GUIDANCE AND CONTROL DEVELOPMENT OF THE MULTIMODE AIRFRAME TECHNOLOGY (MAT) MISSILE

Justin L.S. Walls
Senior Technical Staff
Nichols Research Corporation
4040 South Memorial Parkway, MS 922
Huntsville, AL 35924

This research was supported in part by a contract from the U. S. Army Aviation and Missile Command (AMCOM), Redstone Arsenal, AL 35815.

ABSTRACT

This paper overviews development of the guidance and control system design of the Multimode Airframe Technology (MAT) missile being developed at the Missile Research, Development, and Engineering Center, U.S. Army AMCOM. The teleoperated missile accomplishes high-speed flyout, slowdown for target search using a variable wing/flap configuration, and provides high-speed precision targeting at terminal attack. The control system's objective is to achieve these flight regimes and overall mission objectives. A high fidelity six degree-of-freedom (6DOF) simulation of the missile system was likewise developed that serves as a development and testing environment. Simulation results are given to show the performance of the closed loop system. Confidence in the modeling efforts was obtained through testing of each subsystem and comparing with the six-DOF simulation. The MAT missile simulation results show that the guidance and autopilot functions are achieved with respect to the mission requirements.

INTRODUCTION

The MAT missile mission (40-100km) begins with launch using a turbojet propulsion system given in Figure 1. The missile embarks on a high subsonic fly-out to a pre-determined target area (from mission planning) that may include waypoints. An on-board inertial measurement unit, IMU, and GPS receiver are utilized to provide location coordinates and missile attitude. Once in the target area, the missile deploys the wings in a bi-plane configuration, and utilizes flaps to provide braking to a lower speed allowing target search. After target acquisition, the flaps are retracted, the wings transition from bi-plane into a 'X' cruciform configuration which permits skid-to-turn maneuvering capability in the pitch and yaw axes. The turbojet engine is commanded to maximum throttle to increase the missile velocity during dive and attack. Performance data of the MAT missile is given in Figure 2. These varied missile objectives provide challenges in designing the guidance and control system addressed in this paper.

CONTROL OBJECTIVES AND MODEL DEVELOPMENT

The overall guidance/autopilot functions are summarized as follows:
- Stability augmentation of the airframe to obtain the desired damping and response for expected variations in airframe system parameters
- Placement of the missile into level flight at the commanded altitude above the launch site - altitude-hold function
- Control of missile heading to the commanded azimuth in the absence of guidance commands - attitude-hold function
- Appropriate response to altitude and guidance commands including waypoint control
- Position or heading control of the missile in the midcourse phase of flight, between the start of level flight and target designation
- Generation of proportional navigation guidance and/or pursuit guidance after target designation or during manual tracking
- Simulation of gunner station supplied commands/corrections (manual mode)

A block diagram of missile subsystems is presented in Figure 3.

The missile control system is built around an inner rate stabilization loop shown in Figure 4. Attitude stabilization, altitude control, guidance, and waypoint navigation is achieved for various aerodynamic configurations by using attitude rate, angle rate, and position feedback loops. Gains and filters were chosen in the rate autopilot for each axis to yield a stable response over the range of flight conditions and in each of the flight phases - boost, cruise, search, and endgame. The flight conditions vary dramatically depending on the missile velocity, cg location, and the wing and flap geometries and associated deployment/retraction rates. Wind tunnel data was taken of the flight configurations. The aero data was tabularized and the computed aero values are a function of inputs aero mode, turbojet engine flag, monte-carlo on/off flag, velocity, Reynolds number, dynamic pressure, freestream velocity, engine thrust, cg location, angle of attack, sideslip, Fin positions, flap deflection, monte-carlo variations, and angular velocities [2]. The outputs of the aerodynamic model are the aerodynamic forces (FZ, FY, FX), moments (FLMCG, FLNCG, FLL), coefficients (CZ, CLMCG, CY, CLNCG, CLL, CX), and hinge moment values for each of the four fins and top / bottom flaps. While the 6DOF simulation does not specifically use the aerodynamic coefficients, they are used for the linear autopilot design. The turbojet engine parameters that appear in the input list are used to calculate the aerodynamic fin-in-plume (FIP) effects. Analysis reveals that these FIP effects are necessary to provide adequate control authority.

The data for four representative operating points were generated for analysis of the demonstration flight. The demonstration flight scenario involves each of three flight aerodynamic mode configurations: Top wing at launch, Bi-wing during slowdown and search operations, and an X-wing configuration for the terminal phase of flight. The flaps are restricted to operate in the bi-wing mode. Control of the top and bottom flaps is achieved by individual deployment / retraction commands. Currently, the flaps are commanded to fully deploy or retract, that is, no intermediate command values are assumed for analysis. Although, the deployment / retraction rates along with intermediate aerodynamic effects are used in the 6DOF simulation. Flap status signals are used for confirmation, although overriding logic is provided in case of status signal failure. The precise flap deployment and retraction rates are to be determined. Currently, the flaps deploy in approximately six seconds and retract in three seconds. The bottom flap is retracted at a TBD velocity above the search velocity. This factor is chosen to minimize the overshoot and settling time in achieving the desired search velocity. Representative operating points for boost, cruise, search, and terminal were selected for analysis. The maximum fly-out speed is limited to 550 ft/sec (167.6 m/s). This is likewise the terminal / endgame velocity, while the search velocity is 320 ft/sec (98 m/s). This is the limit of fiber payout speed (167 m/s), although, the aerodynamic model appears to be accurate to mach 0.65 (221 m/s).

The basic form of the inner rate loop remains the same for each autopilot objective (attitude stabilization, altitude-hold, velocity control, and terminal attack), although switching occurs between stabilization filters and gain blocks change. The feedback gain block is a velocity dependent pitch / yaw rate gain QFB_K, RFB_K given by

$$QFB\_K = QFB0 - QFB1 * VMEST$$
$$RFB\_K = RFB0 - RFB1 * VMEST \tag{1}$$

Figure 5 is a block diagram of the Altitude hold loop with an embedded inner rate loop. The feedback gains allow various weighting of the sensor measured feedback angles (PHI1, THT1, PSI1). The feedback gains were tuned to each of the flight phases. The gains used in the autopilot for each axis were chosen to yield a stable response over the range of flight conditions. Gain scheduling is used for each of the modes. The initial system design was completed by performing an analysis (root loci, frequency, and time response) of the system in each configuration using the software package MATLAB. The gains were later tuned using the 6DOF simulation. Differences in the analysis gains and filter parameters and the 6DOF parameters exist due to nonlinearities and approximations used in each case. Each of the cases was analyzed using MATLAB for stability using realistic physical (reference area, diameter, target range, inertia, mass and cg location, and component positions) and atmospheric (dynamic pressure) values for the test flight conditions.

The overall missile body dynamics is composed of both rigid and flexible-body components. The autopilot was initially designed using classical techniques [3] with no flexible body dynamics present. As additional information became available, flex-body mode dynamics were added to the simulation. The reader is referred to [4] for a limited derivation of the dynamic model used. Using modal analysis techniques, the first bending mode for both the pitch and yaw axes was experimentally measured. The modal analysis results are in the form of model frequency, damping, shape, and residues. This information yields the coefficients of a transfer function model. Inputs to the transfer functions (for the MAT missile) are the change in aerodynamic forces (F_{fy}, F_{fz}) on the missile from steady state values and fin torques (T_{fy}, T_{fz}). In each axis (pitch and yaw), the transfer functions for each axis acceleration bending component is

$$Y_i accel = \frac{u_{iy} s^2 (u_{fy} F_{fy}(s) + \theta_{fz} T_{fz}(s))}{s^2 + 2\zeta\omega_y s + \omega_y^2}$$

$$Z_i accel = \frac{u_{iz} s^2 (u_{fz} F_{fz}(s) + \theta_{fy} T_{yz}(s))}{s^2 + 2\zeta\omega_z s + \omega_z^2} \tag{2}$$

while the body angular rates are described by

$$\Psi_i rate = \frac{\psi_{iz} s^2 (u_{fy} F_{fy}(s) + \psi_{fz} T_{fz}(s))}{s^2 + 2\zeta\omega_y s + \omega_y^2}$$

$$\Theta_i rate = \frac{\theta_{iz} s^2 (u_{fz} F_{fz}(s) + \theta_{fy} T_{yz}(s))}{s^2 + 2\zeta\omega_z s + \omega_z^2}, \tag{3}$$

where ζ is the damping ratio of the mode (3%) and the ω's are the axes mode frequencies.

The compensation filter is needed to reduce the acceleration levels caused by exciting and feeding back the flex-body dynamics, and achieve the desired performance for each of the operating conditions. This is accomplished via a third order digital filter. A compensation filter was designed using classical techniques [3] added in the pitch and yaw axes to minimize the flexible-body dynamics over the performance envelope. The filter was designed by iteratively choosing the pole and zeros values and gains that yield satisfactory angles of departure, 60 deg phase margin, and damping ratio. The filter was added to the pitch and yaw rate feedback paths, respectively. For the MATLAB analysis and design, the flex-body modes were linearly added to the rigid body dynamics (not necessarily a good assumption). in an open loop fashion. Actuator feedback that excites the flex-body modes was accounted for in the 6DOF simulation. The simulation provided the values (levels) for the flex-body aerodynamic forces and fin torques used in the MATLAB analysis. The physical airframe parameters were generated for a typical ground target engagement, and the autopilot designed to yield approximately sixty degrees phase margin. A simplified roll axis transfer function is given by

$$\frac{phidnum}{phidden} = \frac{L_d}{s + L_p} \tag{4}$$

where Ld and Lp are defined by

$$L_d = \frac{QAPSD * cld}{I_{xx}} \tag{5}$$

$$L_p = \frac{QAPSDM * clp}{I_{xx}}$$

where QAPSD is the product of dynamic pressure, reference area and missile diameter, and QAPSDM is the product of QAPSD and missile diameter divided by 2*VRW. Cld (aero variable CLL) is the rolling moment resulting from a fin deflection (DELP), and Clp is the roll damping (Clp = -67.7). The roll loop is stabilized using aerodynamic mode and velocity dependent gains. Inertial coupling between the roll and yaw axes is evident as the missile transitions between waypoints and changes aerodynamic configurations. This coupling must be considered in the design to achieving roll stabilization as the missile traverses around the waypoint. A more stringent roll control requirement is necessary during search operations, target lockon, and terminal homing to minimize the optical roll as seen by the missile gunner.

SIMULATION STRUCTURE AND RESULTS

The control system design is embedded within the 6–DOF simulation and uses realistic physical models/parameters (reference area, diameter, target range, inertia, mass and cg location, and component positions), aerodynamic modes, and atmospheric (dynamic pressure, temperature, launch elevation, etc.) values to simulate the test flight conditions. Differences in the results of the analysis and the 6DOF exist due to nonlinearities, linearity assumptions, simplifications, and approximations used in each case.

The use of a 6-DOF simulation allows checkout, design, and model verification of new aerodynamic windtunnel data, hardware subsystem validation, and verification of flight dynamics. Models of individual hardware subsystems are tuned to specific hardware performance when data is available. A significant advantage in using the modular simulation structure is ability to efficiently create and add new models based on hardware testing of each missile subsystem.

The MAT missile simulation was debugged and tested to verify nominal performance for the Eglin Tech Demo flight. The testing suite uses a fixed ground target consistent with the Eglin test range. Once the initial debugging process was completed, the simulation was tested in both deterministic and Monte Carlo fashions. Testing of the second on-board autopilot (Lost link failure mode) has included not only replicating the ground processor simulation results, but also achieving repeatability with several identical deterministic simulation runs.

Monte-Carlo runs were tested and compared for statistical accuracy. For the case of the Eglin demonstration flight, the ground target scenario was executed with a target range of 40 Km with two waypoints as defined by test range boundaries and safety considerations. For a 50 run Monte-Carlo set, the performance statistics are presented for the nominal representative target engagement. Key flight variables are shown in Figures 6 – 11 with ALTPLT being the altitude above ground level in feet, VMEST is the sensor measured velocity (m/sec), PHI1,THT1,PSI1 (rad) are the roll, pitch, and yaw angles, respectively, and YME is the crossrange in meters. Variations within simulation runs exist due to the refinements and upgrades to the main simulation environment and additional error source variables. Similarly, differences from previous simulations are due to the additional modeling efforts incorporated within the simulation such as adding the constraints of the launcher, initial launch angle, tip-off rates, higher fidelity engine / engine controller models, flap misalignments, and separate deployment of the top and bottom wings to the cruciform configuration.

CONCLUSION

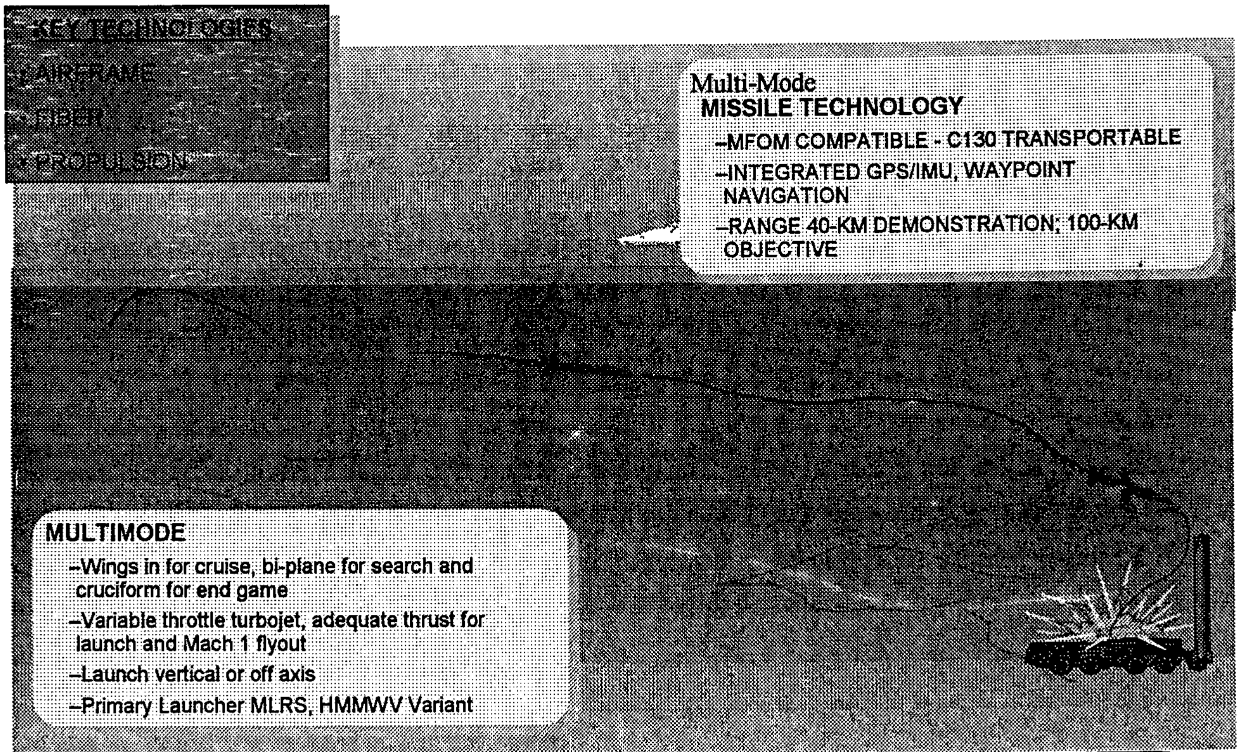
In summary, this paper has presented the control system development, design, and results of a 6DOF simulation, of the MAT missile. The missile models include the aerodynamics, engine, engine controller, velocity controller, actuator, gunner station, and initialization routines. The control system was simulated for the demonstration scenario at Eglin AFB. The MAT simulation shows that the guidance and autopilot functions are indeed achieved with respect to the objectives. Repeatability tests were successfully performed to insure proper variable initialization. Comparison tests between the results of the truth models and realistic models proved to be a valuable debugging tool. Verification and confidence in the simulation was established via Monte-Carlo testing and comparison with expected results. To increase both the MAT simulation's fidelity and capability, the following recommendations for future work are proposed:

- incorporate the latest physical data including missile and fuel mass, CG location, and aero reference
- incorporate addition engine data tables that reflect the current turbojet hardware performance
- modification/validation and testing of the engine controller / fuel pump / turbojet engine model with physical test data
- improve the velocity control algorithm to achieve the mission objectives when sensor failure occurs
- perform extensive Monte-Carlo simulation to optimize missile performance against various threat suites and develop a tactical design and simulation
- perform extensive Monte-Carlo simulation of the demonstration flight to optimize the performance characteristics.

REFERENCES

1. U.S. Army Missile Command, Redstone Arsenal, Alabama, *IOE Version of FOG-M Digital Simulation,* Nichols Research Corporation, March 1989, Report No. NRC-TR-89-061.
2. Walls, Justin L.S., *LONGFOG Aerodynamic / Actuator Modeling and Simulation Development*, Nichols Research Corporation, August 1995, Contract No. DAAH01-91-D-R007.
3. Blakelock, John, Automatic Control of Aircraft and Missiles, John Wiley & Sons, New York, 1965.
4. Garner, Russell, *Development of the ADKEM Transfer Function Through Modal Analysis Techniques,* U.S. Army Missile Command, Redstone Arsenal, Alabama, March 1994, Report Number RD-ST-94-8.

Figure 1. Multimode Airframe Technology (MAT) Concept



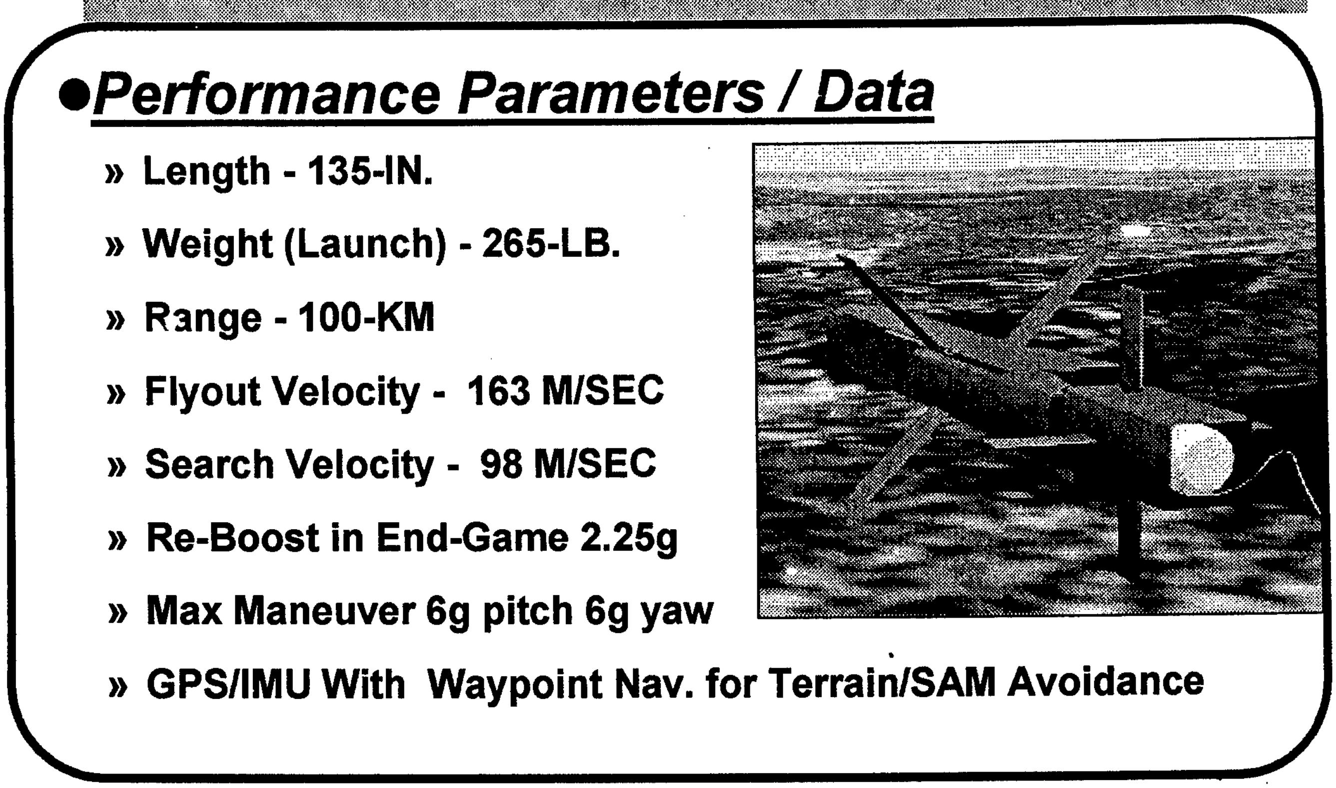


Figure 2. MAT Missile Performance Characteristics

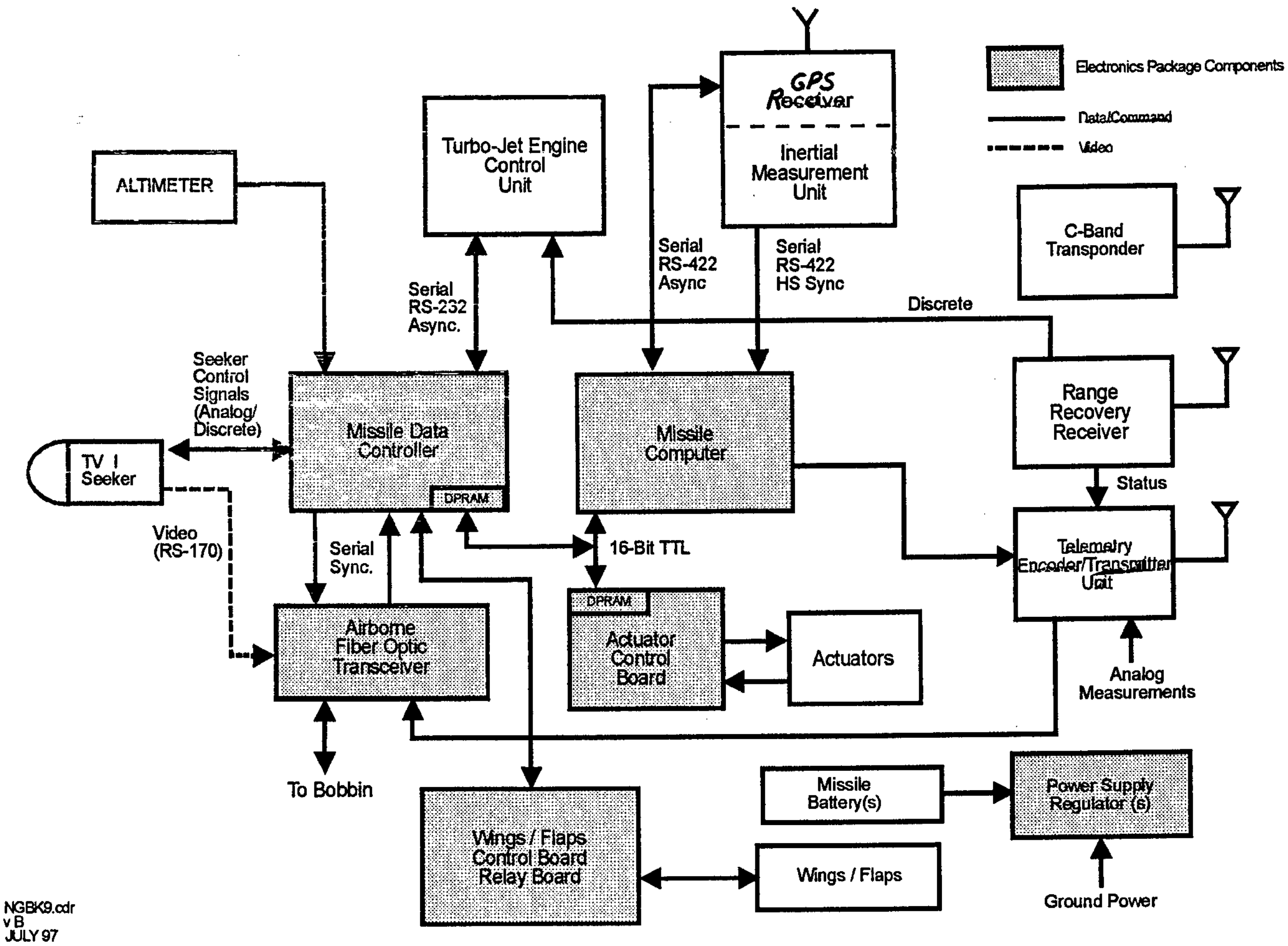
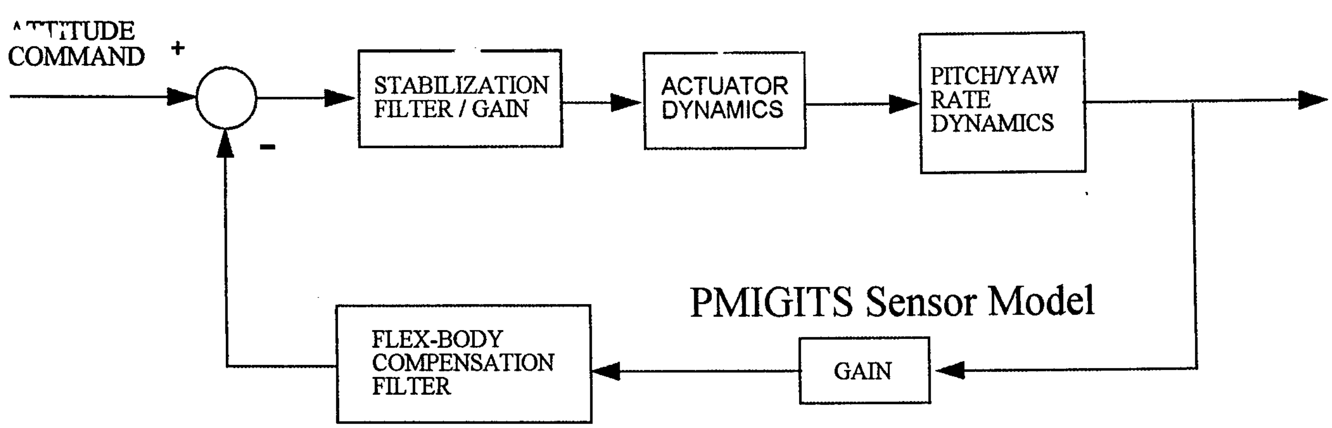


Figure 3. Missile Subsystems

Inner Rate Loop



Loop must be stabilized for all aero configurations and flight conditions

Figure 4. Inner Rate Loop

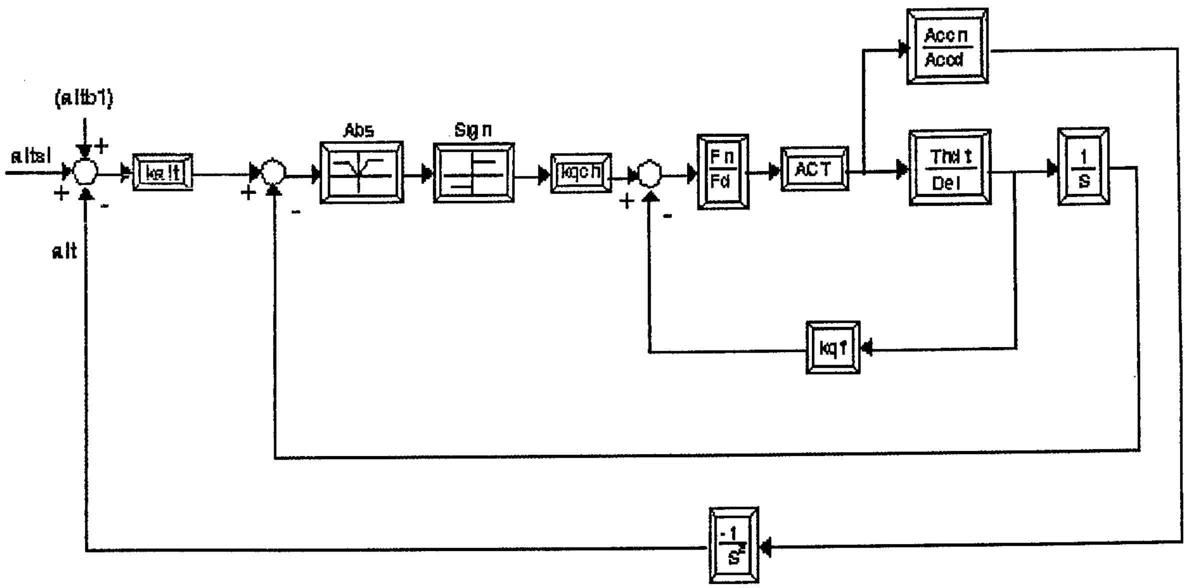


Figure 5. Altitude Hold Loop
Fn / Fd – Stabilization filter

ACT – Actuator transfer function

KQCH - Gain that converts attitude error to pitch rate gain.

KALT - Altitude stabilization gain

ALTB1 - A bias term used to adjust the altitude setpoint.

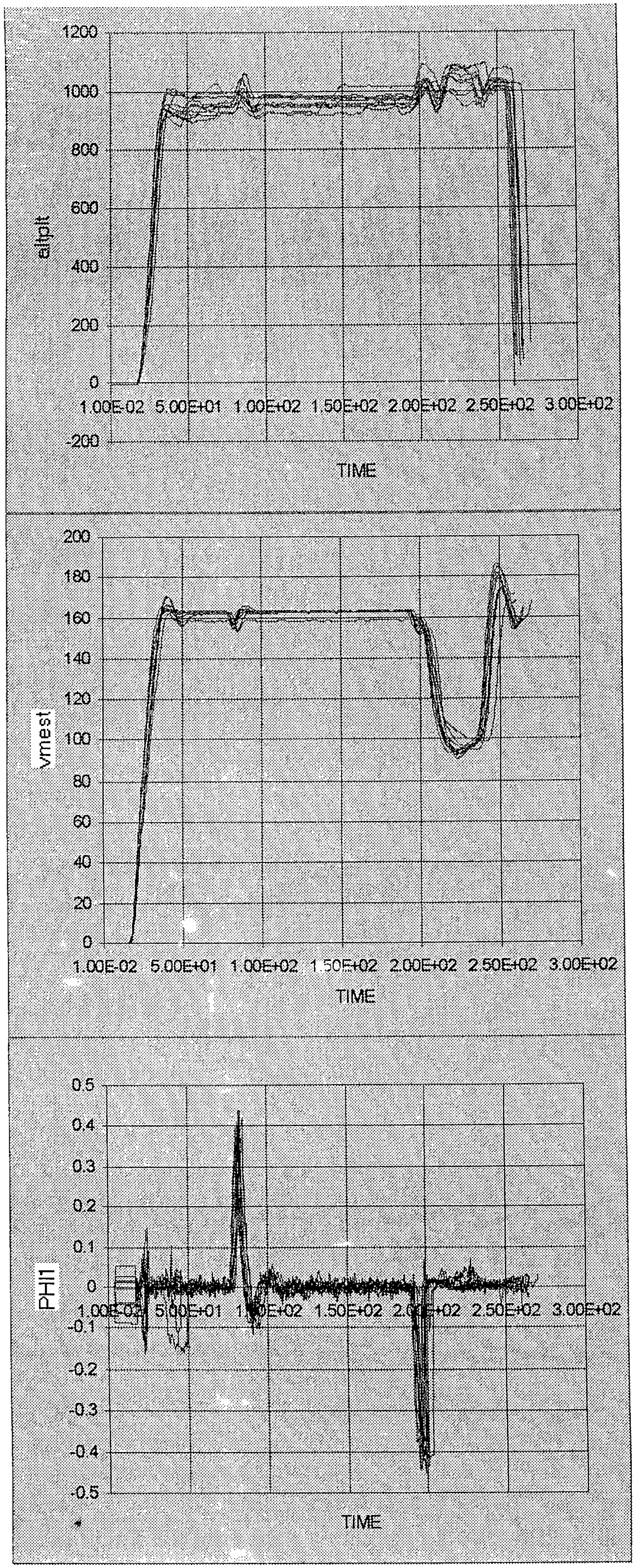
Pitch rate per fin deflection transfer function (Thdt/Del) :

$$\frac{\theta}{\delta}=\frac{M_d * s + M_a * Z\ d - M_d * Z_a}{s^2 + (Z_a + Mq)s + M_a * vm + Mq * Z_a}$$

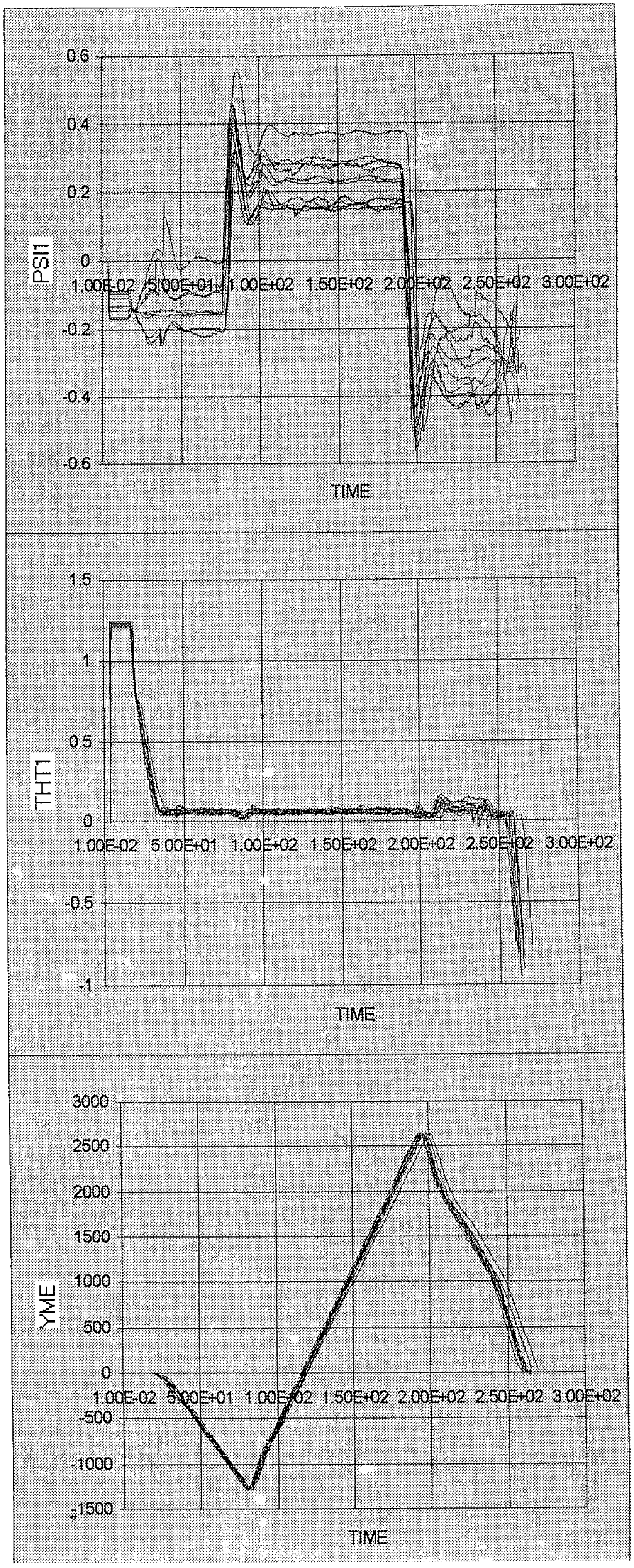
Body Accel. per fin deflection transfer function (Accn/Accd) :

$$\frac{\eta}{\delta}=\frac{Z_\delta * s^2 - Z_\delta * M_q * s - vm * (-M_\alpha * Z_\delta + M_\delta * Z_\alpha)}{s^2 - (Z_\alpha + M_q) * s + M_\alpha * vm + M_q * Z_\alpha}$$

Figures 6 - 8.


1200
1000
800
600
400
200
0
-200
altplt
1.00E-02 5.00E+01 1.00E+02 1.50E+02 2.00E+02 2.50E+02 3.00E+02
TIME
200
180
160
140
120
100
80
60
40
20
0
vmest
1.00E-02 5.00E+01 1.00E+02 1.50E+02 2.00E+02 2.50E+02 3.00E+02
TIME
0.5
0.4
0.3
0.2
0.1
0
-0.1
-0.2
-0.3
-0.4
-0.5
PHI1
1.00E-02 5.00E+01 1.00E+02 1.50E+02 2.00E+02 2.50E+02 3.00E+02
TIME

Figures 9-11.



0.6
0.4
0.2
0
-0.2
-0.4
-0.6
PSI1
1.00E-02
5.00E+01
1.00E+02
1.50E+02
2.00E+02
2.50E+02
3.00E+02
TIME
1.5
1
0.5
0
-0.5
-1
THT1
1.00E-02
5.00E+01
1.00E+02
1.50E+02
2.00E+02
2.50E+02
3.00E+02
TIME
3000
2500
2000
1500
1000
500
0
-500
-1000
-1500
YME
1.00E-02
5.00E+01
1.00E+02
1.50E+02
2.00E+02
2.50E+02
3.00E+02
TIME

AIAA 1998 Missile Sciences Conference

Title of Document *Guidance and Control Development of the Multimode Airframe Technology (MAT) Missile*

PLEASE CHECK THE APPROPRIATE BLOCK BELOW:

☐ ___1___ copies are being forwarded. Indicate whether Statement A, B, C, D, E, F, or X applies.

☒ DISTRIBUTION STATEMENT A:
APPROVED FOR PUBLIC RELEASE: DISTRIBUTION IS UNLIMITED

☐ DISTRIBUTION STATEMENT B:
DISTRIBUTION AUTHORIZED TO U.S. GOVERNMENT AGENCIES ONLY; (Indicate Reason and Date). OTHER REQUESTS FOR THIS DOCUMENT SHALL BE REFERRED TO (Indicate Controlling DoD Office).

☐ DISTRIBUTION STATEMENT C:
DISTRIBUTION AUTHORIZED TO U.S. GOVERNMENT AGENCIES AND THEIR CONTRACTORS; (Indicate Reason and Date). OTHER REQUESTS FOR THIS DOCUMENT SHALL BE REFERRED TO (Indicate Controlling DoD Office).

☐ DISTRIBUTION STATEMENT D:
DISTRIBUTION AUTHORIZED TO DoD AND U.S. DoD CONTRACTORS ONLY; (Indicate Reason and Date). OTHER REQUESTS SHALL BE REFERRED TO (Indicate Controlling DoD Office).

☐ DISTRIBUTION STATEMENT E:
DISTRIBUTION AUTHORIZED TO DoD COMPONENTS ONLY; (Indicate Reason and Date). OTHER REQUESTS SHALL BE REFERRED TO (Indicate Controlling DoD Office).

☐ DISTRIBUTION STATEMENT F:
FURTHER DISSEMINATION ONLY AS DIRECTED BY (Indicate Controlling DoD Office and Date) or HIGHER DoD AUTHORITY.

☐ DISTRIBUTION STATEMENT X:
DISTRIBUTION AUTHORIZED TO U.S. GOVERNMENT AGENCIES AND PRIVATE INDIVIDUALS OR ENTERPRISES ELIGIBLE TO OBTAIN EXPORT-CONTROLLED TECHNICAL DATA IN ACCORDANCE WITH DoD DIRECTIVE 5230.25. WITHHOLDING OF UNCLASSIFIED TECHNICAL DATA FROM PUBLIC DISCLOSURE. 6 Nov 1984 (Indicate date of determination). CONTROLLING DoD OFFICE IS (Indicate Controlling DoD Office).

☐ This document was previously forwarded to DTIC on __________ (date) and the AD number is ______________.

☐ In accordance with provisions of DoD instructions, the document requested is not supplied because:

☐ It will be published at a later date. (Enter approximate date, if known).

☐ Other. (Give Reason)

DoD Directive 5230.24, "Distribution Statements on Technical Documents," 18 Mar 87, contains seven distribution statements, as described briefly above. Technical Documents must be assigned distribution statements.

____
Print or Type Name

____
Authorized Signature/Date

____
Telephone Number