.7) consists of a target-tracking radar unit providing for measurements of the line-of-sight angular rate $\dot{\sigma}(t)$ and of the closing velocity $\dot{r}(t)$. A second-order model approximates the servo dynamics of the sensor.

Glint noise $\varepsilon_{gl}(t)$ and thermal noise $\varepsilon_{th}(t)$ are modelled as coloured and white noise respectively with range depending statistics. The closing velocity $\dot{r}$ is assumed to be measured noise-free. The simulation applies the stochastic linearization approach [21].

For reasons of comparison a *conventional guidance law design* is performed using the proportional navigation law

$$a_{mn} = - \Lambda_n \cdot \dot{r} \cdot \dot{\sigma} \quad . \tag{4.9}$$

According to the considerations of section 4.1 about the comparison of different guidance laws and the design under various flight conditions the performance measure $\Pi$ in eq. 4.4 is used to optimize the system behaviour and to compare the results. Minimizing the performance measure $\Pi$ with respect to the navigation constant $\Lambda_n$ as the only free parameter in this design case the terminal miss distribution parameters shown in table 4.2, case 1 are obtained.

To improve the system performance by *extending the proportional navigation law* with respect to the real world effects as shown in fig. 4.5 the approach of the Gaussian, quadratic optimal control technique is used. As a realization requirement the computational effort is to be kept low.

Following the solution of the controller design technique in section 3.2 the design procedure is separated in a filter and a control law design.

To get estimates $\hat{x}(t)$ of the real world states the *Kalman-Bucy-Filter algorithm* (eq. 3.3) is applied. It uses a real world model of order 8 containing the kinematic states $\sigma(t)$ and $\dot{\sigma}(t)$, the missile and servo dynamic states and the glint noise. In addition a first-order Markov-model is included for target acceleration modelling [21]. To avoid the singular filter case the measurement $z(t)$ is superposed by an artificial white noise. Since $\dot{\sigma}$ is described by the nonlinear differential equation (2.3) a first order Taylor expansion about the nominal conditions (2.9) assuming constant target- and missile velocities was done. Therefore the filter model requires information about the nominal conditions $\bar{r}(t_o)$, $\bar{\sigma}(t_o)$, $\bar{\Theta}_t(t_o)$ and the velocity ratio $\varkappa_v$. The nominal range $\bar{r}_m(t)$ is approximated by $\bar{r}_m(t) = \bar{r}_m(t_o) + \dot{r}(t-t_o)$. Due to the principal character of the example possible considerations about low-order and/or constant gain filter design are beyond the scope of this section.

To avoid the disadvantage of the *quadratic optimal control technique*, i.e. the final time dependency of the solution, requiring high on-line computational effort in the case of changing final time, it is suggested to use a flight-constant, but initial condition depending feedback vector $k_c(t_o)$ for state estimate feedback. Hence the extended guidance law is given by

$$a_{mn}(t) = k_c^T(t_o)\ \hat{x}(t) \quad . \tag{4.10}$$

The design problem consists of finding one suitable set of weighting matrices for the control technique (eq. 3.5), such that the pre-launch computed feedback vector $k_c(t_o)$ causes a satisfying system behaviour in the sense of the performance measure $\Pi$ in eq. 4.4. A suitable set of weighting matrices was found by trial and error. The resulting feedback matrices for the considered flight situations are shown in table 4.2, case 2. They combine the estimates of line-of-sight angular rate, of missile dynamic states and of target manoeuvre to produce the guidance command (eq. 4.10). The improved system performance of about 45% can be briefly explained by the following

- The mean terminal miss is reduced due to fast target manoeuver detection by the filter combined with a damped flight behaviour due to the control law.
- The standard deviation is diminished by the low-pass character of the filter.

This shows the efficiency of applying optimal control techniques to the considered guidance law design problem.

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| design case | target condition | | | | | performance measure ($L_t$ = $10_m$) |
|---|---|---|---|---|---|---|
| | 1 | 2 | | 3 | | |
| | $\sigma_d(t_f)$ | $\bar{d}_m(t_f)$ | $\sigma_d(t_f)$ | $\bar{d}_m(t_f)$ | $\sigma_d(t_f)$ | $\Pi$ |
| I | 3.30 m | - 3.60 m | 4.95 m | - 0.05 m | 3.00 m | 1.0 |
| II | 2.45 m | - 2.65 m | 3.70 m | - 0.20 m | 2.55 m | 0.77 |
| III | 2.25 m | - 1.75 m | 3.51 m | 0.0 m | 2.15 m | 0.65 |

Table 4.1 Numerical results for line-of-sight guidance law comparison

| | case 1 | | case 2 | |
|---|---|---|---|---|
| | engaging condition | | engaging condition | |
| | I | II | I | II |
| Control Gain | $\Lambda_{opt} = 4.6$ | | $k_c = \begin{bmatrix} 0 \\ 5234 \\ -15.3 \\ -1.4 \\ 0 \\ 0 \\ 0 \\ -1.5 \end{bmatrix}$ | $k_c = \begin{bmatrix} 0 \\ 5995 \\ -24.6 \\ -2.3 \\ 0 \\ 0 \\ 0 \\ -1.9 \end{bmatrix}$ |
| mean terminal miss [m] | 1.4 | 1.3 | 0.0 | 0.1 |
| standard deviation terminal miss [m] | 3.1 | 4.4 | 2.77 | 2.74 |
| performance measure $\Pi$, ($L_t$ = 10 m) | 1.02 | | 0.56 | |

Table 4.2 Comparison of conventional proportional guidance law (case 1) and optimal control techniques (case 2)

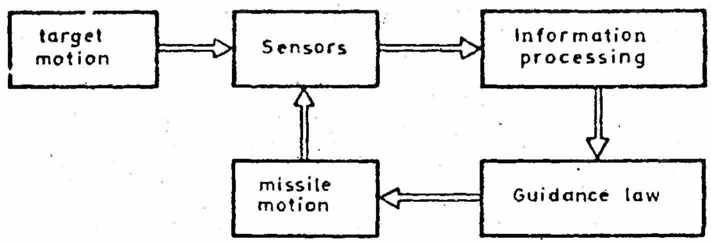
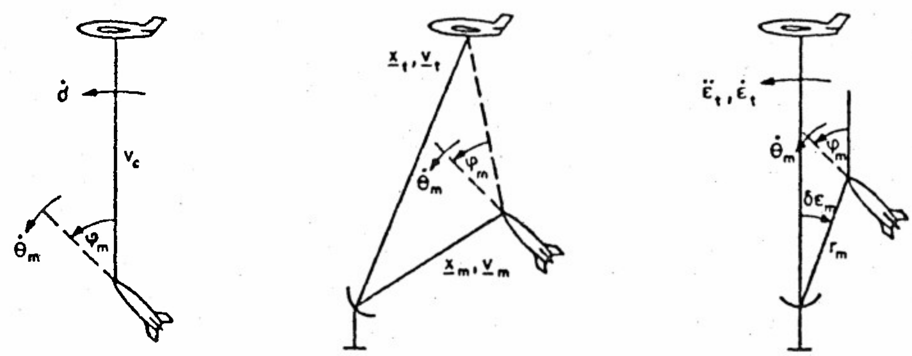



Figure 1.1: Structure of information flow in missile guidance loops

Proportional Navigation: | Constant Bearing: | Line-of-sight Guidance:




**Proportional Navigation:**

- line-of-sight rate $\dot{\sigma}$ closing velocity $v_c$
- missile homing sensor: radar target-tracker, Doppler-radar
- complex onboard processing
- medium acceleration demands
- two-point guidance

**Constant Bearing:**

- position and velocity vectors $\underline{x}_{t,m}$, $\underline{v}_{t,m}$
- missile and target information sensors
- complex onground processing
- low acceleration demands
- two-point guidance (three-point realization)

**Line-of-sight Guidance:**

- line-of-sight rate and acceleration $\dot{\varepsilon}_t$ and $\ddot{\varepsilon}_t$, missile deviation angle $\delta\varepsilon_m$, missile range $r_m$
- target-tracking sensor, missile deviation angle sensor
- complex onground processing
- high acceleration demands
- three-point guidance

Figure 1.2: Principal information and acceleration demands of guidance laws

| | | | | |
|---|---|---|---|---|
| kinematic world | guidance principle → overall guidance problem | | | |
| | steering problem | feedback problem | guidance solution | |
| | • nominal conditions<br>→ steering law | • stability<br>→ stab. feedback | kinematic guidance law = steering law + stab. feedback | guidance law = feedforward + feedback control |
| real world | | • satisfying system performance under real world conditons<br>→ information processing<br>→ feedback control | | |

Figure 1.3: Structuring of the overall guidance problem

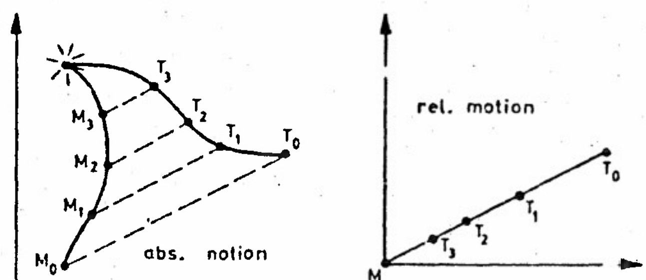



Figure 1.4a: Two-point guidance principle

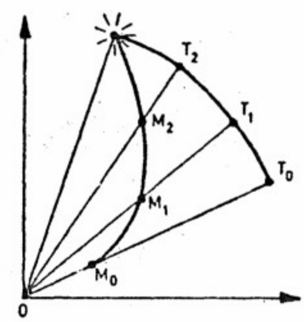


Figure 1.4b: Three-point guidance principles

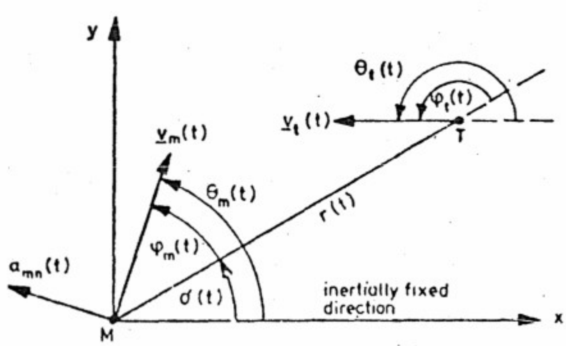



Figure 2.1: Definitions of the kinematic variables for the relative motion between the missile M and the target T

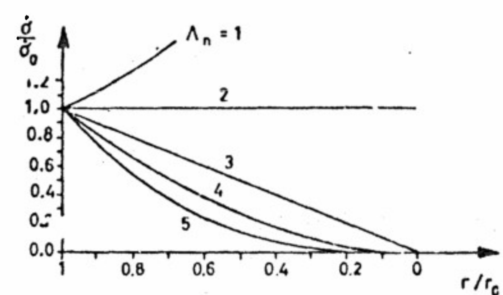


Figure 2.2: Line-of-sight angle rate behaviour for proportional navigation

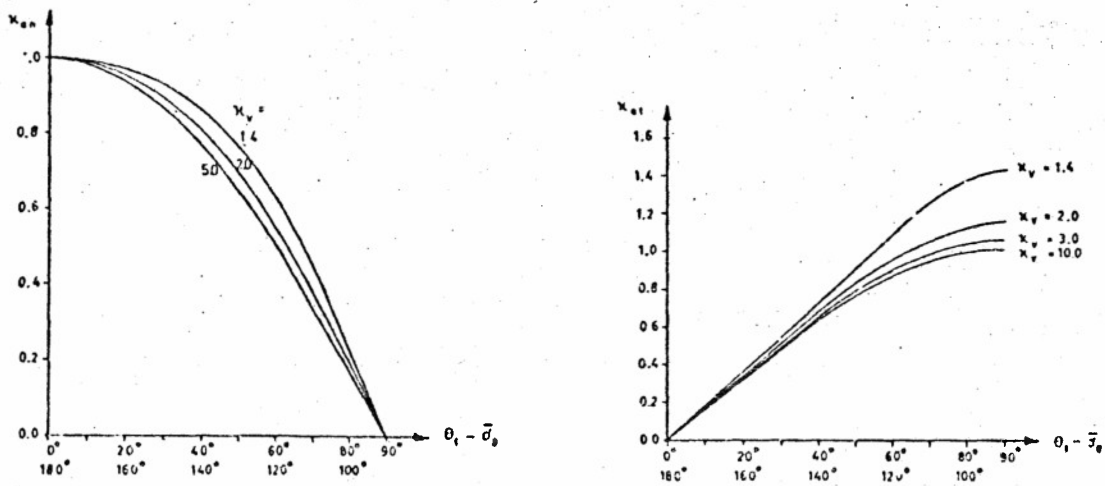



Figure 2.3: Acceleration ratios due to target manoeuvres

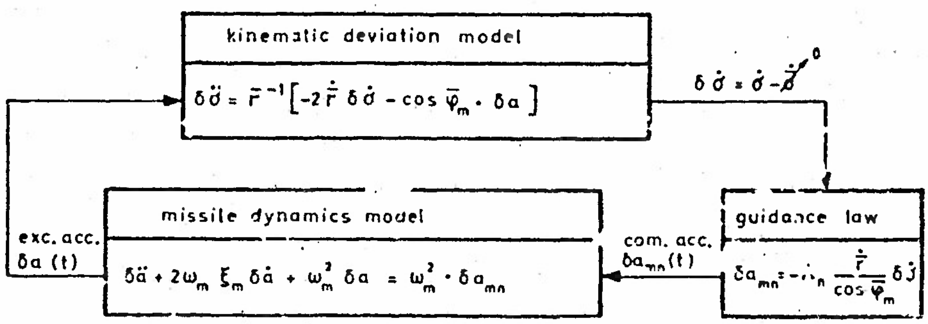



Figure 2.4: Real world model with regard to missile dynamics

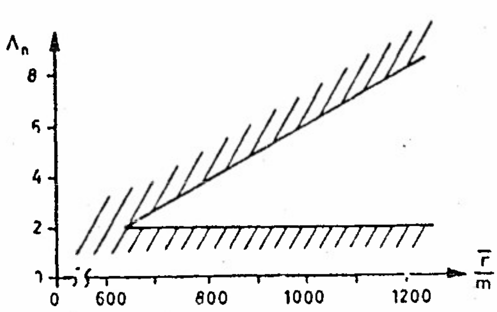



Figure 2.5

Exemplified region for admissible navigation constants ($\omega_m = 5\ s^{-1}$, $\xi_m = 1$, $v_c = 800\ m\ s^{-1}$)

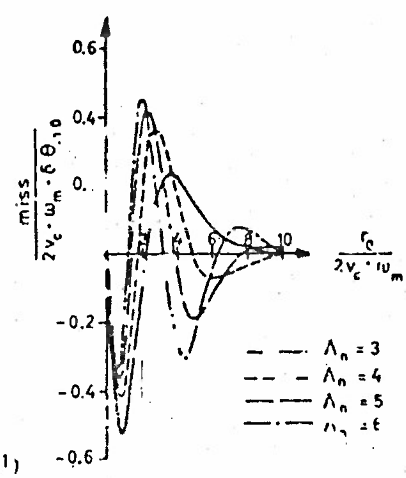



Figure 2.6

Terminal miss distance due to missile dynamics (second-order model with $\xi_m$ = 1.0)

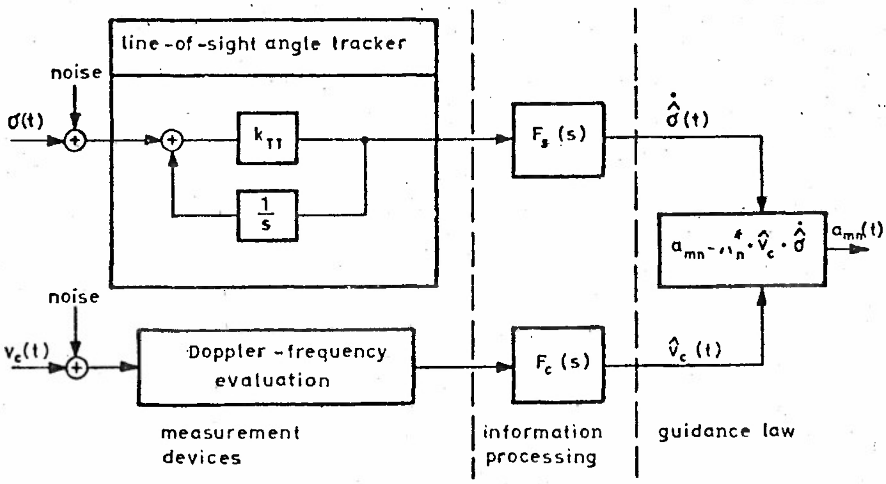



Figure 2.7 Exemplified realization of proportional navigation

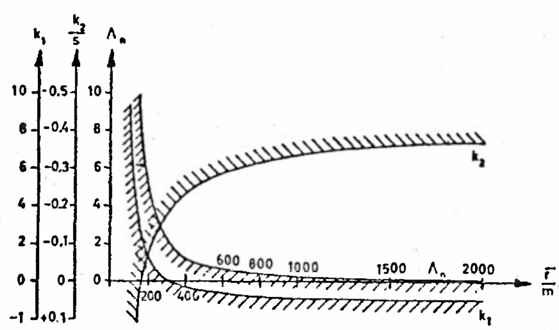



Figure 2.8 Exemplified bounds on stability feedback gains for extended stabilizing feedback law of proportional navigation

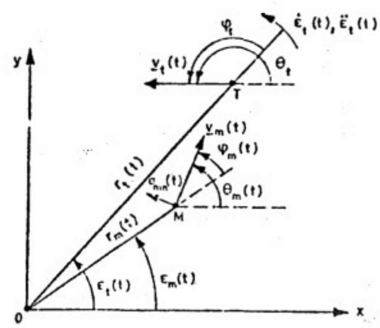



Figure 2.9 Definitions of kinematic relations for three-point guidance

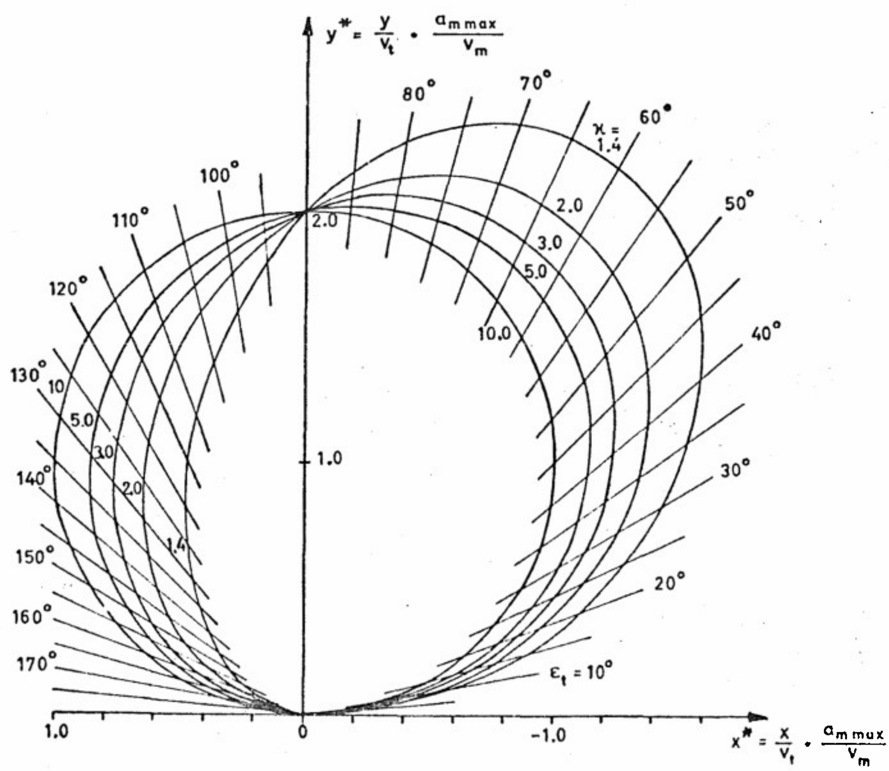



Figure 2.10 Bounds of interception for three-point guidance with constrained missile normal acceleration $|a_{mn}(t)| \le a_{m\,max}$

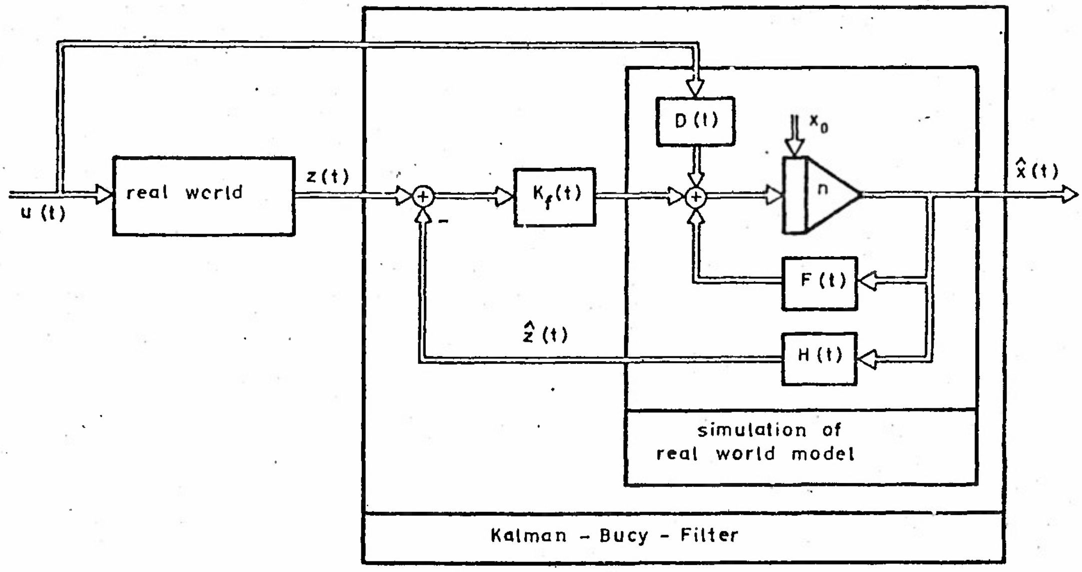



Figure 3.1 Structure of information processing by linear filtering techniques

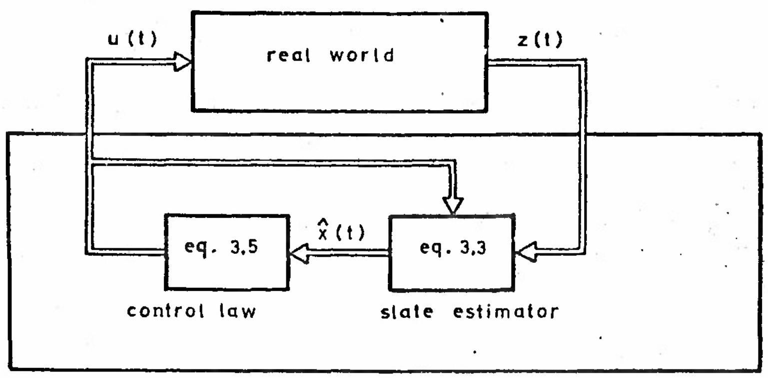



Figure 3.2 Closed-loop structure of Gaussian, quadratic optimal control

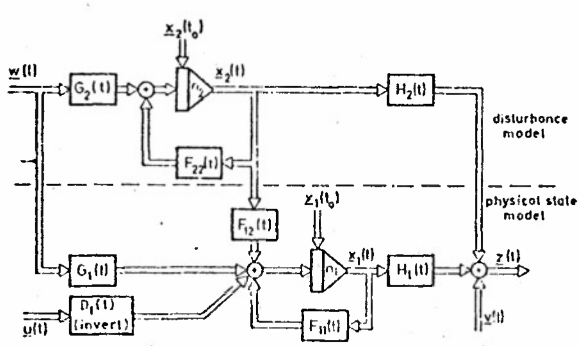
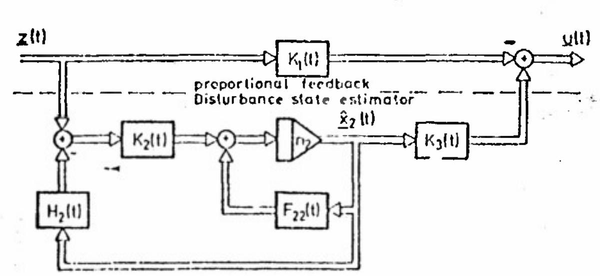



Figure 3.3

System structure for variance optimal disturbance compensation

Figure 3.4

Structure of low-order disturbance compensator



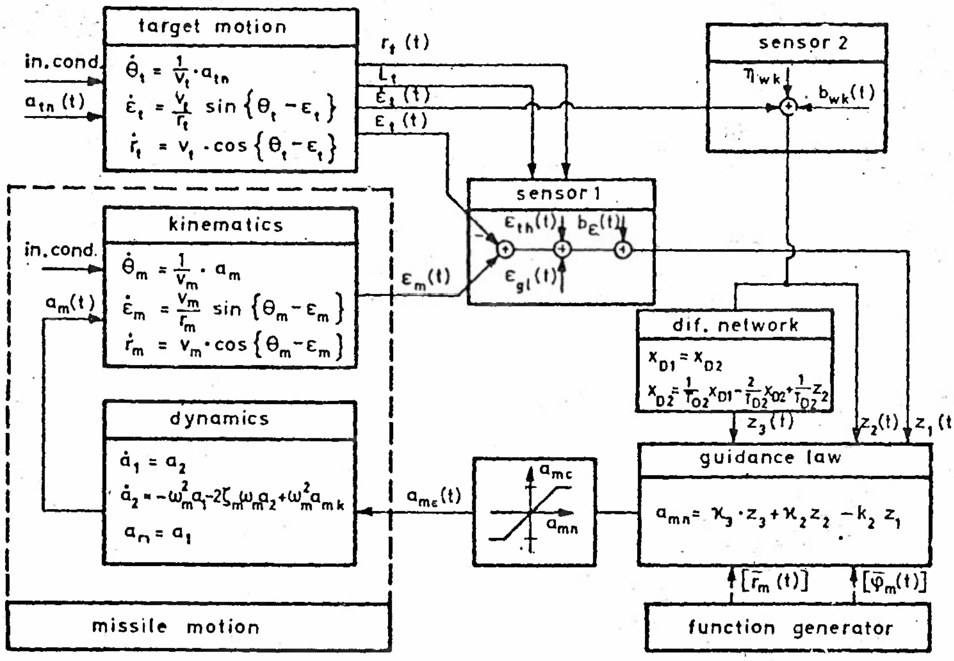



Figure 4.1 Simulation model for line-of-sight guidance laws

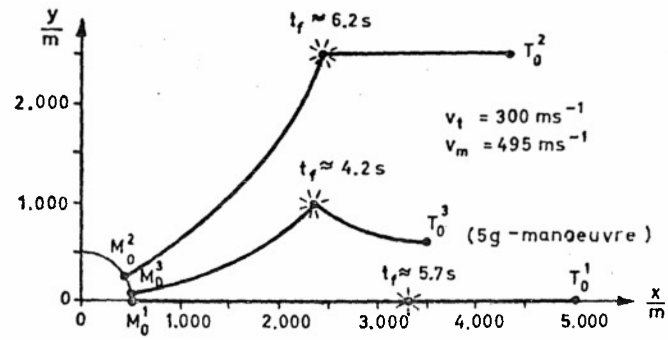



Figure 4.2 Enganging situations for guidance law design and system performance evaluation

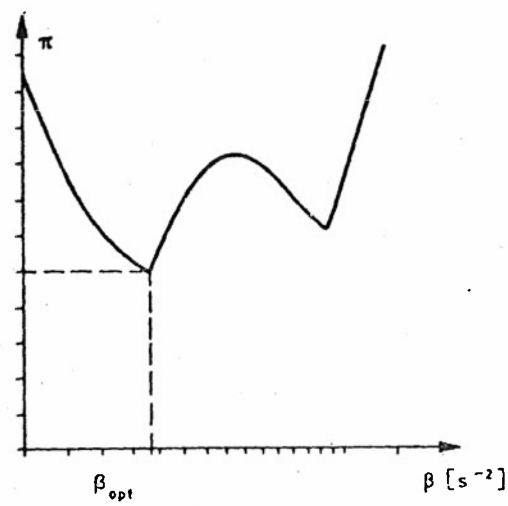



Figure 4.3 Performance measure $\Pi$ as function of the constant feed-back parameter $\beta$

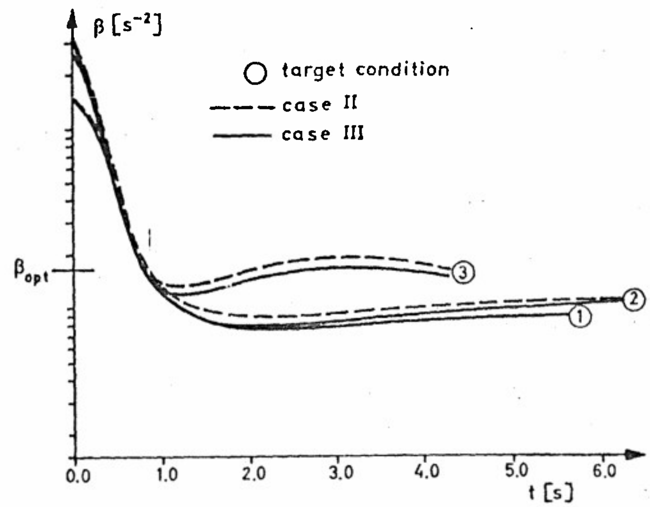



Figure 4.4 Time-varying feedback parameter functions $\beta(t)$

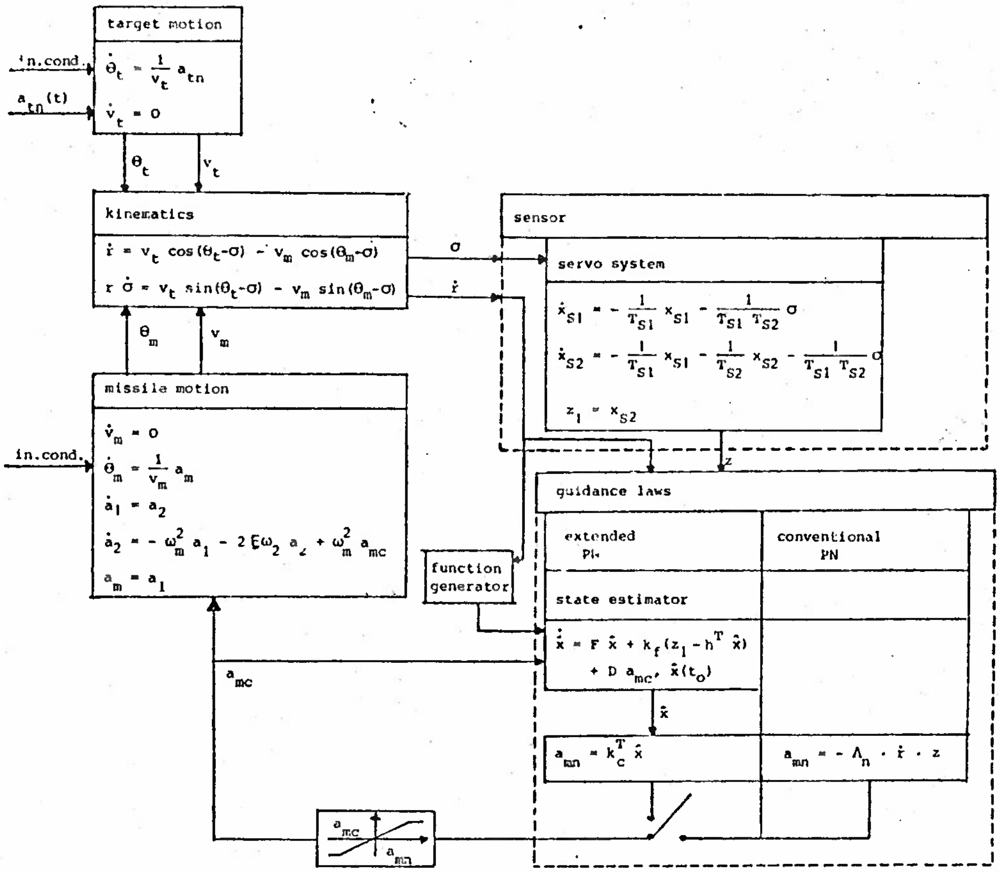



Fig. 4.5 Simulation model for proportional guidance laws

# TESTING OF MISSILE GUIDANCE AND CONTROL SYSTEMS

by
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## SUMMARY

Improved techniques of testing and data recovery which permit accurate math modeling of flight systems have been an important adjunct to the increased growth of simulation. Early simulations tested individual components such as the inertial instruments in physical environments, but as terminal guidance sensors developed, interaction between the physics of the sensed information and the system led to increased ground test complexity.

Television, semi-active laser, infrared, and radar simulation rooms are being grouped around a central computer facility which can service each in turn and which can provide some limited functions for each simultaneously. In some cases these technologies are combined. As an illustrative example, in this paper an improved radar guidance laboratory which allows simultaneous infrared simulations for developing and testing point tracker radar and IR dual mode guidance systems is described. These guidance systems must be tested for target acquisition, discrimination, and tracking capabilities under precisely controlled conditions in a dynamic, real-time, simulated environment. The radar guidance types can be passive at 3 to 5 or 8 to 14 microns.

A short review of system requirements is furnished, and the major laboratory subsystems are described, with emphasis on the features of the rotational and translational motion systems, anechoic chamber, linear array target antenna system, radar generation system, IR target system, and computation. The principal new design features of this laboratory are the linear array target antenna system and the radar generation system, which provides for four distinct radar emitters, each of which can simulate simultaneous, independent RF sources. These sources can be surveillance, surface-to-air missiles (SAM), search or early warning radars, plus radar returns from illuminated targets, and all types of pulsed and continuous wave ECM signals. Phenomena, such as atmospheric attenuation, Doppler shift, target cross section deviation, and glint are also simulated. Criteria used to specify the required system performance, the reasons for criteria selection, and the laboratory test results are also included.

The costs of acquisition and operation are also identified. An important consideration is the expected life of the facility. Technology is continually changing and five to ten years is the limit of useful life without modifications. Examples of obsolescence and modifications of an electro-optic laboratory are given.

## INTRODUCTION

Most simulators have been built specifically to save money, even though it is rather difficult to determine the cost tradeoff. However, if the simulator is to solve a problem that would involve considerable risk to flight vehicles, or if it were extremely difficult to provide the tactical environment, then the discussion on cost effectiveness can be put aside and only the cost of the simulator is of concern.

Nowhere is this more true than in the case of the missile point tracker homing radar guidance system. The extreme difficulty of testing critical performance factors of modern airborne point tracker homing radar guidance systems in actual missile flight tests makes implementation of extensive flight test programs essentially impossible. If such flight test programs could be implemented, cost would be prohibitive. The technical difficulty lies in the fact that realistic, high performance multiple targets, including decoys, cannot be repeatedly provided to test radar seeker guidance system acquisition, tracking, and discrimination capabilities under controlled conditions. Examples of radar-guided missile systems which have multiple air target discrimination and maneuvering high performance target capabilities are strategic long range bomber defense missiles, long range tactical air defense missiles, and advanced tactical air-to-air missiles. The more sophisticated the targets which these weapon systems must engage, the higher closure rates, and higher target maneuvering rates, make safe, realistic flight tests all but impossible. Table I presents a partial list of technical problem areas which require experimental evaluation for advanced air target guidance systems, in the proper context of the intended mission and anticipated RF and tactical environments.

A point tracker homing guidance system tracks a point source which appears to be located at infinity and can be electro-optical, laser, infrared, or RF types, or combinations of these. The RF point tracker guidance system can be passive, semi-active, or active. Specific characteristics follow.

1. Passive system - The missile homes on a target which is emitting a radar signal, such as a tracking station. Here only the characteristics such as frequency, antenna pattern, and signal strength need to be provided.

2 Semi-active system - The target is illuminated by a source external to the missile. Again, a signal duplicating that emanating from the target must be furnished for the missile. But in addition to this (for most semi-active systems), pulses of the proper frequency, timing, etc., which the missile would receive from the ground station must be supplied.

3 Active system - The frequency, timing, etc., of the return pulses must be derived from the radar transmitter aboard the missile.

TABLE I

Typical Mission Requirements not Capable of Evaluation Using Flight Tests

| Mission Requirements | Potential Technical Problem for Guidance |
|---|---|
| 1. Home on target | Main lobe clutter rejection<br>Guidance signal crosstalk |
| 2. High closure rate | Maneuver limits<br>Acquisition sensitivity<br>Radome induced anomalies |
| 3. Low altitude impact | Multipath guidance |
| 4. Formation targets | Target discrimination<br>Guidance processing logic |
| 5. All weather | Guidance noise |
| 6. High altitude | Long range acquisition<br>Receiver dynamic range<br>Side lobe clutter rejection<br>Receiver sensitivity |
| 7. ECM | Target discrimination logic<br>Miss distance |

Further, in addition to the single return from the target, which should include such effects as scintillation, glint, and atmospheric attenuation (e.g., rain), other elements of the environment may be included. There may be multiple targets or decoys, ground clutter, multipath returns, and a number of simulated ECM signals.

The RF point tracker system operates as follows:

Use - Provide update target position information.

Techniques - Active tracking in range, angle, or Doppler (or range, angle, and Doppler), or passive tracking in angle, frequency, PRF, etc., of emitting targets.

Unique Processing - Clutter, multipath, glint, ECM, multi-targets, etc., rejection, or discrimination, or rejection and discrimination.

Problems -

Glint
Scintillation
Multipath
Sensitivity
Clutter
Electromagnetic environment
Multiple target environment
Sensor errors
Kinematics (scenario and platform).

A further complication is that advanced guidance systems may be multimode and therefore able to hand-over from one guidance mode to another (passive RF to active RF, or RF to IR, etc.) on different frequency bands.

Four elements of simulation are required to subject a missile guidance system to a simulated environment which approximates a large part of the flight envelope.

1 A target which the missile can fly against in as near a real-world environment as possible.

2 A seeker system to track the target and issue guidance commands.

3 Computers to convert the guidance signals to control surface positions, simulate the aerodynamics and kinematics of the missile, and control the missile environment.

4 A transport system to maintain the spatial relationship of the missile with respect to the target.

During the 1970's many government and commercial organizations designed and built testing facilities in an attempt to furnish valid simulations. One of these, still in operation, was built by Martin Marietta at Orlando in 1966. The facility allows the designer to:

1 Determine the guidance system characteristics;

2 Measure the guidance system performance;

3 Evaluate and optimize the seeker performance over the extremes of its environment;

4 Integrate the guidance system with the missile prior to putting hardware in the field and well in advance of flight testing.

The guidance development center has been continually expanded and consists of an electro-optical guidance laboratory, an RF/IR point tracker guidance laboratory, an RF correlator guidance laboratory, a pilot display laboratory, and a computer laboratory. This paper will discuss the RF/IR guidance laboratory, with primary emphasis on the RF aspects.

The RF/IR guidance laboratory was recently expanded to provide a capability of developing and testing advanced point tracker RF and IR guidance systems well into the 1980's. The overall simulation requirements generated during an extensive preliminary design study are:

1 Perform full mission, real-time, closed-loop dynamic simulation.

2 Verify missile system capabilities to search for, acquire, and accurately track targets in the presence of ECM and evasive flight tactics.

3 Test critical performance factors of modern air target homing radar and IR guidance systems.

4 Allow full mission hand-off from one guidance mode to another.

5 Perform simulation of active, semi-active, and passive radar systems.

6 Test under repeatable conditions.

7 Perform system integration and checkout prior to flight test.

All of these requirements are important; however, items 2 and 3 really represent the main reasons for simulation.

THE RF/IR GUIDANCE LABORATORY AND ITS MAJOR SUBSYSTEMS

1. General

Figure 1 shows the original layout of the guidance development center. The radar guidance laboratory interfaces with the computer laboratory and operates independently of the rest of the GDC. A block diagram of a point tracking sensor is shown in Figure 2. The sensor uses either a conical scan or a monopulse system to keep the antenna gimbal pointed at the target. Gyros mounted on the gimbal and pickoffs on the gimbal are used to provide steering signals to the missile. This type of tracking is in wide use with conventional airborne radars to track targets which can be characterized essentially as point sources of radiation.

To understand in more detail how this is accomplished, consider the four simulation elements previously mentioned. For the target, the laboratory must furnish a radar signal which "looks" like the signal the seeker is designed to track. To provide the RF environment, it is necessary to have a shielded anechoic chamber and an extensive RF generation capability. The shielding isolates the system from all the unwanted radiation of the outside world. The anechoic chamber prevents spurious reflection of the signals transmitted in the laboratory. The RF generators must cover the range of modern tracker systems, and must provide for the other elements of the RF environment. GDC equipment can cover the frequency range from upper UHF to Kμ-band (0.5 to 18 GHz). The seeker, while usually supplied by the customer, must be interfaced to the lab.

All of its guidance signals must be converted to dc analog voltages for the computer simulation, and all radar signals except those coming from the target and the target environment must be exchanged with the proper lab devices. If a hardware (instead of a simulated) autopilot is used, such accelerometer and rate gyro signals as it requires must be provided in the proper form from the computer simulation. A simulated seeker is also usually mechanized on the computer for lab and computer checkout. With the seeker mounted on the three-axis flight table, the steering signals from the seeker hardware are used to drive the mathematical model simulated on the computer. The output of the mathematical model provides input signals to the translation and rotational drives. The target transport system positions the target in the Y and Z

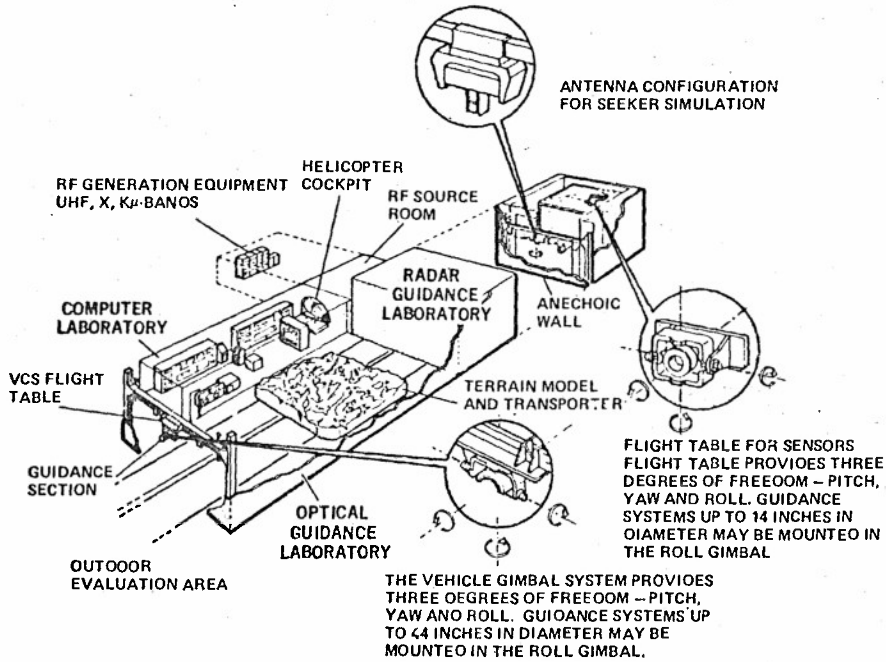



Figure 1. Missile Radar Guidance Laboratory

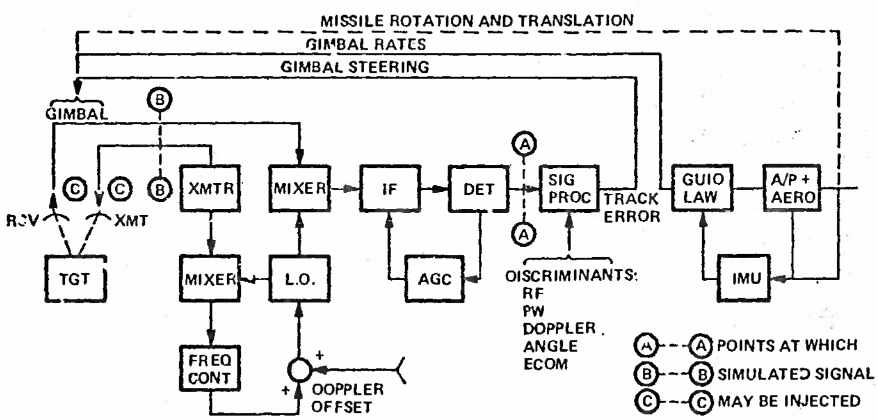



Figure 2. Point Tracking Sensor

positions to simulate the missile lateral and vertical translations. The sixth degree of freedom, longitudinal range, is electronically controlled as power, as a function of range.

A block diagram depicting the overall simulation of a missile in flight is shown in Figure 3. It is composed of computer performed simulation, translational and rotational degrees of freedom equipment, a target model, and a guidance seeker package. The GDC gimbal (3-axis flight table) provides the missile reference frame based on an inertial reference frame. All forces and moments on the airframe are calculated from the missile reference frame. Division by mass properties (inertial) then gives the accelerations in the same frame. Integration of these accelerations, then, gives the translations (u, v and w) and rotational (p, q and r) velocities in this body frame of reference. These velocities must then be transformed to an inertial frame of reference to be corrected for commanding the velocities of the three degrees of translation. Thus, by adding the three degrees of translation, the true dynamic spatial relationship of the missile relative to the target is obtained. By having this angular and rotational interaction, closed-loop simulation permits the computer representation of the aerodynamics, kinematics, autopilot, and actuators to experience the same dynamic environment that the seeker experiences under actual flight conditions. Even launch dynamics and wind buffeting effects can be simulated with realistic forces being applied to the seeker under test via the three-axis table and its translational capability. Figure 4 presents a sketch of the simulation configurations. The main elements are:

1 Anechoic chamber inside an electromagnetically shielded room;

2 Two-axis electromechanical transport system which provides lateral and vertical translation of the target antenna system;

3 Two-axis gimbal for rotation of the target antenna system;

4 Target antenna system which provides RF radiation;

5 Three-axis gimbal for real-time rotation of the system under test;

6 RF generation system which provides RF signals for radiation;

7 IR generation system for point IR sources;

8 Hybrid computer for computation in dynamic closed-loop, real-time simulation.

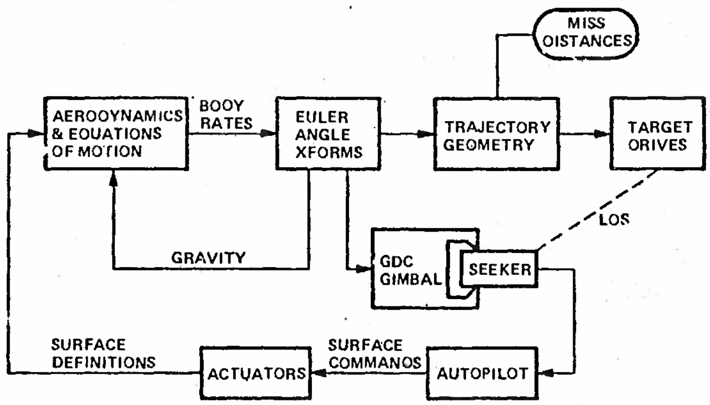



Figure 3. Missile Flight Simulation

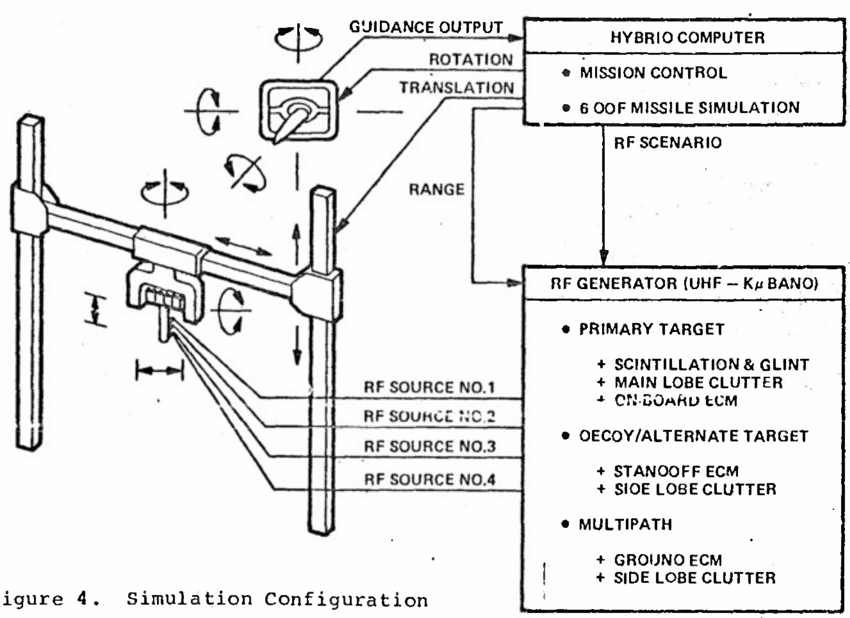



Figure 4. Simulation Configuration

## 2. Anechoic Chamber and Shielded Room

The anechoic chamber (Figure 5) is 25 feet high, 25 feet wide and 32 feet deep. The chamber was specified to provide anechoism from 0.5 to 18.0 GHz, although it will operate very well up to 50 GHz at a reflection level ranging from 30 to 45 dB on axis. The chamber is located inside an electromagnetically shielded room which is specified for insertion losses for plane waves greater than 80 dB from 0.5 to 50 GHz, and greater than 70 dB for electric fields from 200 kHz to 1 MHz.

### 3. Three-Axis Flight Table

The flight table located in the back of the anechoic chamber contains the guidance seeker and electronics. The flight table carries the seeker through three-degrees-of-freedom: pitch, yaw, and roll in that order of sequence. The flight table can accept a seeker package up to 14 inches in diameter weighing up to 100 pounds. Its main performance parameters are:

| | Pitch | Yaw | Roll |
|---|---|---|---|
| Displacement | ±120 deg about vertical | ±45 deg | 360 deg continuous |
| Velocity (max) | 200 deg | 200 deg | 750 deg |
| Accuracy | 0.05 deg | 0.05 deg | 0.05 deg |
| Bandwidth | 15 Hz | 15 Hz | 20 Hz |

### 4. Two-Axis Translational System

The translational system is located at the transmitting end of the anechoic chamber and consists of the horizontal beam and supporting columns as well as the lateral carriage. Trajectory motion of the missile in the lateral and vertical direction is provided by motion of the translational system carrying the transmitting linear array target antenna system that the seeker is tracking. The main characteristics of the translational system are:

| | Lateral | Vertical |
|---|---|---|
| Displacement | 22 feet | 22 feet |
| Velocity (max) | 4 feet | 6 feet |
| Accuracy | 0.25 inch | 0.25 inch |
| Bandwidth | 3 Hz | 3 Hz |

### 5. Two-Axis Gimbal

The two-axis gimbal assembly (Figure 6) is located on the lateral carriage. The main function of the gimbal is to point the linear array target antenna system toward the seeker contained in the three-axis flight table as the two-axis translational system moves about in the chamber. The separation of the centerline of the two-axis gimbal is 25 feet, with the lateral carriage centered at the transmitting end of the chamber. The gimbal is designed to position a 250 pound payload. The main characteristics of the two-axis gimbal are:

| | Yaw | Pitch |
|---|---|---|
| Displacement | ±25 deg | ±25 deg |
| Velocity (max) | 15 deg | 15 deg |
| Accuracy | 0.1 deg | 0.1 deg |
| Bandwidth | 10 Hz | 10 Hz |

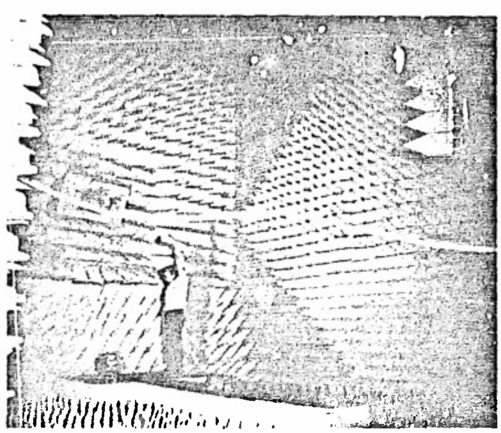


Figure 5. Anechoic Chamber

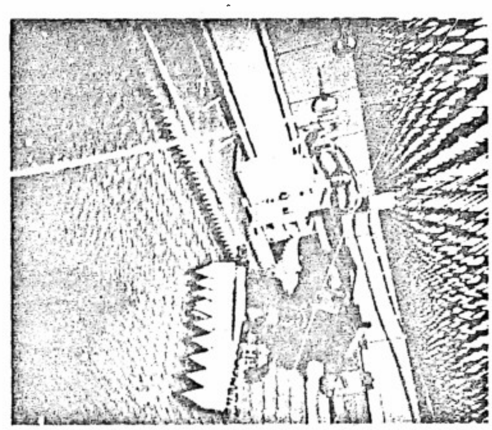


Figure 6. Two-Axis Transitonal and Gimbal Assembly

6. Linear Array Target Antenna System

The linear array target antenna system (Figure 7) consists of an electronic tee array of 13 log periodic antennas which can move three simultaneous, independent RF targets around the array. This array is mounted on the two-axis gimbal. A 16-bit Nova-2 minicomputer is dedicated to positioning these three targets on the array and is under control from the hybrid computer system.

7. RF Generation Equipment

The RF generation equipment (Figure 8) for point tracker guidance systems covers a range of 0.5 to 18.0 GHz. Two sets of RF generation equipment are used. One covers from 0.5 to 12.4 GHz while the other covers from 12.4 to 18.0 GHz. The RF generators are under hybrid computer control and form the RF pulses when commanded. The outputs of these RF generators are fed to the target motion system.

8. The IR Generation System

The IR generation system is located on the two-axis gimbal beneath the antenna array. The IR system is colocated with an RF antenna that can be operated independently of the antenna array. The RF antenna operates between 12.4 and 18.0 GHz, while the two IR sources can be operated between 3-5 and 8-14 microns. This system is designed for use for air-to-ground dual mode RF/IR guidance systems.

9. Hybrid Computer and Software

The hybrid computer system, which is dedicated to operating the GDC laboratories, (Figure 9) consists of Xerox Sigma-5 digital computers and several EAI 231-RV analog computers.

In general, the analog computers accept the guidance commands of the seeker and simulate the aerodynamics of the missile and target to determine the trajectory. The digital computer controls the timing and processing of the radar signals for the target and other elements of the environment.

In addition to the system and hybrid software, the software needed to generate the electromagnetic and natural environment is:

1. Target characteristics (Doppler, scintillation, glint, range delay, attenuation, pulsewidth change, emitter characteristics).

2. Environmental models (terrain backscatter, atmospheric effects, multipath).

3. RF run time library (attenuation, pulsewidth, pulse delay, frequency).

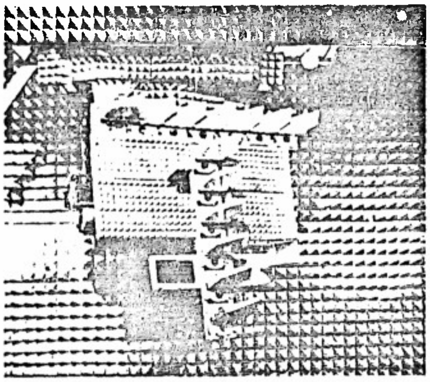


Figure 7. Linear Array Target Antenna System

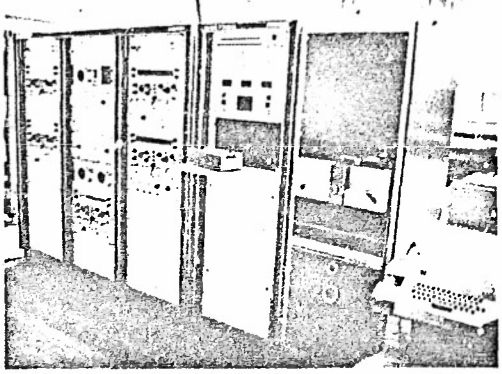


Figure 8. RF Generation Equipment

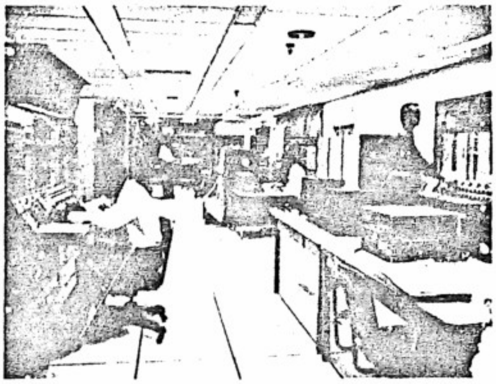


Figure 9. Hybrid Computer System

MAJOR CONSIDERATIONS

The major considerations in determining the simulation systems specifications were that the simulator provide a dynamic, real-time environment for a multi-mode guidance missile system which would allow extensive tests and evaluations to be performed. The following simulation system specifications were established:

- o Tracker types - point trackers, active and semi-active coherent and non-coherent, passive
- o Number of independent simultaneous targets - four
- o Total field of view : 45 degrees vertical and horizontal
- o Target angular rates - Up to 28 deg/s
- o Hand-over from one mode to another (active, passive)
- o Crossing targets
- o Types of signals - CW or pulsed
- o Operating frequency range - 0.5 to 18.0 GHz
- o RF power density at test aperture - 10 mW/m$^2$ (max) per target
- o Dynamic range - 100 dB
- o S/N ratio - 40 dB
- o Maximum PRF - 320 kHz
- o Frequency agility

Any frequency within a subband or a pulse-to-pulse basis Doppler shift capability is included for simulation of the radar types. It was felt that four independent simultaneous target types would be any form of decoys, main target, various ECM's clutter, multipath, glint, and scintillation. Each of these target or environment types would be a computer software model and would be called when needed. The number of targets could be considerably increased providing the PRF rate of the target is under 100 kHz. These targets then could be time shared on a channel, each with independent PRF rates and pulse widths.

Ideally, the total field of view would be greater than 100 degrees because of the large gimbal look angles of radar guidance systems. However, since this would be prohibitive in cost and very difficult to achieve mechanically, a 45 degree FOV is available in the GDC RF chamber. This is sufficient for most mission profiles.

The hand-over from one guidance mode to another is imperative for full mission system testing, to determine not only initial and final conditions, but to determine transients and boresight problems. Calculations showed that it would be optimum to achieve up to 140 mW/meter$^2$ on the aperture system under test for simulator close-in flight and a large power radiant target in a passive mode. However, in actual practice it is not practical to provide that much power because of large TWT requirements at the source. The frequency agility requirement is necessary to follow the frequency variation of the system under test.

PRINCIPAL NEW DESIGN FEATURES

The concept of an RF guidance simulation facility is not new; an RF guidance laboratory was a part of the GDC when it was built in 1966 and consisted of an open-ended anechoic chamber, three-axis flight table, two-axis translational system and a very limited S-band RF generation system. Computation facilities consisted of two analog computers. In 1972, the computational facilities were increased to the hybrid computer complex discussed earlier.

The expansion of the original RF guidance laboratory to the one that now exists incorporated new design features such as the target motion system, the RF generation system, and the RF/IR system.

1. Target Motion System

<u>Specifications</u>

The top level specifications for the target motion system are summarized as follows:

Operating Frequency - 0.5 GHz - 18 GHz

Number of simultaneous independent targets - four, including decoys standoff, and on-board ECM, clutter, and multipath

Additional return signal source capability - clutter, multipath, and ground band ECM

Multiple target angular separation - 0 to 9 degrees

Total field of view - 45 degrees

Relative target missile crossing angular rates

Velocity = 28 deg/s$^2$
Acceleration = 19.5 deg/s$^2$

Target positioning accuracy (including anechoic chamber RF and target motion system) = $\pm 3$ milliradians (rms)

Polarization - linear (vertical or horizontal)

Maximum test aperture diameter - 11 wavelengths

RF power density at test aperture 10 mW/m$^2$ (max) per target channel

Glint bandwidth - 10 Hz

Four basic radiating configurations were considered to meet these specifications, and the following conditions or criteria of design were assumed:

1 The test aperture diameter = $D_{12.4} \geq 10$ in at f = 12.4 GHz.

2 The aperture diameters that may be tested over the entire frequency region of $0.5 \leq f \leq 18$ GHz are subject to both the far-field condition, $R \geq 2.56\, D^2/\lambda$, and to the condition that the illumination across the test aperture not vary by more than $\varepsilon = 1$ dB (+0.5 dB) for all except approach 1. The latter condition requires that $R \geq 3.337\, d_e/\lambda\, D$.

3 The radiation path length $\equiv$ R may be 25 ft for approaches 2 and 3; the maximum chamber cross section is 20 x 20 ft. For the large fixed array (approach 4), $18 \leq R \leq 30$ ft; for this case, the maximum chamber cross sectional dimensions available for the array are L x L = 24 ft x 24 ft.

4 The array element spacing $\equiv d_e \geq 6$ in, based on estimated minimum element size required for low frequency ($\geq$ 0.5 GHz) operation.

The four approaches were:

1 Fixed single or multiple radiating elements.

2 Multiple fixed and electromechanically drive elements.

3 Fixed and electromechanically driven limited dimension matrix array.

4 Fixed large dimension matrix array.

Approach 1

This approach was based on locating fixed single or multiple radiating elements on a vertical wall at the rear of the anechoic chamber. As the simulation of glint and multipath was required and that the angle of arrival of the returns be controlled, approach 1 was eliminated.

Approach 2

This approach used three individual servo-driven elements as target radiators. The signals emitted are direct analogs of the actual RF signals that would be encountered in flight. Target-to-missile relative motion is provided by moving the target for trajectory simulation and moving the flight table to simulate missile short period motions. The radiators can be moved relative to one another to represent target return, glint, decoys, ECM, and multiple targets, or combinations thereof. The radiators are attached to the horizontal beam and are translated over the 20 ft x 20 ft extent of the anechoic chamber mouth.

Some particular limitations of this concept were:

1 The phase centers of the radiating elements cannot be colocated closer than 1.8 degrees.

2 Assuming angular glint with bandwidths up to 10 hertz as anticipated for targets with 100-ft cross range extent at slant ranges as small as 1000 feet to be simulated, the position rates and accelerations upon triad would become unmanageable.

Approach 3

The third approach was called the hybrid antenna system and uses two linear arrays of two-element subarrays. The linear arrays are each of length L = 4 ft and form a tee configuration. The entire tee array is moved electromechanically across the anechoic chamber mouth for large target translations. Vernier target motion and glint simulations are achieved by phase center scanning (PCS) within each two-element subarray (each representing a target source).

For R = 25, the far field condition allows $D_{12.4}$ = 10.6 inches. Then, the element spacing in each linear array will be $d_e$ = 0.767 $D_{12.4}$ = 8.10 in. For L = 4 ft = 48 in = (N-1)$d_e$, there are N = 7 elements per linear subarray or a total of $N_T$ = 2N = 14 elements for the linear tee array. The ratio D/λ = 1 at 12.4 GHz. To satisfy the test aperture illumination criterion of ±0.5 dB, the same D/λ at frequencies above 12.4 GHz must be maintained because $d_e$ = 0.767 $D_{12.4}$ is fixed.

Three simultaneous targets were chosen to be simulated by this approach. Each two-element subarray can handle all three targets and a crossing target capability exists. As each simulated target is translated across each subarray, the nearest adjacent subarray element is switched in to form a new subarray, thereby allowing a smooth hand-over and translation of the target centroid across each arm of the tee.

The linear tee array is a special and simplified form of matrix array (array of subarrays) considered in approach.

In order to maintain an attractive cost, the number of antennas was limited to 13 with options to add up to 120 more radiating elements at a later date.

Approach 4

This approach employs a large fixed matrix array which would be located on a wall at the rear of the chamber. Configuration of this approach contained from 500 to 900 radiating elements. It consisted of radiating elements placed on an equilateral triangular grid, forming three-element subarrays or triads. This system also met desired performance criteria; however, it was eliminated because of cost and anticipated technical problems associated with the $SPN_ST$ switch where $N_S$ = $N_T/3$ = 34; the problem here lies in its practical implementation for reasonably low insertion loss, high average power handling, and phase equality between outputs ports, for the entire 0.5 to 18.0 GHz region. Further, this approach does not allow a combination RF/IR capability as no mechanical target motion is involved.

Selected Approach

Approach 3 was selected. The block diagram is shown in Figure 10. The antennas chosen for meeting the broad frequency requirement of 0.5 to 18.0 GHz low mass and ease of fabrication were log-periodic antennas. The base dimension is 8 inches and the length to apex is 13 inches. Ordinarily, the 8-inch base dimension would render these antennas useful down to 1 GHz (0.5λ). By loading the elements of the array, these antennas can be made to operate down to 0.5 GHz. A gain of 7 dB and a half-power beamwidth of 70 degrees is characteristic of these structures.

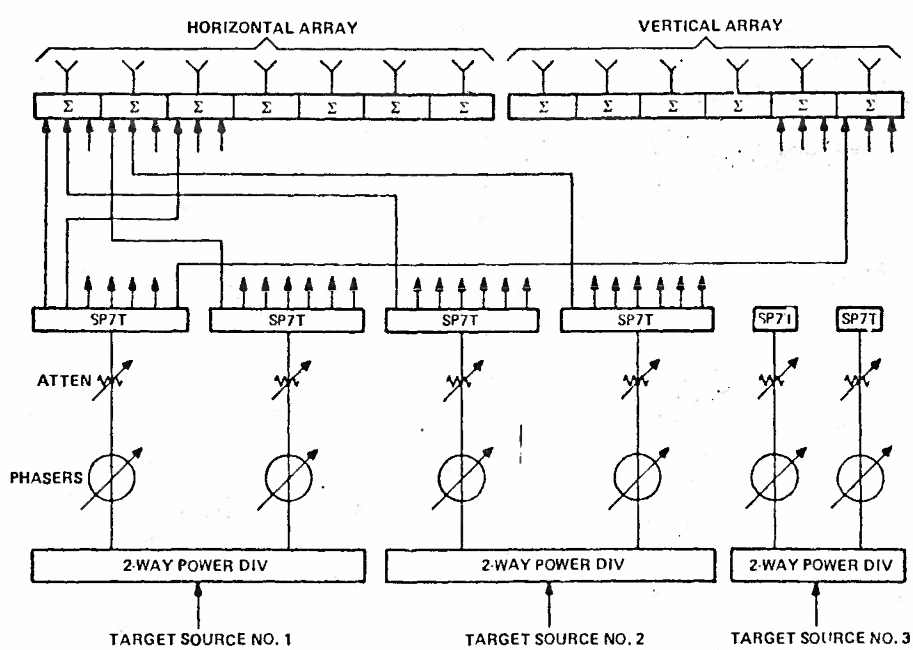



Figure 10. Tee Matrix Array

2. RF Generator

Specifications

The top specifications for RF generation are summarized as follows:

Non-Coherent Point Tracker Channel Specifications

| | |
|---|---|
| Frequency | 1.0 - 12.4 GHz |
| Instantaneous frequency diversity | 400 MHz |
| Dynamic range | 100 dB minimum |
| Amplitude resolution | 0.2 dB minimum |
| Amplitude accuracy | ±2 dB maximum |
| Types of signals | CW and pulses |
| Pulsewidth (pulsed mode) | 50 ns to 80 μs |
| Lower power (input to RF generator) | -20 dBm minimum (internal coupler provided for coherent ECM) |
| S/N (broadband) | 40 dB minimum |
| Pulsewidth resolution | 50 ns |
| Range delay | 1 ms maximum |
| Range resolution | 50 ns |
| Pulse repetition rate | 100 Hz - 100 kHz |
| Range rate | 13,000 ns/s maximum |
| Range rate resolution | 50 ns/s |

FACILITY COST

To determine the value of laboratory ownership, both initial acquisition and operating costs must be considered. Table II is a summary of the initial acquisition cost for a complete laboratory. If it is desired to build more than one laboratory - for example, radar, electro-optical, infrared - a corresponding increase in test equipment and a smaller increase in the other factors must be included. When considering recurring costs, it is evident that the most significant items are those associated with labor. The costs shown in Table III are representative.

TABLE II

Initial Acquisition Costs

| Item | 1979 $ (millions) |
|---|---|
| Building | 4 |
| Computers | 5 |
| Test Equipment | 11 |
| Controls | 2 |
| Total | 22 |

TABLE III

Operating Costs/Shift/Year

| Skill | | Number | 1979 $ (thousands) |
|---|---|---|---|
| | Programming | 3 | 150 |
| Labor | Operating | 2 | 80 |
| | Maintenance | 2 | 80 |
| | Supervision | 1 | 60 |
| | Subtotal | | 370 |
| Materials | 1% of Hardware | | 90 |
| | Total | | 460 |

A previous lecture has shown the powerful economic and political justification for simulation; however, the global savings may not be important to the immediate program manager. If a facility is to be used extensively by projects, its weekly rental charges must be low. To establish a reasonable return on investment, the depreciation, operating costs and profit must be established based upon expected operation of the facility. Figure 11 illustrates the reduction in costs per week, as additional shifts are worked in the same facility. To increase operations substantially beyond two shifts requires [illegible] economic aspects, most new simulation facilities include at least two Simulation Laboratories such as shown in Figure 1. The multiple laboratorics also have the advantage of stabilizing work loads and allowing for income while modifications and changes are being made.

With consideration of all the above aspects, it is apparent that this type of facility will cost $2,000,000 per year. Few countries or companies could afford such a laboratory for their individual requirements unless there are multiple weapon developments being pursued. The laboratory at Martin Marietta is currently running slightly below four shifts and rents laboratory time to the United States Government (Army, Navy and Air Force), foreign governments, and as a third priority to other commercial firms such as General Dynamics, which has conducted an F-16 weapons program there.

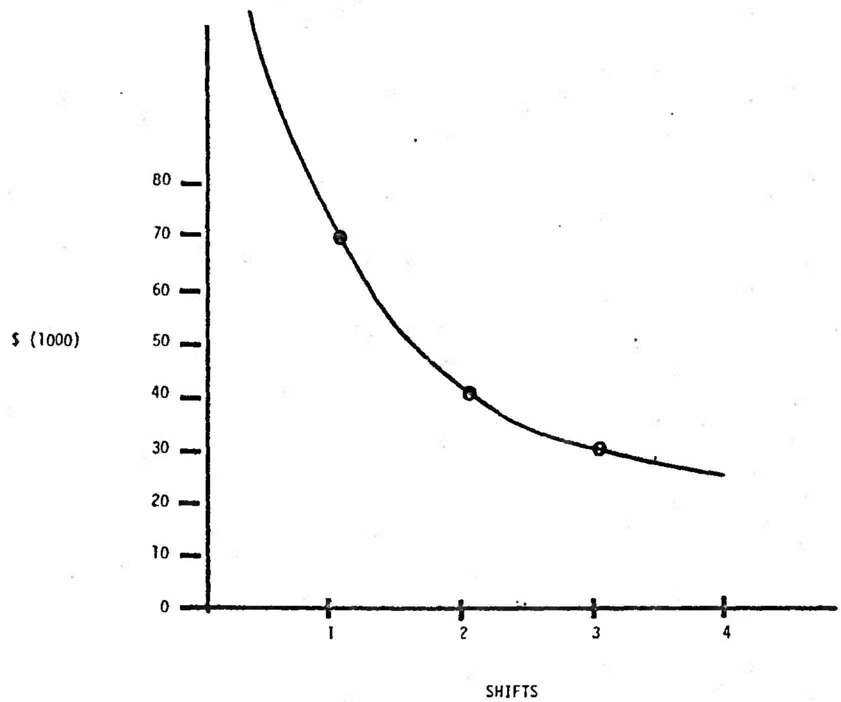


Figure 11. Cost Per Week as a Function of Shift

ELECTRO-OPTIC IMPROVEMENTS

In an earlier lecture ("Laboratory Technique and Evaluation Methodology," Series #52), the operation of an electro-optic facility to design and evaluate weapons was described. During the intervening time, technology has progressed to the point that such systems are becoming obsolete. These weapons are constrained by the operator's acquisition of the target through the weapon optics. This restricted the operator to limited ranges and a relative small terrain area in front of the vehicle flight path. Now that it has been conclusively demonstrated that tracking systems can hit the targets they are locked on to, the weapon problem becomes one of finding the target at greater ranges. It was necessary to modify the laboratory to permit evaluation of the more complex task. To illustrate the changes required, let us examine a mission scenario.

If a helicopter flying through its tactical mission is considered (Figure 12), the flight gunner or pilot must perform a target search and detection through the windscreen with the unaided eye. Then, through the use of the optical sensor (the aided eye) he acquires the target through the fire control display, and the recognition or identification and tracking task is initiated. In some cases, the aircraft is maneuvered into position through the boresight of the aircraft, or in others the fire control gimbals are torqued to the line of sight. The weapons or turret alignment is thus effected, and the task may consist of transmitting target coordinate and code information. However, if the helicopter is to deliver weapons, the missile launch sequence is initiated, and the weapon is fired and left to track the target automatically to the point of impact.

As can be seen, there is heavy interplay between the man and the machine. Without this interplay in the early development tests, the total effectiveness of a system is not known until after a rigorous flight test program has been performed or feedback from subsequent field operations has been received.

It is also worthy of note that in the real world, scoring is accomplished in terms of the end result which is singular in nature - "Was the selected target hit?" This is not so in the simulated world where check points can be used to determine where the greatest man/machine errors exist.

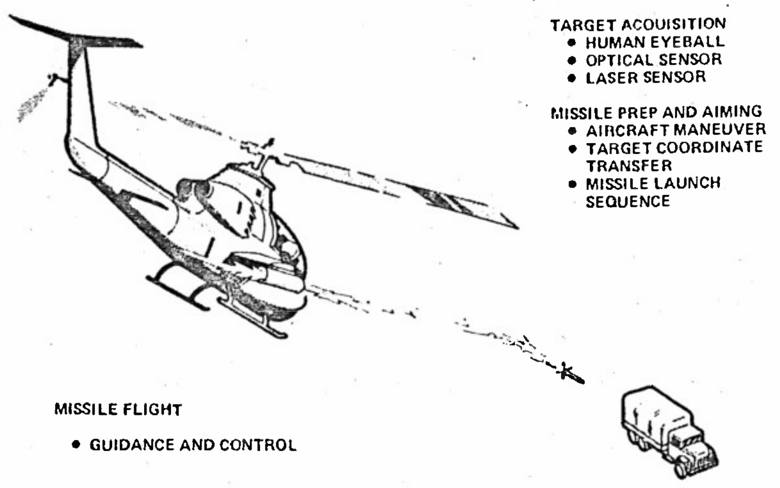



Figure 12. Helicopter System Mission

The major subsystems to fully accomplish the simulation are the crew station, the visual scene displays, the motion base, the large payload gimbal and the computer software program (see Figure 13).

The terrain model is 80 by 40 feet with a large selection of optical targets. It can be scaled selectively from 1:200 to 1:1200 thus permitting extended target search or terrain masking experiments. The lighting is variable between 200 and 2000 foot candles with color mixing (Figure 14).

The vehicle gimbal system (VGS) (Figure 15), is a versatile gimbal system that accepts a large payload and is interchangeable with the present flight table provided in the optical chamber. The gimbals are servo power driven. The gimbal order of yaw, pitch, and roll is conducive to aircraft maneuvers such as those required for pylon turns.

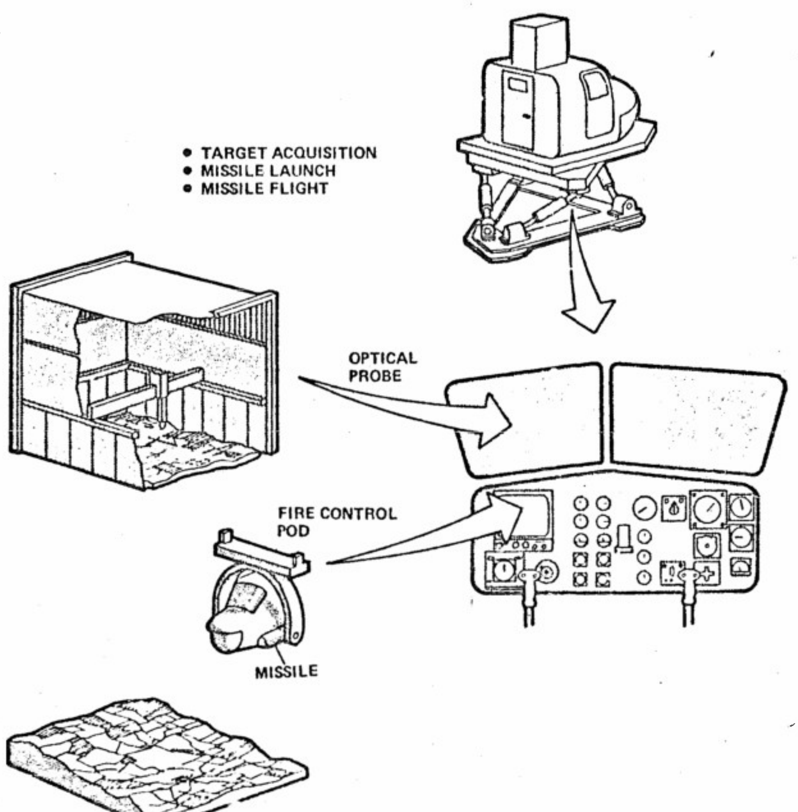



Figure 13. Simulation Elements

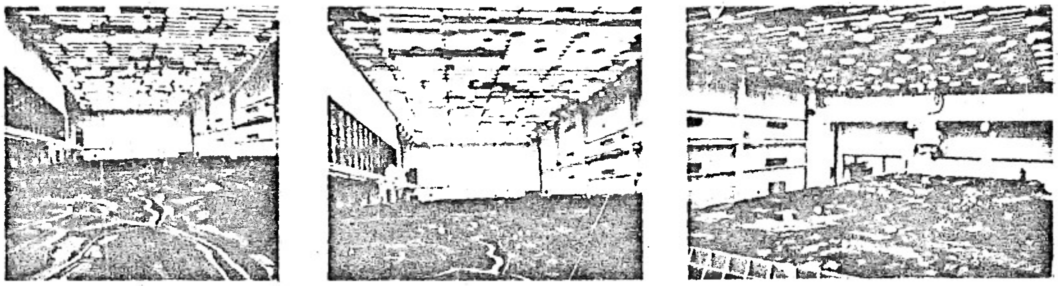


Figure 14. Terrain Model

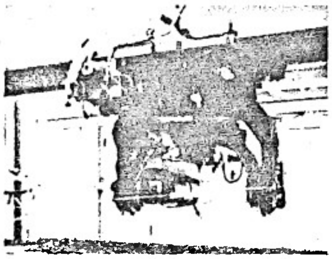


Figure 15. Vehicle Gimbal System

The optical probes move on a two axis transport over the terrain model to simulate six degrees of freedom and provide the pilot with both an out the window scene and a weapon system sensor display (Figure 16).

The crew environment is the cockpit area of a helicopter or aircraft. Actual fuselage sections provide a realistic environment for the pilot and gunners to perform their tasks on a six-degree-of-freedom motion base (Figure 17). The motion base is designed for computer control. Six hydraulic actuators produce the degrees of freedom with a performance that will satisfy motion cues for helicopter and aircraft flight simulation.

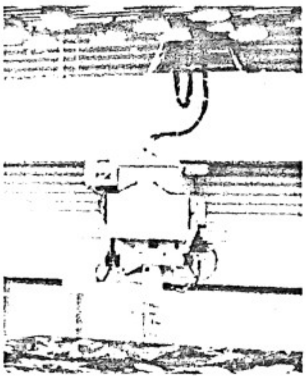
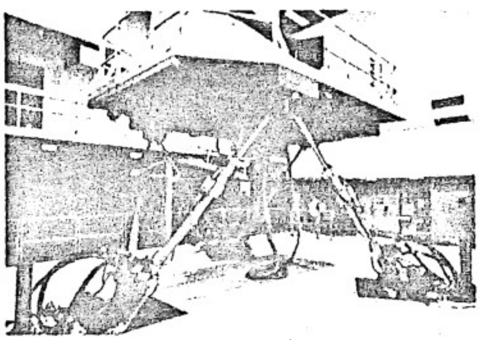


Figure 16. Optical Probes and Gantry

Figure 17. Motion Base



One of several aircraft, ship, or tank cockpits (Figure 18) can be mounted on this motion base to provide a most realistic operator workload. The hybrid computers control these simulation elements in a variety of combinations to achieve specific mission objectives.

Weapon systems, fire control systems, equipment integration, pilot workload, avionics configurations, sensor/display combinations, and other concepts are evaluated for total system acceptability. These factors are rated for advantages and disadvantages from an operability standpoint, thus ensuring optimum system concepts before baseline hardware is designed.

CONCLUSION

Complex simulations of missile guidance and control systems are now within the state-of-the-art. Such simulations hold the potential for reducing costs and schedules on development programs. They can also increase system performance by allowing systematic evaluation of variables. However, since weapon development can quickly make technology obsolete, flexibility of the simulation facility's configuration and ingenuity in its design are of paramount importance if the facility is to be cost-effective.

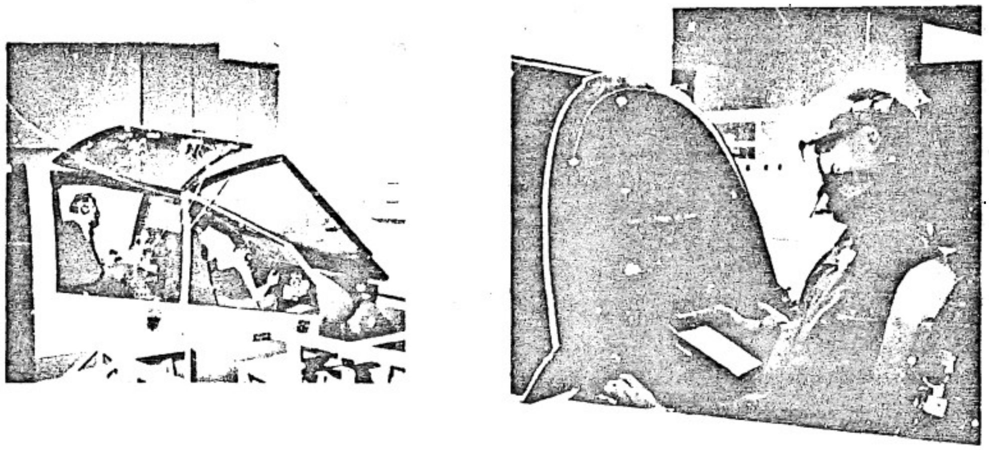


Figure 18. Helicopter and Aircraft Cockpits

# BIBLIOGRAPHY

## Guidance and Control for Tactical Guided Weapons

This Bibliography with Abstracts has been prepared to support AGARD Lecture Series No. 101 by the Scientific and Technical Information Branch of the US National Aeronautics and Space Administration, Washington, D.C., in consultation with the Lecture Series director, Mr. C.T.Maney.

# SECTION 1

## MIDCOURSE GUIDANCE AND CONTROL AND GENERAL NAVIGATION

Application of an extended Kalman filter to an advanced fire control system
A/MAYBECK, P. S.; B/LUTTER, R. N. A/(USAF, Institute of Technology, Wright-Patterson AFB, Ohio); B/(USAF, Avionics Laboratory, Wright-Patterson AFB, Ohio) In: Conference on Decision and Control, and Symposium on Adaptive Processes, 16th, and Special Symposium on Fuzzy Set Theory and Applications, New Orleans, La., December 7-9, 1977, Proceedings. Volume 1. (A79-14957 04-63) Piscataway, N.J., Institute of Electrical and Electronics Engineers, Inc., 1977, p. 1192-1195.
ABS:An extended Kalman filter is developed to aid the tracking of an air-to-air missile from a maneuvering target aircraft. The filter exploits knowledge of the dominant aerodynamically induced lift and drag forces of a non-thrusting missile employing proportional navigation guidance. The filter provides both dynamic tracking estimates in a local inertial frame and estimates of pertinent parameters including the proportional navigation constant. Initial research has established the feasibility of this modeling approach to tracking filter development, and current efforts are fully exploring its performance capabilities. The objective is a filter that will provide both accurate, robust tracking estimates and meaningful threat predictive capabilities 77/00/00 79A15018

Missile guidance for low altitude air defense
A/NESLINE, F. W. A/(Raytheon Co., Missile Systems Div., Bedford, Mass.) In: Guidance and Control Conference, Palo Alto, Calif., August 7-9, 1978, Technical Papers. (A78-50159 22-01) New York, American Institute of Aeronautics and Astronautics, Inc., 1978, p. 516-523.
ABS:Practical design considerations for a low altitude radar guided air defense missile are presented. Low altitude target signals return to the receiver mixed with large clutter signals from ground and large multipath signals from smooth sea. The doppler effect and the Brewster angle effect are used to separate true target returns from clutter and multipath contaminations. Sensor design factors, including clutter and multipath rejection, doppler resolution, and sensor stabilization are discussed in the context of a complete missile guidance and control system. Basic contributors to miss distance are discussed with a quantitative miss distance example showing the role of missile lateral acceleration capability.
AIAA 78-1317 78/00/00 78A50216

Strapdown seeker guidance for tactical weapons
A/CALLEN, T. R.; B/EHRICH, R. O. A/(USAF, Armament Laboratory, Eglin AFB, Fla.); B/(Rockwell International Corp., Columbus, Ohio) In: NAECON '78; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 16-18, 1978. Volume 2. (A78-49851 22-04) New York, Institute of Electrical and Electronics Engineers, Inc., 1978, p. 697-704.
ABS:A description is presented of the work accomplished thus far in connection with a study which is to determine the best combination of guidance law structure, signal processing techniques, and achievable seeker and sensor accuracy requirements for the effective use of strapdown seekers with tactical guided weapons. An evaluation is provided of proportional navigation utilized as the guidance scheme for air-to-surface weapons with strapdown seekers. Attention is given to mathematical models of air-to-surface weapons, methods of generating line-of-sight rates, seeker models and error sources, and simulation results. The results of the study thus far indicate, as expected, that proportional navigation with reasonable navigation gain is sufficiently sensitive to seeker and scale factor errors to make implementation extremely difficult. 78/00/00 78A49928

Navigation computation in terrestrial strapdown inertial navigation systems
A/BAR-ITZHACK, I. Y. A/(Technion - Israel Institute of Technology, Haifa, Israel) IEEE Transactions on Aerospace and Electronic Systems, vol. AES-13, Nov. 1977, p. 679-689.
ABS:Various computational schemes for computing the translational velocity and position relative to earth, a computation which has to be performed by the processor of a strapdown inertial navigation system, are compared. A split-coordinate approach was selected in which the differential equations are solved at different computational rates. The differential equations of this scheme are developed and the assumptions on which the development is founded are stated. It is shown that the split-coordinate approach can be used only if the gravitational acceleration is assumed to be piecewise constant and is updated at the slowest computational rate. Then the intermediate-rate computation has to be carried out in an earth-fixed coordinate system rather than in a local-level, local-north system.
77/11/00 78A23413

A low cost GPS navigation set for tactical weapons
A/SCHMITT, A. F.; B/DEPRIEST, C. D. A/(Teledyne Systems Co., Northridge, Calif.); B/(USAF, Armament Laboratory, Eglin AFB, Fla.) Institute of Navigation, Annual Meeting, 33rd, Costa Mesa, Calif., June 22-24, 1977, Paper. 14 p.
ABS:The Air Force has begun competitive development of a Global Positioning System (GPS) weapon guidance system for glide bombs and missiles. The primary objective of the considered program is to develop and demonstrate a GPS weapon midcourse guidance system capable of reliable handover to any one of several terminal systems. A GPS receiver navigates by making direct ranging measurements from its position to four of eight transmitting satellites in view at any time. The satellites' precise positions are furnished by data contained in a navigation message superposed upon the transmission of each satellite. A receiver of simplified design, designated the GPS Class M Receiver, is to function in a mother-daughter approach together with the launch aircraft (the mother). 77/06/00 78A17910

Tactical guidance requirements for strapdown inertial
A/MUELLER, C. E.; B/PHELPS, R. K.; C/SCHEIDENHELM, R. A/(Honeywell Systems and Research Center, Minneapolis, Minn.); B/(Honeywell, Inc., Defense Systems Div., Minneapolis, Minn.); C/(Honeywell, Inc., Avionics Div., St. Petersburg, Fla.) In: NAECON '77; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 17-19, 1977. (A78-15551 04-33) New York, Institute of Electrical and Electronics Engineers, Inc., 1977, p. 433-440. USAF-supported research.
ABS:An analysis of tactical inertial performance requirements for three strapdown inertial guidance system mechanizations - pure inertial, RAC aided inertial, and GPS aided inertial - is described. Cost-optimal performance requirements are determined for a family of powered and unpowered guided conventional weapons. Stochastic sensor error modeling, velocity-matching transfer alignment, and optimal and suboptimal Kalman filtering are also discussed. 77/00/00 78A15604

Tactical Global Positioning System Guidance
A/DEPRIEST, C. D. A/(USAF, Armament Laboratory, Eglin AFB, Fla.) In: NAECON '77; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 17-19, 1977. (A78-15551 04-33) New York, Institute of Electrical and Electronics Engineers, Inc., 1977, p. 418-423.
ABS:The paper reviews the status of the Tactical Global Positioning System Guidance Program, which was organized to develop and test a demonstration GPS guidance system at the Air Force Armament Laboratory. The system concept incorporates a low cost inertial guidance subsystem with a GPS 'Class M' missile receiver. Attention is given to the program philosophy which encompasses concurrent development of system configuration and Class M Receiver by two independent contractors, and which will ultimately lead to free flight demonstrations of tactical GPS midcourse guidance. 77/00/00 78A15603

Low cost inertial guidance with GPS update for tactical weapons
A/COX, J. W. A/(USAF, Armament Laboratory, Eglin AFB, Fla.) In: NAECON '76; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 18-20, 1976. (A77-37352 17-33) New York, Institute of Electrical and Electronics Engineers, Inc., 1976, p. 623-626.
ABS:The Air Force Armament Laboratory has recently initiated new efforts to develop a Low Cost Inertial Guidance Subsystem (LCIGS) for tactical guided weapons. The efforts are to: (1) define the inertial strapdown subsystem requirements, (2) design, develop and test an engineering model LCIGS, and (3) develop new low cost inertial instruments for long range improvement of performance and reduction cost. The result will be a design data package suitable for a competitive, non-proprietary procurement of inertial strapdown subsystems. LCIGS will be optimally designed to receive position and/or velocity updates from alternative navigation schemes such as Navstar Global Positioning System. A digital processor will be designed to handle system integration for a modular family of tactical missile configurations. 76/00/00 77A37432

A guidance concept for air-to-air missiles based on nonlinear differential game theory
A/POULTER, R. A.; B/ANDERSON, G. M. B/(USAF, Institute of Technology, Wright-Patterson AFB, Ohio) In: NAECON '76; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 18-20, 1976. (A77-37352 17-33) New York, Institute of Electrical and Electronics Engineers, Inc., 1976, p. 605-609.
ABS:This paper discusses a new guidance concept for air-to-air missiles based on differential game theory using realistic nonlinear system dynamics and kill probability as the payoff. Current missiles use proportional navigation, a feedback guidance method that is an optimal differential game pursuit strategy if the intercept problem is modeled as a simple linear system with quadratic cost. With this more realistic problem formulation, proportional navigation is no longer optimal. Computer simulations of the intercept problem comparing proportional navigation to this new guidance concept are presented for maneuvering targets.

Considerable increases in kill probability are achieved with this new concept thereby providing the incentive to develop a real-time version. 76/00/00 77A37429

Morphological thinking on the synthesis of a beam-riding missile system

A/MAHAPATRA, P. R. A/(Indian Institute of Science, Bangalore, India) Institution of Electronics and Telecommunication Engineers, Journal, vol. 22, Oct. 1976, p. 689-693.

ABS:In this paper, the concept of morphological approach to a general system design problem, first evolved by Zwicky, has been applied to the case of a beam-riding missile system. In a detailed study of the morphological box for the system, such sub-system combinations as are feasible but not conventionally employed are brought under close scrutiny to yield dimensions in which conventional beam-rider system engineering can expand to include the technology of today. The particular case of the combination of various types of terminal guidance system with the principal beam-riding guidance system has been studied in some depth and detail. 76/10/00 77A28068

Covariance error analysis of a missile trajectory in an atmospheric flight

A/BAR-ITZHACK, I. Y.; B/BAR-GILL, A. A/(Technion - Israel Institute of Technology, Haifa, Israel) (Israel Annual Conference on Aviation and Astronautics, 18th, Tel Aviv and Haifa, Israel, May 19, 20, 1976.) Israel Journal of Technology, vol. 14, no. 1-2, 1976, p. 37-46.

ABS:This paper presents an analysis of position and orientation errors during the atmospheric flight of a missile. The autopilot employs body angle guidance where true missile orientation is measured by a directional gyro and vertical gyro. The analysis applies the covariance propagation technique to the error state vector. The mathematical model is a six-degree-of-freedom model: angular time constants of the autopiloted missile are negligible. The linear state equation of the error vector is obtained by a piecewise linearization of the nonlinear airframe model about the reference trajectory. This analysis is applied to an assumed model of a missile similar to the U.S. Navy Condor. It is concluded that the error sources, which are the major contributors to the final error, are the wind, the error in the determination of the zero lift drag coefficient, the thrust deviation, the deviation in the atmospheric conditions and the gyro initial misalignments and their drifts. 76/00/00 77A15031

Laser inertial platform for Army missiles

A/JOHNSTON, J. V.; B/PUGH, R. E. B/(U.S. Army, Missile Command, Redstone Arsenal, Ala.) In: Engineering in a changing economy; Proceedings of the Southeast Region 3 Conference, Clemson, S.C., April 5-7, 1976. (A76-47201 24-99) New York, Institute of Electrical and Electronics Engineers, Inc., 1976, p. 221-223.

ABS:The system aspect of a laser gyro strapdown inertial measurement unit applicable to US Army missiles is examined. The sensor assembly contains a single triad that mounts the three ring laser gyros and three accelerometers. Attention is given to the unique conditions for application of laser gyro IMUs to Army missiles, the general parameters of strapdown IMUs, strapdown navigation software, operating modes, and the performance evaluation system. 76/00/00 76A47227

Guidance of ballistic flight vehicles

A/MOGILEVSKII, V. D. Moscow, Izdatel'stvo Mashinostroenie, 1976, 208 p. In Russian.

ABS:The book gives a systematic exposition of the fundamental questions of the theory of guidance of flight vehicles with ballistic flight segment, at the same time giving attention to the design of guidance systems and the construction of basic guidance algorithms. After posing the general problem of guidance and giving a formal description of ballistic motion, the book examines the problem of determining the optimal program for control of injection, giving attention to single-parameter and multiparameter control laws. Synthesis of an optical control law for satellite rendezvous is also studied along with the problem of autonomous navigation and simplified methods for controlling engine cutoff. 76/00/00 76A43424

A missile laser gyro rate sensor

A/MORRISON, R. F.; B/STRANG, C. B. A/(Sperry Rand Corp., Sperry Gyroscope, Great Neck, N.Y.); B/(Martin Marietta Aerospace, Orlando, Fla.) In: Guidance and Control Conference, San Diego, Calif., August 16-18, 1976, Proceedings. Conference sponsored by the American Institute of Aeronautics and Astronautics. New York, American Institute of Aeronautics and Astronautics, Inc., 1976. 6 p. Research supported by Martin Marietta Aerospace.

ABS:The paper describes the Sperry SLIC-7 laser gyro, designed to satisfy severe dynamic and environmental requirements of an advanced interceptor missile. The laser gyro has the following design features: (1) a low expansion Cer-Vit material for perimeter stability; (2) all-mirror, multilayer dielectric corner reflectors; (3) a helium-neon gas discharge tube; (4) a lock-in avoidance mechanism; and (5) a configuration which provides a unique arrangement of three axes of laser gyros integrated into a common

structure. The laser gyro electronics assembly - signal processor, laser gyro control, and power supply - is described, and preliminary test results are presented.
AIAA 76-1967 76/00/00 76A41496

Command to line-of-sight guidance - A stochastic optimal control problem
A/KAIN, J. E.; B/YOST, D. J. B/(Johns Hopkins University, Laurel, Md.) In: Guidance and Control Conference, San Diego, Calif., August 16-18, 1976, Proceedings. (A76-41426 20-12) New York, American Institute of Aeronautics and Astronautics, Inc., 1976, p. 356-364.
ABS:A command to line-of-sight (CLOS) guidance design approach using modern stochastic optimal control theory is discussed. CLOS guidance requires a wide guidance bandwidth in order to follow a threat maneuver. Yet the LOS noise (beam jitter) inherent in any LOS tracking scheme must be attenuated in order to prevent excessive control surface saturation. The stochastic describing function (CADET) is used to model the aerodynamic control surface saturation nonlinearity allowing the 'linear' stochastic optimal control theory to be applied. Results from a sample airframe indicate near optimal performance using a realizable nonlinear guidance compensation against a randomly maneuvering threat.
AIAA 76-1956 76/00/00 76A41467

Large angle-of-attack missile control concepts for aerodynamically controlled missiles
A/ARROW, A.; B/YOST, D. J. B/(Johns Hopkins University, Laurel, Md.) In: Guidance and Control Conference, San Diego, Calif., August 16-18, 1976, Proceedings. (A76-41426 20-12) New York, American Institute of Aeronautics and Astronautics, Inc., 1976, p. 247-254.
ABS:A coupled or cross axis autopilot control concept is presented which can provide a significant increase in the usable range of angle-of-attack for aerodynamically controlled tactical missiles without an attendent increase in either steering response time or roll channel bandwidth. The unique feature of the coupled control system design is that adequate airframe stability can be maintained in the presence of aerodynamic cross coupling at large angles-of-attack without either physically rolling the airframe to a preferred orientation with respect to the airstream or increasing the bandwidth of the roll system. Stability is maintained by intentionally cross coupling sensor signals among the control channels (i.e., roll sensor signals into steering control and vice versa) at large angles-of-attack. In effect, the intentional cross channel coupling partially cancels the destabilizing effect of the interchannel aerodynamic coupling.
AIAA 76-1944 76/00/00 76A41455

Radar guidance of missiles
A/IVANOV, A. A/(Raytheon Co., Bedford, Mass.)
In: International Radar Conference, Arlington, Va., April 21-23, 1975 Record. (A75-45182 23-32) New York, Institute of Electrical and Electronics Engineers, Inc., 1975, p. 331-335.
ABS:Guided missile development dates from the latter days of World War II. Of the many types of guidance used, the most successful has been semi-active radar homing. This paper highlights the CW semi-active approach and presents typical implementations of a missile-borne seeker. Extension of these systems to other than CW waveforms and ground radar considerations are also discussed. 75/00/00 75A45230

Fluidics in missile control systems
A/GRIFFITH, W. A. A/(U.S. Army, Missile Research, Development and Engineering Laboratory) Fluidics Quarterly, vol. 6, July 1974, p. 31-42.
ABS:Fluidics components and systems have been used in experimental missile control systems for the past decade. The advantages of fluidic systems in comparison to conventional devices are evaluated. Fluidic systems have fewer moving mechanical parts, thereby reducing wear and contamination, and increasing reliability and storage life. Attention is given to design considerations, taking into account a number of problems which have been satisfactorily solved. Fabrication techniques for use in the research and development of fluidic missile control systems must be versatile and responsive. Some of the techniques currently used for missile fluidics fabrication are machining, etching, and diffusion bonding. 74/07/00 75A14338

SLIC-7 laser gyro investigations
A/ABOALE, J.; B/DREXEL, W. Sperry Rand Corp., Great Neck, N. Y.
ABS:This program is the initial phase of an effort to develop laser gyro technology for application to low cost inertial missile guidance systems. As part of the research and development effort two investigations of SLIC-7 laser gyro were conducted: (1) thermal analysis and (2) storage and shelf life evaluation. The key to quick reaction and performance stability is the design to minimize thermal gradients and, therefore, thermal sensitivities. Test results disclose that the thermal gradients across the discharge tubes are the key factors. Analyses indicate that the SLIC-7 gyro can meet the low thermal sensitivity goals and the storage and shelf life requirements postulated for the missile guidance systems.
AD-A059028 SG-4240-1027 AFAL-TR-78-32 78/03/00 79N13021

Laying and a ·ing study for advanced land combat systems, part 2

A/JOHNSTON, J. V. Army Missile Research and Development Command, Redstone Arsenal, Ala. (Technology Lab.)

ABS: New developments in inertial equipment have led to new guidance mechanizations. Computer technology in the past ten years has made outstanding advances in size speed, and lower cost. These advances have brought to the forefront the implementation of strapdown guidance. The purpose of this report is to update a previous study to include strapdown mechanizations.

AD-A054127 DRDMI-T-78-43-PT-2 78/02/22 78N30076

Laser-gyro strapdown inertial system applications

A/LEVINSON, E. Sperry Rand Corp., Great Neck, N. Y. (Gyroscope Div.) In AGARD Strap-Down Inertial Systems 48 p (SEE N78-26124 17-04)

ABS: The following laser gyro strapdown inertial systems are described: (1) tactical air to surface missile midcourse guidance; (2) shipboard fire control: attitude reference; and (3) aircraft inertial navigation. Mission requirements system configuration, alignment techniques, and existing hardware and software are delineated for each application. Error analysis simulation and test data are presented which demonstrate the capacity of the laser gyro system to meet the specific application requirements.

78/05/00 78N26130

Modular digital missile guidance, phase 3

A/LANGLEY, F. J. Raytheon Co., Bedford, Mass. (Missile Systems Div.)

ABS: This report presents the results of the third phase of a study to investigate the feasibility of modular digital guidance and control systems for air-to-air missile applications The studies involved the analysis of functions for digital implementation in all classes of air-to-air missiles and the derivation of computer requirements in terms of throughput memory, architectural features, modularity and compatible software characteristics. Phase III validated the performance and effectiveness of the macromodular microcomputer family defined in Phase II, on an individual module basis; as whole microcomputers; and as federated microcomputer systems applied to specific generic missile types, using digital simulation techniques. In summary, the studies have shown that modular digital guidance and control is both feasible and effective in improving missile performance and flexibility to counteract changing threat situations and advancing technology. Using a common microcomputer bus interface. (microbus), a family of fourteen microcomputer macromodules, in various configurations, will support the entire range of air-to-air missile functions. Further, the Navy standard electronic module (SEM), in either SEM-1A or SEM-2A configurations, provides a practical means of packaging the macromodules and maintaining the standard microbus interface.

AD-A042466 BR-9448 77/05/04 77N32210

A Kalman filter application to the advanced tactical inertial guidance system of the air-launched low volume ramjet cruise missile

A/VANOEVENDER, J. A. Naval Postgraduate School, Monterey, Calif.

ABS: A Montecarlo simulation is conducted to ascertain performance of the ATIGS system which is a proposed air-launched cruise missile configuration. The simulation is conducted within a local-level inertial frame consisting of down-range, cross-range, and up as primary reference vectors. Efforts are made to measure the relative effects associated with the intended pure position reset provided by a MICRAD sensor as compared with those effects which could be expected from a linear suboptimal Kalman filtering scheme used in conjunction with the MICRAD sensor.

AD-A039338 76/12/00 77N30156

DME surface to surface missile demonstration system analysis IBM Federal Systems Div., Owego, N. Y.

ABS: System design, analysis and tests were performed in preparation for a joint U.S. Army/Air Force demonstration flight test. This flight test will involve use of the Air Force Advanced Location and Strikes System (ALSS) Distance Measuring Equipment (DME) navigation and guidance techniques for control of a Hawk missile on a surface-to-surface trajectory. Requirements are defined for the interface between the DME Weapon Guidance Subsystem and the Hawk flight control system. A DME data link antenna system was designed. Scale model antenna pattern test results are presented which validate the performance for the DME, Telemetry and Command Destruct antennas needed for flight tests. A projected flight test geometry is defined relating the missile trajectory and ALSS ground station locations. Functional requirements are identified for modifications necessary to the ALSS software to provide missile control during the flight tests.

AD-A032155 IBM-76-L61-017 76/10/31 77N23182

Simulation models and baseline guidance and control for indirect-fire missiles with strapdown-inertial guidance

A/JORDAN, W. E. Army Missile Research, Development and Engineering Lab., Redstone Arsenal, Ala. (Guidance and Control Directorate.)

ABS: The simulation models and baseline guidance and

control described in this report were developed to define performance requirements for the airframe, propulsion, guidance, autopilot, and control systems for strapdown-inertially guided indirect-fire missiles. A type of proportional navigation guidance using missile to target relative velocity and position is derived and has the property of being able to shape the missile trajectory for range extension and instrument error minimization. Typical inputs for inertial instruments and control system performance and sizing are obtained.
AD-A024977 RG-76-41 76/01/00 77N12102

Optimal multiple-arc trajectories for an all-aspect air-to-air missile
A/LAWHERN, R. A. California Univ., Los Angeles.
ABS:A three-phase algorithm of steepest descent is developed to investigate the nature of optimal turning strategies in the all-aspect air to air missile. The missile is aimed in approximately the correct direction for interception. The burnout velocity of the missile is maximized with burnout velocity angle constrained to the proper direction for interception. The terminal line of sight angle rate is minimized subject to a constraint on terminal closing range. The algorithm is tested in missile engagements for which optimal behavior is known. Optimal burnout velocity is typically obtained within 5% in 15 total iterations of the algorithm. During trajectory angle optimization, accuracies typical of miss distances less then 100 feet are obtained for all engagements. Several missile-target engagements are examined. 76/00/00 77N11082

Out of line of sight missile link Optelecom, Inc., Gaithersburg, Md.
ABS:This report describes development aimed at producing an optical fiber communication link between a missile and its launch point for transmission of TV data from the missile to the launch point and command signals in the reverse direction. Optical fibers having a loss of 30 db/km were fabricated that were paid out from a spool at speeds of greater than 300 ft/sec.
AD-A024560 76/04/00 77N10421

Weapon delivery impact on active control technology
A/SMITH, H.; B/CARLETON, O. B/(AFFDL) Air Force Armament Lab., Eglin AFB, Fla. In AGARD Impact of Active Control Technol. on Airplane Design 14 p (SEE N75-30027 21-01)
ABS:The need for cooperative efforts among the laboratories/test-organizations and users is emphasized to improve and properly match aircraft pointing and armament component accuracies to achieve the maximum effectiveness with conventional weapons. The Data Measurement Programs of the Armament Development and Test Center/Air Force Armament Laboratory are discussed, including the results and plans for the Instrumented Rack/Bomb and Gunnery Pipper/Fireline Trace and Impact Pattern Model Programs. The Active Control Technology Programs of the Air Force Flight Dynamics Laboratory including objectives, designs, and results of the Tactical Weapon Delivery (TWeaD) Program are discussed. The objectives of the Multimode Control and the Control Configured Vehicle/Advanced Fighter Technology Integrator Programs are delineated. It is concluded that incorporation of active control technology and matched armament component accuracies in future weapon systems shows promise for considerable improvement in the effectiveness of unguided weapons. 71/06/00 75N30040

# SECTION 2

# TERMINAL GUIDANCE AND CONTROL

A singular perturbation analysis of optimal thrust control with proportional navigation guidance

A/CALISE, A. J. A/(Dynamics Research Corp., Wilmington, Mass.) In: Conference on Decision and Control, and Symposium on Adaptive Processes, 16th, and Special Symposium on Fuzzy Set Theory and Applications, New Orleans, La., December 7-9, 1977, Proceedings. Volume 1. (A79-14957 04-63) Piscataway, N.J., Institute of Electrical and Electronics Engineers, Inc., 1977, p. 1167-1176.

ABS:This paper derives a nonlinear optimal thrust control law for a missile using proportional navigation guidance to intercept a maneuvering target. It is shown that using singular perturbation techniques combined with a multiple time scaling approach leads to a control solution that has an algebraic feedback form. A state transformation (similar to the energy state transformation used in aircraft analysis) that can be used to extend the analysis is also derived. Numerical results are given for a short range air-launched missile, and comparisons are made to proportional navigation guidance with boost-coast propulsion. The results show that optimal TMC greatly improves performance for missile launches at high aspect angles relative to the target velocity vector, and when launching from a lag condition relative to the line-of-sight. 77/00/00 79A15014

Optical image processing for missile guidance

A/CASASENT, D.; B/SAVERINO, M. B/(Carnegie-Mellon University, Pittsburgh, Pa.) In: Optical signal and image processing; Proceedings of the International Optical Computing Conference, San Diego, Calif., August 23, 24, 1977. (A79-11986 02-35) Bellingham, Wash., Society of Photo-Optical Instrumentation Engineers, 1977, p. 11-20.

ABS:The paper summarizes the results of a current program in which optical pattern recognition (OPR) is applied to missile guidance. The emphasis in the program is on maintaining correlation in the presence of various image degradations that invariably occur between the input and reference function. The two novel aspects of OPR that are used to reduce the effects of such degradations are control of matched spatial filter parameters and the use of space-variant OPR techniques. Various types of degradations between on-line input imagery and stored reference imagery are considered. 77/00/00 79A11989

Millimeter wave seeker technology

A/OLTMAN, H. G.; B/BEEBE, M. E. B/(Hughes Aircraft Co., Canoga Park, Calif.) In: Guidance and Control Conference, Palo Alto, Calif., August 7-9, 1978, Technical Papers. (A78-50159 22-01) New York, American Institute of Aeronautics and Astronautics, Inc., 1978, p. 148-158.

ABS:The fabrication processes and tests conducted thus far on the 94 GHz microstrip integrated circuit show the feasibility for integrating a microstrip/dipole antenna on the same substrate with other RF circuitry. Scaled frequency tests showed good results which were generally confirmed at 94 GHz. Although microstrip line losses are higher than waveguide losses when compared on a basis of equivalent lengths, such is not true when compared on a basis of equivalent circuits. Losses for equivalent circuits are essentially the same - about 0.8 dB for the described circuit. The approach to fabricate seeker circuits using photo replicating techniques promises to yield low cost, reproducible circuits. It remains to integrate mixers and other seeker components to determine performance limits.

AIAA 78-1259 78/00/00 78A50176

Electro-optic guidance system integration as applied to missiles

A/KNIGHT, G. C.; B/LOPEZ, J. B/(General Dynamics Corp., Pomona Div., Pomona, Calif.) In: Systems integration and optical design II - Another look; Proceedings of the Seminar, Reston, Va., April 18-21, 1977. (A78-40220 17-31) Bellingham, Wash., Society of Photo-Optical Instrumentation Engineers, 1977, p. 110-116.

ABS:Consideration is given to the integration of an electrooptical guidance system into a missile seeker, noting the active, semiactive, and passive guidance modes. The optical system is described with reference to the dome, primary mirror, plane first-surface secondary mirror, field lens, collection lens, and gyrooptics system. The integration of the gyroscopic and optical functions is reviewed and the main causes of gyro drift are identified. The guidance system is discussed in terms of the evolution of the Redeye seeker. 77/00/00 78A40230

Solid-state active RF missile seeker - Future role in all-weather systems

A/MAURER, H. A. A/(Hughes Aircraft Co., Canoga Park, Calif.) In: Military electronics defence expo '76; Proceedings of the Conference, Wiesbaden, West Germany, October 6-8, 1976. (A78-14926 03-33) Geneva, Interavia, S.A., 1977, p. 172-188.

ABS:The paper presents some system considerations concerning the use of solid state active RF missile seekers in all-weather air defense and air-to-air missile systems. Attention is then given to the role of active RF homing guidance, principal supporting technologies, the solid state RF power combiner, and advanced seekers. 77/00/00 78A14935

An adaptive terminal guidance scheme based on an exponential cost criterion with application to homing missile guidance
A/SPEYER, J. L. A/(Charles Stark Draper Laboratory, Inc., Cambridge, Mass.) In: Conference on Decision and Control, 6th, and Symposium on Adaptive Processes, 14th, Houston, Tex., December 10-12, 1975, Proceedings. (A77-12426 02-66) New York, Institute of Electrical and Electronics Engineers, Inc., 1975, p. 660-665.
ABS:For a linear stochastic system minimizing the expected value of an exponential function of a quadratic yields a control law for the terminal guidance problem which operates linearly on the estimated states. The control gains are explicit functions of the error variance in estimating the state. It is shown that the control gains can be calculated by combining a precalculated matrix determined by a backward integration in time with the error variance calculated forward in time. If the measurement variance is estimated in real time, then the error variance must also be calculated in real time. The control scheme will then be adaptive reflecting the estimated quality of the information. The adaptive control scheme is applied to the terminal phase of a homing missile where the measurement variance is estimated on-line. 75/00/00
77A12453

A transformation approach to the terminal control problem
A/SLATER, G. L. A/(Cincinnati, University, Cincinnati, Ohio) AIAA Journal, vol. 14, Sept. 1976, p. 1206-1209.
ABS:The application of optimal control theory to the terminal control problem is investigated. It is shown that a transformation to an alternate problem yields additional insight into the structure of the optimal control, and presents a convenient computational tool for analytical and numerical studies. In particular, if the performance index is a function of only a few of the state vector components, a significant reduction in the numerical effort is possible, and the Riccati matrix can be obtained by a simple numerical quadrature. Application to a missile guidance problem is analyzed. 76/09/00 76A45754

Optimal and suboptimal guidance for a short-range homing missile
A/STOCKUM, L. A.; B/WEIMER, F. C. A/(Rockwell International Corp., Columbus, Ohio); B/(Ohio State University, Columbus, Ohio) IEEE Transactions on Aerospace and Electronic Systems, vol. AES-12, May 1976, p. 355-361.
ABS:Optimal and suboptimal guidance laws for short-range homing missiles are developed and compared to the commonly mechanized guidance law of proportional navigation. The optimal controller is derived as an optimal feedback regulator; the suboptimal controller is an approximation of the optimal regulator and consists of time-varying proportional navigation plus a time-varying gain term times a calculated target acceleration. Monte Carlo studies of the three controllers show that the optimal and suboptimal controllers are much superior to proportional navigation for the case of combined constant target acceleration, line-of-sight rate noise, and missile acceleration saturation. 76/05/00
76A34182

An adaptive terminal guidance scheme based on an exponential cost criterion with application to homing missile guidance
A/SPEYER, J. L. A/(Charles Stark Draper Laboratory, Inc., Cambridge, Mass.) IEEE Transactions on Automatic Control, vol. AC-21, June 1976, p. 371-375.
ABS:For a linear stochastic system minimizing the expected value of an exponential function of a quadratic yields a control law for the terminal guidance problem which operates linearly on the estimated states. The control gains are explicit functions of the error variance in estimating the state. It is shown that the control gains can be calculated by combining a precalculated matrix determined by a backward integration in time with the error variance calculated forward in time. If the measurement variance is estimated in real time, then the error variance must also be calculated in real time. The control scheme will then be adaptive reflecting the estimated quality of the information. The adaptive control scheme is applied to the terminal phase of a homing missile where the measurement variance is estimated on-line. 76/06/00
76A33308

Miss distance position and attitude measurement system
A/POLHEMUS, W. L. A/(Polhemus Navigation Sciences, Inc., Burlington, Vt.) In: NAECON '75; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, June 10-12, 1975. (A75-37623 18-01) New York, Institute of Electrical and Electronics Engineers, Inc., 1975, p. 563-568.
ABS:This paper presents an overview of an electromagnetic transducing concept which would appear to make possible the measurement of relative position and attitude of a missile during close approach to a target drone. The concept is being applied to a variety of very short range guidance, control, navigation and position determination problems. Based on performance of present generation equipment it would appear feasible to provide a miss distance position and attitude sensing system capable of acquiring and tracking a missile when it is within 800 to 1000 feet of the target drone. Missile attitude would be

an output of the system. 75/00/00 75A37695

The generation and application of capture regions in aircraft/missile engagements
A/SALMON, D. M.; B/MEIER, L., III B/(Systems Control, Inc., Palo Alto, Calif.) In: Annual Allerton Conference on Circuit and System Theory, 11th, Monticello, Ill., October 3-5, 1973, Proceedings. (A75-118B7 02-59) Urbana, University of Illinois, 1974, p. 526-535.
ABS:This paper introduces the idea of a capture region in the context of analyzing aircraft/missile engagements. The capture region is the set of initial states of the vehicles from which the missile can eventually intercept the aircraft no matter what evasive maneuvers the aircraft makes. Techniques for numerically generating capture regions are described and some example applications are presented. 74/00/00 75A11893

Silicon waveguide line scanning antenna for millimeter waves
A/KLOHN, K. L.; B/HORN, R. E.; C/FREIBERGS, E.; D/JACOBS, H. Army Electronics Technology and Devices Lab., Fort Monmouth, N. J. (C:Jacobs, Harold)
ABS:Recent demands for a very high resolution radar in terminal homing for missiles and shells and for radar surveillance in general, have generated a need for developing new concepts in low cost millimeter wave antennas. A means of providing electronic line scanning rather than mechanical scanning is desirable in order to reduce system complexity and high cost. It is especially important to eliminate the use of gimbals to mechanically scan an antenna, since they are expensive and slow. This paper describes the design and experimental findings of a novel approach for a side-looking electronic line scanner consisting of a dielectric (silicon) rectangular rod with periodic perturbations on one side. Angular scan is achieved by varying the frequency while the actual numerical values of the scan angles are a function of operating frequency, waveguide size (height and width) and perturbation spacing. An alternative approach was explored where the frequency was held fixed and the effective guide wavelength was varied electronically by modulating the conductivity of a PIN diode mounted on the dielectric waveguide. Antennas were designed for the n = -1 spacial harmonic at operating frequencies in the 55 to 100 GHZ range.
AD-A056456 78/06/00 79N11275

Coherent optical correlation in real time for missile terminal guidance
A/CHRISTENSEN, C. R.; B/UPATNIEKS, J.; C/GUENTHER, B. D. Army Missile Research and Development Command, Redstone Arsenal, Ala.
ABS:The use of area correlation in terminal guidance requires that the system cross correlate a stored reference with the observed scene and have the capacity for handling variations in aspect angle, rotation, scale and intensity. This correlation must be made in real time at a low false alarm rate. Digital techniques can accomplish the preceding requirements but have several limiting characteristics. The number of resolution elements that can be processed is limited by the available core memory. Even with well-chosen algorithms, a large number of multiplications and additions are required and these increase with the number of resolution elements. Parallel processing can reduce the time required to perform this very large number of operations but requires increased complexity and cost.
AD-A056421 78/06/00 79N10114

Sandia inertial terrain-aided navigation system
A/HOSTETLER, L. D.; B/BECKMANN, R. C. Sandia Labs., Albuquerque, N. Mex.
ABS:The concept of utilizing radar-derived terrain profile data for improving the guidance accuracy of weapon delivery systems has received considerable attention in the quest of zero CEP. One of the more successful concepts, to date, was the terrain contour matching (TERCOM) system being developed for cruise missile applications. Basically it obtained a position fix by correlating a radar-altimeter-derived terrain profile with stored topographical data. The location of the best match was taken to be the position of the vehicle. A sequence of such position fixes was used to update an inertial navigation systems.
SAND-77-0521 77/09/00 78N21070

Optimal guidance for modular weapons with digital autopilots
A/YOUNGBLOOD, J. N. Alabama Univ., University. (Bureau of Engineering Research.)
ABS:The optimal linear quadratic guidance problem with an accelerating target is investigated. The missile state is partitioned into a kinematic state and an airframe state. Both the penalty-weighted and constained terminal state cases are treated. The resulting optimal guidance law requires estimates of target accelerations which are derived via linear observers. Results of a point mass missile, a one-time constant missile, and a two-time constant missile are given.
AD-A040449 BER-212-09 AFOSR-77-0691TR 77/04/00 77N29204

Application of differential dynamic programming to an air-to-air missile guidance problem modeled as a differential game

A/FERRARIS, A. H. Air Force Inst. of Tech., Wright-Patterson AFB, Ohio. (School of Engineering.)

ABS:An intercept problem between an air-to-air missile and an aircraft is modeled as a zero sum, free final time differential game which includes nonlinear dynamics and a payoff related to the kill probability. Previous research has shown that the currently used guidance scheme, proportional navigation, is nonoptimal in this type of problem formulation and a higher kill probability is possible with a guidance law based upon a differential game theory. A differential dynamic programming method is applied to the intercept problem in the search for a real-time solution. A convergence control procedure is introduced in an attempt to enhance the convergence of the typically long-time solution methods. The closed-loop guidance law which results is compared to both proportional navigation and some exact open-loop solutions by means of an off-line simulation on a CDC 6600 computer. The method does not yield a real-time solution for this problem and does not give improvement over a proportional navigation scheme.

AD-A034B96 GA/MC/760-7 76/12/00 77N24B98

Self-guidance system of an antiaircraft guided missile

A/NEUPOKOEV, F. Army Foreign Science and Technology Center, Charlottesville, Va. Transl. into ENGLISH from the book ""Strelba Zenitnymi Raketami'' Moscow, Voyenizdat, 1970 p 151-162

ABS:The report describes the homing system of an antiaircraft guided missile. The system can be an active one or a semiactive one; in an active system, the energy source, illuminating the target and the signal receiver is in the missile itself. In a semiactive one, the electromagnetic energy for illuminating the target is not contained in the missile.

AD-A000247 FSTC-HT-23-0680-74 74/06/21 75N17414

# SECTION 3

# SYSTEM INTEGRATION ANALYSIS

Missile control employing control moment gyros
A/ALEXANDER, J. D.; B/DANNENBERG, K. D. B/(Computer Sciences Corp., Huntsville, Ala.) In: Conference on Decision and Control, and Symposium on Adaptive Processes, 16th, and Special Symposium on Fuzzy Set Theory and Applications, New Orleans, La., December 7-9, 1977, Proceedings. Volume 1. (A79-14957 04-63) Piscataway, N.J., Institute of Electrical and Electronics Engineers, Inc., 1977, p. 220-225.
ABS:This analysis investigated the feasibility of employing control moment gyros for control of highly spinning tactical missiles. The missile dynamic equations were linearized through quasi-static analysis about the trim conditions and the gyro precession amplitude was assumed small so that a linear analysis was possible. The gyros were assumed 'perfect' so that the induced roll torque was negligible. Within the limits of the assumptions, the concept appears highly feasible.
77/00/00 79A14967

Bank-to-turn /BIT/ autopilot technology
A/MCGEHEE, R. M.; B/EMMERT, R. I. A/(USAF, Armament Laboratory, Eglin AFB, Fla.); B/(Rockwell International Corp., Missile Systems Div., Columbus, Ohio) In: NAECON '78; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 16-18, 1978. Volume 2. (A78-49851 22-04) New York, Institute of Electrical and Electronics Engineers, Inc., 1978, p. 688-696.
ABS:A bank-to-turn (BIT) steering mechanization has been developed and evaluated for a tactical missile concept, resulting in exceptional short range performance, by employing the maximum maneuvering capability of an unsymmetrical airframe. The control system employs BIT steering, an adaptive autopilot, and proportional navigation (with closing velocity). The BIT steering mechanization allows the large 'g' capability of the airframe pitch axis to be applied in a direction to reduce the total line-of-sight rate. The adaptive autopilot assures adequate performance throughout a large flight envelope, without exceeding critical values of angle-of-attack and side-slip. A small amount of skid-to-turn maneuvering in conjunction with proportional navigation with closing velocity provides high accuracy against maneuvering targets from all aspects. 78/00/00
78A49927

Flight-vehicle stabilization systems /Stabilization of ballistic and antiaircraft rockets/
A/KUZOVKOV, N. T. Moscow, Izdatel'stvo Vysshaia Shkola, 1976, 304 p. In Russian.
ABS:The book deals with closed-loop stabilization and control systems and the mathematical models of rockets and their autopilots. Equations of ballistic and antiaircraft rockets, allowing for longitudinal flexural vibrations and for sloshing in the fuel tanks are derived. The application of methods of modal control to the determination of the structure and parameters of a stabilization system for launch vehicles with mode interaction is demonstrated. The root-locus curve of a closed-loop system is analyzed. Means of stabilizing rotating rockets are examined, along with methods of synthesizing control and guidance systems. Gyroscopic and (platformless) inertial methods of measuring the angular position of a rocket are outlined. 76/00/00
78A15216

Real-time trajectory control using augmented energy management
A/CLAROS, L. N.; B/CRIGLER, S. W. B/(Martin Marietta Aerospace, Orlando, Fla.) In: Guidance and Control Conference, Hollywood, Fla., August 8-10, 1977, Technical Papers. (A77-42751 20-35) New York, American Institute of Aeronautics and Astronautics, Inc., 1977, p. 101-108.
ABS:A real-time near-optimal trajectory controller has been formulated and optimized for implementation onboard an advanced integral rocket ramjet missile. This controller, called Augmented Energy Management (AEM), is based in form on a feedback control law which results from the application of extended energy management (EEM). The AEM controller is developed by modifying the EEM law to include an approximation to the optimal altitude profile, and a density correction filter and energy rate feedback to account for vehicle performance. Numerical examples demonstrate that AEM results in range benefits over a suboptimal controller. Further examples show that when off-nominal conditions are present, AEM is less range sensitive, assuring greater targetable range.
AIAA 77-1052 77/00/00 77A42764

An extended Kalman filter fire control system against air-to-air missiles, volume 2
A/CUSUMANO, S. J.; B/DEPONTE, M., JR. Air Force Inst. of Tech., Wright-Patterson AFB, Ohio. (School of Engineering.)
ABS:This Appendix contains the graphical results of the Monte Carlo analysis of this study. The plots will be presented in sets. All sets will include the dynamic state error plots.
AD-A055637 AFIT/GE/EE/77-13-VOL-2 77/12/00 78N32091

Application of a maximum likelihood parameter estimator to an advanced missile guidance and control system
A/DAYAN, R. Air Force Inst. of Tech., Wright-Patterson AFB, Ohio.
ABS:The problem of parameter estimation using tracking information is examined. Two models are developed and used to estimate the misalignment angles of the inertial system of a missile after its launch. The estimation is based on maximum likelihood concepts. The amount of information extracted from the tracking measurements and the missile specific forces measurements is analysed. A feasibility study of the two models is conducted. The second model uses the aerodynamic model of the missile in order to enhance its estimation ability. Doing this, it incorporates more non-linearities than the first model. These severe non-linearities were found to offset the advantage it had in terms of information gathering. The first model is much simpler in its concept. Yet, it is still able to gather the information needed and its performance is very comparable to the one of the second model. The simplicity and linearity of the first model make it especially attractive.
AD-A055188 AFIT/GGC/EE/77-3 77/12/00 78N31149

Computer program for generating gyroscopic and dynamics stability factors as a function of range
A/FREDERICK, D. L. Frankford Arsenal, Philadelphia, Pa. (Munitions Development and Engineering Directorate.)
ABS:This report describes a computer program which gives gyroscopic and dynamic stability factors as a function of range by combining two existing programs, SPINNER and TRAJE. The program generates air-to-air, air-to-ground, and ground-to-ground trajectories. Inputs to the program are the aerodynamic coefficients obtained from SPINNER computer program, projectile weight and cross sectional area, projectile muzzle velocity, vehicle velocity, and initial angle of fire. Specific information provided includes time, velocity, spin, gyroscopic and dynamic stability factors, and altitude versus range for air-to-ground, air-to-air, and ground-to-ground trajectories. The air-to-ground and air-to-air trajectories include fixed wing and helicopter launchings.
AD-A019242 FA-TN-74020 74/08/00 76N23176

Single engagement laser semiactive system Martin Marietta Aerospace, Orlando, Fla.
ABS:This report documents a study to reexamine the technology of laser semiactive guidance. The objective was to find methods of accomplishing high terminal guidance accuracy with minimum complexity. All three principle system elements were examined: laser designator, the seeker, and the missile control system. While this study chose a general case ballistic ground-to-ground missile as the application, most of the conclusions apply to other missile systems as well.
AD-A005667 OR-13177 74/06/00 75N28416

Angle-of-attack control of spinning missiles
A/PLATUS, D. W. Aerospace Corp., El Segundo, Calif. (Lab. Operations.)
ABS:The application of controlled wind-fixed pitch and yaw moments to control the coning angle of a spinning missile is described. The open-loop response to applied pitch and yaw moments is derived, and the closed-loop behavior of an angle-of-attack control system is analyzed. The angle-of-attack undamping due to a yaw moment is shown to be equal to that for steady roll resonance, and resonance is shown to be a limiting case of yaw moment undamping when the roll rate is equal to the critical frequency. The practical implementation of an angle-of-attack control system is described, and several applications are discussed, including passive angle-of-attack damping, roll lockin prevention, and drag control.
AD-786781 TR-0075(5240-10)-3 SAMSO-TR-74-208 74/07/30 75N13922

Modular digital missile guidance system study
A/HALL, B. A.; B/TAINOR, W. V. Raytheon Co., Bedford, Mass. (Missile Systems Div.)
ABS:This report addresses the feasibility and application of modular digital computers for the guidance function for several classes of air to air missile. Functional requirements, design techniques and algorithms for performing sensor track and stabilization, filtering and estimation, guidance and vehicle control are defined. The requirements imposed upon a digital implementation are determined in terms of throughput, memory and architecture.
AD-784969 BR8073 74/06/30 75N12054

# SECTION 4

# SYSTEM SIMULATION

All-digital correlation for missile guidance
A/CLARY, J. B.; B/RUSSELL, R. F. A/(Research Triangle Institute, Research Triangle Park, N.C.); B/(U.S. Army, Missile Research and Development Command, Redstone Arsenal, Ala.) In: Applications of digital image processing; Proceedings of the International Optical Computing Conference, San Diego, Calif., August 25, 26, 1977. (A79-12003 02-35) Bellingham, Wash., Society of Photo-Optical Instrumentation Engineers, 1977, p. 36-46.
ABS:The requirements for producing a hardware real-time all-digital area cross-correlator are reported. Two-dimensional digital cross-correlation algorithms are evaluated noting two distinct approaches: the straight-forward product and sum cross-correlation algorithm and a high-speed algorithm using the fast Fourier transform. Two digital hardware implementation schemes are presented and an algorithm mechanization procedure is described. It is shown that for a 128 x 128 area cross correlation, the high-speed algorithm reduces the total number of multiples required by about two orders of magnitude as compared to the alternative approach. An all-digital design has been postulated assuming the use of ultra-high-speed, special-purpose, fixed-point, binary-arithmetic hardware. It is found that about 500 integrated circuits, requiring 350 W of power, are necessary to cross-correlate 128 x 128 picture data in real time. 77/00/00 79A12009

Some aspects of valid EMC testing of missiles
A/TSAI, L. L.; B/WU, T.-K.; C/DARONE, R. D.; D/BROWN, G. L. B/(Mississippi, University, University, Miss.); D/(U.S. Army, Missile Command, Redstone Arsenal, Ala.) IEEE Transactions on Electromagnetic Compatibility, vol. EMC-20, May 1978, p. 306-313.
ABS:Integral-equation and numerical techniques are used to determine guidelines in valid electromagnetic compatibility (EMC) testing of missiles. Both thin-wire and body-of-revolution modeling are used. Investigated are two primary aspects, (1) If a near-zone source is used rather than plane-wave incidence, how far must the source be for valid simulation, and (2) how significant is the presence of the rocket exhaust (or plume) in determining subsystem response, and need it be included for valid EMC testing. The simulation validity conclusions reached for the models without apertures apply directly as well to the real-life body with apertures. Numerical results are given over the frequency range of 50-200 MHz to help establish guidelines on testing validity. 78/05/00 78A37123

NAECON '77; Proceedings of the National Aerospace and Electronics Conference, Dayton, Ohio, May 17-19, 1977
Conference sponsored by the Institute of Electrical and Electronics Engineers. New York, Institute of Electrical and Electronics Engineers, Inc., 1977, 1333 p. (For individual items see A78-15552 to A78-15718)
ABS:Consideration is given to design to cost/life cycle costing, flight control, aerospace power system developments, software-compatible avionics processors, operational simulation in lab testing, pointing, tracking and stabilization, and design and integration of avionics digital systems. Attention is also given to tactical guided missiles, higher order languages, airborne communication systems, signal and sensor processing, high capacity memories, fire control technology, electrical insulation for high voltage aircraft systems, airborne radar, display devices, laser gyros, microprocessors, and navigation technology. 77/00/00 78A15551

Complete statistical analysis of nonlinear missile guidance systems - SLAM
A/ZARCHAN, P. A/(Raytheon Co., Missile Systems Div., Bedford, Mass.) In: Guidance and Control Conference, Hollywood, Fla., August 8-10, 1977, Technical Papers. (A77-42751 20-35) New York, American Institute of Aeronautics and Astronautics, Inc., 1977, p. 419-429.
ABS:The Statistical Linearization-Adjoint Method (SLAM), a computerized approach for obtaining complete statistical analysis of nonlinear missile guidance systems, is described. The adjoint technique and covariance analysis, two computerized methods for generating and analyzing rms miss distance and other factors, are reviewed; a computerized technique employing statistical linearization in conjunction with covariance analysis is also discussed. These methods all avoid resort to Monte Carlo techniques that would require a large number of trials to simulate nonlinearities in a stochastic missile guidance system. The SLAM approach, which combines the adjoint technique and statistical linearization, is capable of identifying the chief contributors (e.g., random and step target maneuvers, glint and fading noise) to the total rms miss distance. A sample missile intercept problem is run for each of the approaches discussed; results from the SLAM program and those from the statistical linearization-covariance analysis method are found to be in agreement.
AIAA 77-1094 77/00/00 77A42799

Air combat maneuvering training in a simulator
A/MESHIER, C. W.; B/ROBERTS, J. P. A/(Vought Corp., Dallas, Tex.); B/(USAF, Tactical Fighter Weapons Center, Nellis AFB, Nev.) In: Visual and Motion Simulation Conference, Dayton, Ohio, April 26-28, 1976, Proceedings. (A76-29476 13-53) New York, American Institute of Aeronautics and Astronautics, Inc., 1976, p. 73-82.
ABS:The Aerial Combat Engagement Simulation (ACES) program of the U.S. Tactical Air Command is considered. The program involves the use of a fixed-base visual fighter simulator as a training device to improve the combat skills for operational fighter pilots. The tasks to be simulated are partly related to the employment of radar missiles, heat-seeking missiles, and the 20 mm cannon. Overhead projectors provide each pilot with a computer-generated image of the threat aircraft, a horizon and ground plane, and the F-4E lead computing optical sight system. The effectiveness of the ACES program is evaluated on the basis of the experience which has been obtained in one year of training 76/00/00 76A29486

A new approach in generating missile launch opportunity
A/YI, C. J.; B/CARLSON, D. G. B/(Honeywell, Inc., Minneapolis, Minn.) American Institute of Aeronautics and Astronautics, Guidance and Control Conference, Boston, Mass., Aug. 20-22, 1975, 6 p.
ABS:Launch opportunities for air-to-air missiles can be predicted independently of radar ranging information. By studying the causes of missile miss in the tail-pursuit Air Combat Maneuvering (ACM) environment, the missile launch envelope prediction equations can be reformulated without using range data. This new approach operates accurately under the dynamic conditions of ACM under heavy ECM. The algorithm requires a minimum of digital computation. Simulation results show close correlation between this new algorithm and the existing algorithms using radar data.
AIAA PAPER 75-1120 75/08/00 75A41681

Missile guidance system transformation equations
A/GIBBONS, J. E. Analytic Sciences Corp., Reading, Mass.; California Univ., Livermore. Lawrence Livermore Lab.
ABS:Factors affecting the effectiveness of missile forces are presented, along with an overview of the principal phases of the ICBM flight test program. The principal technical results provide the mathematical basis for relating individual error descriptions in the various coordinate systems used.
UCRL-13827 TR-772-1-2 77/08/01 7CN32170

Application of manned air combat simulation in the development of flight control requirements for weapon delivery
A/BERGER, J. B.; B/MEYER, R. P.; C/CARLETON, D. L. C/(AFFDL) McDonnell Aircraft Co., St. Louis, Mo. In AGARD Flight Simulation/Guidance Systems Simulation 20 p (SEE N76-29287 20-09)
ABS:Manned air combat simulations were conducted to develop requirements for tactical advanced aircraft/weapon systems in which precision tracking and weapon delivery are optimized through flight control system design. The objectives were to (1) develop analytical pilot models that relate weapon delivery accuracy to the entire integrated aircraft/displays/sight/geometry system for air-to-air and air-to-ground weapon delivery tasks, (2) validate and incorporate these pilot models into the Terminal Aerial Weapon Delivery Simulation (TAWDS) digital computer program, and (3) use the TAWDS program to determine how aircraft flying qualities affect air-to-air gunnery, and air-to-ground gunnery and bombing weapon delivery effectiveness. The TAWDS program enables a digital simulation to be performed on various closed loop weapon delivery systems under manual tracking control for predicting and evaluating weapon delivery accuracy. Tracking performance results, acquired from analytical pilot simulations, are compared with those obtained from the manned simulations, and the Tactical Weapon Delivery (TWeaD) flight test development programs. These results indicate that the judicious use of the all-digital analytical weapon delivery program in conjunction with manned simulation studies provides a very cost effective approach in designing, developing, and optimizing advanced aircraft/weapon delivery systems. The evaluation of flying qualities for piloted advanced aircraft, performing air-to-ground weapon delivery tasks in terms of weapon system effectiveness, is shown to be feasible for determining and establishing flight control requirements.
76/06/00 76N29311

Design of an all-attitude flight control system to execute commanded bank angles and angles of attack
A/BURGIN, G. H.; B/EGGLESTON, D. M. Decision Science, Inc., San Diego, Calif.
ABS:A flight control system for use in air-to-air combat simulation was designed. The input to the flight control system are commanded bank angle and angle of attack, the output are commands to the control surface actuators such that the commanded values will be achieved in near minimum time and sideslip is controlled to remain small. For the longitudinal direction, a conventional linear control system with gains scheduled as a function of dynamic pressure is employed. For the lateral direction, a novel control system, consisting of a linear portion for small

bank angle errors and a bang-bang control system for large errors and error rates is employed.
NASA-CR-145004 76/01/00 76N27247

Harpoon missile airborne command and launch system availability model
A/MOON, J. L., III Naval Postgraduate School, Monterey, Calif.
ABS:Two models are developed for calculating the availability of the Harpoon missile airborne command and launch system (HACLS). The first model is a semi-Markov process. Its assumptions are validated using the subsequently developed computer simulation model. Both models are exercised with parametric variations, the critical parameters being mean time to failure, mean time to repair, and severity of the operating environment. Less critical parameters are maintenance efficiency and, for the simulation only, maintenance time to repair probability distribution. A major discovery in this paper is that the standard definition of availability does not prove to be adequate when used to determine system availability in a complex framework of operations.
AD-A018307 75/09/00 76N22243

An estimator for an anti-aircraft gun fire control system
A/PARR, J. M. Naval Postgraduate School, Monterey, Calif.
ABS:Kalman filtering techniques using a rotated coordinate system were applied to tracking problems encountered in fire control systems. Two models of target motion were considered: a constant-velocity model and a model which assumes correlated random accelerations. Estimators derived from these models were evaluated using Monte-Carlo simulations of constant-velocity and maneuvering targets. An algorithm developed to calculate prediction accuracy data for time intervals based on an approximation of the time of flight for a 5 inch/54 Caliber projectile was used to obtain prediction accuracy statistics for evaluating estimator performance.
AD-A005763 74/12/00 75N29858

A flight simulator study of missile control performance as a function of concurrent workload
A/CORKINDALE, K. G. G. Royal Air Force Inst. of Aviation Medicine, Farnborough (England). In AGARD Simulation and Study of High Workload Operations 6 p (SEE N75-12587 03-53)
ABS:Eight pilots took part in a part task simulation of the delivery of a stand-off air-to-surface guided weapon. The attack phase of a sortie was simulated. This phase lasted some 3 minutes and included a low level run to the weapon release area, weapon release, target detection on the TV monitor display and the aiming of the missile at the target. Four levels of workload were studied. The results showed that: (1) performance at the missile control was degraded by increases in concurrent workload; and (2) manual flight control and auto-pilot monitoring were adversely affected by concurrent missile control tasks.
74/10/00 75N12592

# SECTION 5

# SYSTEM TESTING

Some aspects of valid EMC testing of missiles
A/TSAI, L. L.; B/WU, T. K.; C/DARONE, R. O.; D/BROWN, G. B/(Mississippi, University, University, Miss.); D/(U.S. Army, Army Missile Command, Redstone Arsenal, Ala.) In: Electromagnetic compatibility; Proceedings of the Second Symposium and Technical Exhibition, Montreux, Switzerland, June 28-30, 1977. (A78-39076 16-32) New York, Institute of Electrical and Electronics Engineers, Inc., 1977, p. 445-451.
ABS:Laboratory EM compatibility testing of missiles is discussed, with emphasis on the validity of near-zone sources as models for the actual plane wave incidence, and the importance of rocket exhaust or plume in determining subsystem response. Near-zone source effects for various incidence angles are analyzed, and coupling through apertures to interior cavities is assessed. For near-grazing incidence angles, source distances greater than 20 ft provide valid simulation for 10-ft models, while for near-broadside incidence, a source distance greater than 40 ft is needed for 10-ft models. Missile plume effects are investigated through use of the thin-wire and body-of-revolution models. 77/00/00 78A39115

A new test rig for measuring the spin stabilised rocket characteristics
A/ROY, P. K.; B/RAMASWAMY, V. B/(Armament Research and Development Establishment, Poona, India, Defence Science Journal, vol. 27, Oct. 1977, p. 155, 156.
ABS:A test rig designed to measure the forward thrust, chamber pressure and rate of spin of a spin-stabilized rocket under flight conditions is described; values of the three parameters may be presented continuously as a function of time. The test rig has been used to study the test firing of spin-stabilized rockets developing a maximum forward thrust of 2000 kgf and exhibiting a spin maximum of 4000 rpm. 77/10/00 78A33468

Review of MIL-STD-1670/AS/ 'Environmental criteria & guidelines for air-launched weapons'
A/SCHAFER, H. C. A/(U.S. Naval Weapons Center, China Lake, Calif.) In: Environmental technology '76; Proceedings of the Twenty-second Annual Technical Meeting, Philadelphia, Pa., April 26-28, 1976. (A77-26027 10-31) Mount Prospect, Ill., Institute of Environmental Sciences, 1976, p. 387-389. 76/00/00 77A26063

Simulation of pyrotechnic shock in a test laboratory
A/POWERS, D. R. A/(McDonnell Douglas Astronautics Co., Huntington Beach, Calif.) In: Environmental technology '76; Proceedings of the Twenty-second Annual Technical Meeting, Philadelphia, Pa., April 26-28, 1976. (A77-26027 10-31) Mount Prospect, Ill., Institute of Environmental Sciences, 1976, p. 5-9.
ABS:The paper tries to explain why generating a pyrotechnic shock spectrum that identically matches one obtained during flight does not necessarily mean that a satisfactory test has been performed. The nature of failures from shock loading is investigated and pyrotechnic shock simulation involving the following test devices is reviewed: drop testers, shakers controlled by shock synthesizers, and pyrotechnic shock fixtures. 76/00/00 77A26030

Component mode analysis of the Harpoon missile, a comparison of analytical and test results
A/GUBSER, J. L.; B/ZARA, J. A. B/(McDonnell Douglas Astronautics Co., St. Louis, Mo.) American Society of Mechanical Engineers, Winter Annual Meeting, Houston, Tex., Nov. 30-Dec. 4, 1975, 33 p.
ABS:Harpoon is a U.S. Navy anti-ship missile capable of being launched from aircraft, surface ships and submarines. As part of the overall structural dynamic design evaluation, modal analyses and tests were performed in support of aeroelastic and forced response studies. The mod l analysis of various missile configurations was accomplished using a component mode coupling technique with component modes consisting of those for (1) the Harpoon body assuming all aerodynamic surfaces to be rigid and (2) each of the aerodynamic surfaces, cantilevered. Modal survey tests were conducted on each type of aero surface in the cantilevered condition as well as on missile configurations. Test and analysis results are compared for a number of the aerodynamic surfaces and a single missile configuration.
ASME PAPER 75-WA/AERO-6 75/11/00 76A21853

Determination of aerodynamic coupling derivatives through flight test
A/DRISCOLL, T. R.; B/STOCKDALE, R. C.; C/SCHELKE, F. J. C/(Martin Marietta Aerospace, Orlando, Fla.) American Institute of Aeronautics and Astronautics, Guidance and Control Conference, Boston, Mass., Aug. 20-22, 1975, 7 p.
ABS:The control of a highly responsive surface-to-air missile is dependent upon the aerodynamic characteristics of the selected airframe. Although the aerodynamic forces and moments are obviously important, the partial derivatives of these variables are the primary

characteristics that determine the stability of the control system. Of particular interest are the aerodynamic coupling derivatives (those forces and moments induced in one autopilot channel by changes in another channel). These coupling characteristics can be accurately determined during the flight test program by (1) including test sequences that excite these coupling forces and produce a measurable response, and (2) developing a technique to equate these flight test responses to numerical values of the aerodynamic derivatives. This paper describes such a technique developed for a typical surface-to-air missile and presents results based on actual flight test sequences.
AIAA PAPER 75-1119 75/08/00 75A41680

Accelerated reliability testing under vibroacoustic environments
A/MEEKER, D. B.; B/PIERSOL, A. G. A/(U.S. Naval Missile Center, Point Mugu, Calif.); B/(Bolt Beranek and Newman, Inc., Canoga Park, Calif.) In: Reliability design for vibroacoustic environments; Proceedings of the Winter Annual Meeting, New York, N.Y., November 17-21, 1974. (A75-18135 06-38) New York, American Society of Mechanical Engineers, 1974, p. 139-155.
ABS:This paper discusses the design of accelerated reliability tests for complex aerospace systems exposed to vibroacoustic environmental loads. Past efforts to formulate an appropriate relationship for the tradeoff between the duration and the intensity of applied loads are reviewed. The results of recent experimental studies of the failures of an airborne missile under simulated captive flight loads are then presented. These results suggest that the overall mean-time-to-failure (MTTF) of the missile is inversely proportional to approximately the fourth power of the rms value of the vibration environment. However, the results also reveal that the distribution of failures among different types of components varies significantly with the vibration level. For example, mechanical wearout and electromechanical malfunctions constitute the majority of failures at the lower vibration levels while vacuum tube failures dominate at the higher vibration levels. It follows that if a reliability test of a complex system is accelerated too severely, the test might produce failure distributions which are not representative of service experience. 74/00/00 75A18139

Comparison of theoretical and measured signal and noise outputs of a passive 35-GHz radiometer
A/KASTE, O. C. Ballistic Research Labs., Aberdeen Proving Ground, Md.
ABS:Two expressions for calculating radiometer antenna temperature changes when a target is present are derived and compared favorably with a simple expression frequently used in the literature. One of the new expressions can be used to calculate the output signal pulse of a radiometer as it passes near or over a target, and can be used when the radiometer antenna axis is tilted from the vertical. Experimental data are compared with theoretical data; agreement is generally very good. A theoretical expression for the noise output of a radiometer was obtained from the literature. Values from this expression are found to be in good agreement with noise data obtained from laboratory and field measurements.
AD-A040366 BRL-MR-2745 77/04/00 77N29137

A wind tunnel investigation of impulse effects on the motion of an impulse correction guidance missile
A/USELTON, B. L. ARO, Inc., Arnold Air Force Station, Tenn. AEDC
ABS:Wind tunnel tests were conducted for the Air Force Armament Laboratory to obtain experimental data at Mach number 3 on an impulse correction guidance system. The guidance system is based on the principle of impulse correction. The purpose of the test program was to determine if the interaction of the impulse explosion with the supersonic airflow caused an effect on the model motion. The small-amplitude free-oscillation technique was used to obtain data on a 0.5 scale model at angles of attack from 1 to 4.4 deg at Reynolds numbers, based on model length, of 12.2 x 100,000 and 23.1 x 100,000. The explosion which produced the impulse affected the model flow field. However, this perturbed flow field did not produce any significant effects on the model motion, apparently because of the short action time involved.
AD-A024210 ARO-VKF-TR-75-155 AEDC-TR-76-6 AFATL-TR-75-164 76/04/00 77N10126

Navy evaluation F-11A in-flight thrust control system
A/SIMPSON, W. R.; B/COVEY, M. W.; C/PALMER, D. F.; D/HEWETT, M. D. Naval Air Test Center, Patuxent River, Md.
ABS:A Navy evaluation to determine the potential advantages and disadvantages of in-flight thrust control (IFTC) on a tactical airplane was conducted using a modified F-11A airplane as a testbed. The conceptual development program also utilized a second unmodified F-11A for baseline data and pilot familiarization training. Flying qualities, performance, engine effects, durability, and utility of IFTC to mission tasks such as air combat maneuvering (ACM), air-to-ground weapons delivery, approach and waveoff, landing roll-out and infrared signature suppression were evaluated during the 6-month program. The prototype IFTC in the configuration evaluated increased the tactical capabilities of the F-11A airplane despite the

limited capability of the testbed, indicating potential increases in tactical capabilities of future fighter/attack airplanes which incorporate thrust control capability.
AD-A019954 NATC-SA-75R-75 75/12/15 76N25204

Missile radar guidance laboratory
A/MONROE, R. D.; B/GREGORY, P. C. Martin Marietta Aerospace, Orlando, Fla. In AGARD Range Instrumentation, Weapons Systems Testing and Related Techniques 20 p (SEE N76-23283 14-09)
ABS:An improved radar guidance laboratory which allows simultaneous infrared simulation for developing and testing point tracker radar and IR dual mode guidance systems which will be operational in the 1980's is described. These guidance systems will be tested for target acquisition, discrimination, and tracking capabilities under precisely controlled conditions in a dynamic, real-time simulated environment. The radar guidance types can be passive, semi-active or active, covering a frequency range from 0.5 to 18.0 GHz. The IR guidance systems can be passive at 3 to 5 or 8 to 14 microns. A short review of system requirements is furnished, and the major laboratory subsystems are described, with emphasis on the features of the rotational and translation motion systems, anechoic chamber, linear array target antenna system, radar generation system, IR target system, and computation. The principal new design features of this laboratory are the linear array target antenna system and the radar generation system which provides for four distinct radar emitters each of which can simulate simultaneous, independent RF sources. These sources can be surveillance, SAM, search or early warning radars, plus radar returns from illuminated targets, and types of pulsed and continuous wave ECM signals. Phenomena such as atmospheric attenuation, Doppler shift, target cross section deviation, and glint are also simulated. Criteria used to specify the required system performance, the reasons for criteria selection, and the laboratory test results are also included. 76/02/00 76N23302

Wind tunnel test results for the direction controlled antitank DCAT missile at Mach numbers from 0.64 to 2.50
A/MARTIN, T. A.; B/SPRING, D. J. Chrysler Corp., New Orleans, La. (Space Div.)
ABS:Wind tunnel test results are presented to show aerodynamic characteristics over the Mach number range of 0.64 to 2.50 of the DCAT missile. Data are presented showing the interference created by the rear mounted reaction control system. Two candidate fins were installed on the model during tests: a flat folding fin and a curved wrap around fin.
NASA-CR-140750 AD-784121 RD-73-27 73/10/12 75N11021

# SECTION 6

# WEAPON PERFORMANCE

Supersonic aerodynamic characteristics of a tail-control cruciform maneuverable missile with and without wings
A/SPEARMAN, M. L.; B/FOURNIER, R. H. B/(NASA, Langley Research Center, High-Speed Aerodynamics Div., Hampton, Va.) National Aeronautics and Space Administration. Langley Research Center, Hampton, Va. In: Atmospheric Flight Mechanics Conference, Palo Alto, Calif., August 7-9, 1978, Technical Papers. (A78-46526 20-08) New York, American Institute of Aeronautics and Astronautics, Inc., 1978, p. 162-165.
ABS:The aerodynamic characteristics for a winged and a wingless cruciform missile are examined. The body was an ogive-cylinder with a 3.5 caliber forebody; an overall length-to-diameter ratio of 11.667; and has cruciform tails that were trapexoidal in planform. Tests were made both with and without 72.9 deg cruciform delta wings. The investigation was made for Mach numbers from 1.50 to 4.63, roll attitudes of 0 and 45 deg, angles of attack from -40 to 22 deg, and tail control deflections from 10 to -40 deg. The purpose is to determine the influence of the aerodynamic behavior on the design choice for maneuverable missiles intended primarily for air-to-air or surface-to-surface missions. The results indicate that the winged missile with its more linear aerodynamic characteristics and higher lift-curve slope, should provide the highest maneuverability over a large operational range.
AIAA 78-1351 78/00/00 78A46544

Missile aerodynamic sensitivity analysis
A/QUAM, O. L. A/(Dayton, University, Dayton, Ohio) American Institute of Aeronautics and Astronautics, Aerospace Sciences Meeting 16th, Huntsville, Ala., Jan. 16-18, 1978. 11 p.
ABS:An investigation was conducted into the relative importance of aerodynamic parameters on air-to-air missile performance. Although the effect of drag on range is quite obvious, the effect of damping derivatives on the performance of a homing missile is quite subtle. A simplified method of generating analytical estimates of miss distance showed surprisingly good agreement with numerical trajectory computations. The resulting ranking of parameters in descending order of importance on both range and miss distance is: drag and lift curve slope; stability and control moment derivatives; control force derivative and nonlinear lift; damping derivatives and a nonlinear moment parameter.
AIAA PAPER 78-113 78/01/00 78A20690

Performance optimization of an air-to-air missile design
A/EICHLER, J. A/(Negev, University, Beersheba, Israel) Journal of Spacecraft and Rockets, vol. 14, June 1977, p. 376, 377.
ABS:The paper describes a merit function for air-to-air missile designs which includes the salient design features and is based on performance, as evidenced by the capability to achieve a hit against a maneuvering target from a large number of different launch conditions. The problem is quantified by selecting 5 angles of line of sight (LOS) radii, 3 directions of launch for each LOS radius, and two types of maneuvering targets, thereby giving a 30-point quantification for the measure of missile performance. The merit function is calculated by finding for each case the minimum radius of launch that results in a hit. The merit function together with a Davidson variable metric conjugate gradient optimization technique is used to find an optimum set of design variables of a fictitious missile design.
77/06/00 77A34298

Optimal switching criteria for two-position configuration controls
A/GLAROS, L. N., JR. A/(Martin Marietta Aerospace, Orlando, Fla.) Journal of Spacecraft and Rockets, vol. 14, Feb. 1977, p. 124, 125.
ABS:Optimal switching criteria for two-position configuration controls in aerospace vehicles are developed. Performance measures to be optimized include: minimum time-to-climb, minimum fuel-to-climb, and maximum range with a fixed fuel weight. The controls can be applied to afterburners (on/off), folded wings (deployed/folded), and two-position variable-geometry nozzles. The switching criteria depend solely on the point performance capabilities of the vehicle and are totally independent of boundary conditions. An example is analyzed for maximum range of a ramjet missile featuring variable-geometry nozzles. 77/02/00 77A22709

Naval tactical air warfare of the future
A/PETERSEN, F. S. A/(U.S. Navy, Naval Air Systems Command, Washington, D.C.) American Institute of Aeronautics and Astronautics, Annual Meeting and Technical Display Incorporating the Forum on the Future of Air Transportation, 13th, Washington, D.C., Jan. 10-13, 1977, 7 p.
ABS:An overview of prospective tactical weaponry and naval aircraft for the quarter-century ahead is presented with a brief survey of progress during the past quarter-century as reference point. Missiles propelled by integral rocket ramjet systems and with terminal guidance offering zero circular error probability via SMAC (scene matching area correlators), quality midcourse adaptive

guidance which is compact, lightweight, and low-cost, sophisticated data displays, and V/STOL combat aircraft are seen as major systems of promise, along with low-cost night attack weaponry. Advances in onboard electrical systems, composite materials, simulation and training systems, digital flight controls, and weather predictions are also anticipated. Fuel resources are seen as a major problem looming ahead.
AIAA PAPER 77-333    77/01/00    77A18251

Computer design requirements for digital air-to-air missiles
A/HALL, B. A.; B/LANGLEY, F. J.; C/WEFALO, K. O. C/(Raytheon Co., Missile Systems Div., Bedford, Mass.) In: Guidance and Control Conference, San Diego, Calif., August 16-18, 1976, Proceedings. (A76-41426 20-12) New York, American Institute of Aeronautics and Astronautics, Inc., 1976, p. 514-533.
ABS:It is shown that modular digital guidance and control is effective in improving air to air missile performance. Using a common bus interface, a family of ten major computer function elements, hybrid LSI macromodules, will support the entire range of missile functions. Digital guidance and control systems will partition into four autonomous and asynchronous groups: (1) target seeker, (2) gimballed platform stabilization, (3) flight control, and (4) warhead fusing. Federated microcomputer systems enable separable missile functions to be matched with a computer's processing capability, and provide the desired subsystem autonomy for modular design, manufacture, assembly, test, maintenance and subsequent modification without system disruption.
AIAA 76-1977    76/00/00    76A41483

Supersonic aerodynamic characteristics of a Sparrow 3 type missile model with wing controls and comparison with existing tail-control results
A/MONTA, W. J.    National Aeronautics and Space Administration, Langley Research Center, Hampton, Va.
ABS:An experimental investigation was conducted on a model of a wing control version of the Sparrow III type missile to determine the static aerodynamic characteristics over an angle of attack range from 0 deg to 40 deg for Mach numbers from 1.50 to 4.60.
NASA-TP-1078 L-11715    77/11/00    78N12041

The technology of precision guidance--changing weapon priorities, new risks, new opportunities
A/DIGBY, J.    RAND Corp., Santa Monica, Calif.
ABS:For centuries most of the things shot by military men at their enemies have missed their target. The remarkable thing about the new generation is that it is now possible for forces to possess weapons in large numbers each of which has a high probability of hitting its target with a single shot. This article discusses the implications of these weapons, which are officially called precision-guided munitions or PGMs. Usually, this simply means a bomb or missile that is guided in its terminal phase. Thus, the term includes many anti-tank weapons (including some which receive steering signals over thin wires) and air-defense missiles, as well as the laser-guided bombs which attracted so much public attention.
AD-A026653 P-5537    75/11/00    77N14441

Design and analysis of air-to-air missile using digital control thesis
A/CALLEN, T. R.    Air Force Inst. of Tech., Wright-Patterson AFB, Ohio.    (School of Engineering.)
ABS:The design of automatic control systems is one of the most critical and important tasks that the air-to-air missile control engineer must accomplish. The advantages of low cost, high reliability and low power requirements, along with the small space requirements, make digital controllers a very attractive device for this purpose. This thesis presents the engineering techniques that can be employed to develop a mathematical model of a generic missile and also to design a digital controller for the system. The basic missile's stability and performance are evaluated in both the continuous and discrete domains, for angles of attack of 0 and 30 degrees. The effects of sampling time are demonstrated, and direct digital design techniques are presented, with the resulting digital controllers being evaluated as to their effect on system performance. Pitch rate control is investigated in addition to pitch attitude control.
AD-A019853 GE/EE/75-17    75/12/00    76N26267

Effects of certain configuration parameters on a particular air to air interceptor missile with optimal guidance
A/RUSSAK, I. B.    Naval Postgraduate School, Monterey, Calif.
ABS:During the summer of 1973 at the request of the Naval Missile Center, Pt Mugu, CA, the author initiated a study to determine how much potential performance limits could be improved for a conceptual air to air interceptor missile through the use of a variable thrust engine together with optimal guidance techniques. Significant

improvements n performance were achieved in the sense of reducing time to intercept. The present work extends those results by showing the sensitivity of this improvement to configuration changes such as the inclusion of aerodynamic surfaces and thrust-impulse level changes to the missile.
AD-A019941 NPS-54RU75103 75/.0/00 75N26266

Results of some investigations of differential gain theory applied to air-to air systems. Volume 4: Intercepting an accelerating target using quasi-optimal control for air-to-air systems
A/EIDE, H. F. California Univ., Los Angeles. (School of Engineering and Applied Science.)
ABS:The technique of quasi-optimum control, developed by B. Friedland is applied to the problem of intercepting an accelerating target with an air launched missile. The problem is first stated as an optimal control problem with non-linear system equations and a quadratic cost functional. Several techniques of finding feedback type optimal controls which have been developed by solving various approximations to the original problem are then described. None of these techniques makes explicit use of a measurement of lateral target acceleration. Three control techniques are then developed which use measurements of target acceleration to develop closed form controls. The first control is found by solving a linearized version of the original problem. The other two controls are quasi-optimum controls which use the solution of the linearized problem to obtain a new feedback type solution that is more nearly optimal for the original problem. Based on performance in simulations of typical attack geometries and target maneuvers, these three control techniques are found to be superior to previous techniques.
AD-A020094 AFFDL-TR-75-76-VOL-4 75/08/00 76N25150

Demonstration of multistation/CSP capability for range control
A/GABLER, R. T.; B/HABER, J. M. Wiggins (J. H.) Co., Redondo Beach, Calif.
ABS:The required accuracy in impact prediction for range control purposes can be achieved only through better velocity estimation. Coherent signal processing (CSP) was developed to provide the C band radars with a Doppler or range rate capability and in a multistation solution to give the required better velocity estimate. Even though various studies tend to show the possibility of substantial improvement through the use of multistation with Doppl.r solutions, very limited use has been made of multistation solutions for real time range control purposes. A primary purpose of this task was to develop the means for and evaluate the capability of CSP and a range instrumentation system for meeting the IIP requirements of range control. To make the evaluation as complete and definitive as necessary for the purpose, both random and systematic error in IIP were evaluated and for both nominal trajectories and for missile failure cases. Computer programs were developed to facilitate this evaluation.
TR-73-7005-1 73/04/07 76N22428

Direct statistica analysis of missile guidance systems via CADET (covariance analysis describing function technique)
A/TAYLOR, J. H.; B/PRICE, C. F. Analytic Sciences Corp., Reading, Mass.
ABS:The Covariance Analysis Describing Function Technique (CADET) -- a technique for the efficient direct-statistical analysis of nonlinear systems with random inputs -- is extended in scope to permit the study of a complicated, highly nonlinear model for a tactical missile homing guidance system. Numerous parameter sensitivity studies are performed with selected cases verified by the monte carlo method. The validity of the assumptions underlying the CADET theory is investigated and the impact of possible errors in these assumptions on the accuracy of CADET is assessed. In every realistic situation studied, CADET provided accurate missile performance projections with a small fraction of the computer time required for a comparably reliable monte carlo analysis.
AD-783098 TASC-TR-385-1 74/08/01 75N10162

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**14. Abstract**


As a result of significant and extensive developments in modern control theory in recent years there is a need to keep under continuous review their possible impact upon the design of tactical guided weapons. It is the purpose of this Lecture Series, therefore, to summarize the state-of-the-art of guidance and control for tactical weapons and to pay particular attention to GW simulation techniques (digital, hardware-in-the-loop development, validation) and the testing of missile guidance and control systems.

The other principal subject areas to be reviewed are weapon delivery (including targeting and acquisition), missile control techniques, and current guidance techniques (both mid-course and terminal, guidance sensors, and processing). Finally, consideration will be given to future trends.

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