سایت اختصاصی

مهندسی کنترل

AIAA Guidance, Navigation, and Control Conference and Exhibit
21 - 24 August 2006, Keystone, Colorado

Guidance and Navigation System Design

For a Ship Self Defense Missile

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This paper describes the design and performance evaluation of a ship self defense missile. The missile airframe characteristics and the guidance and navigation system requirements needed to accomplish the mission are discussed. A missile guidance law design that achieves the objectives of small miss distance and a required terminal impact angle is demonstrated. The IMU and navigation system requirements necessary to guarantee a high probability of target acquisition by a strapdown, narrow field-of-view, infrared seeker are determined. It is shown that an accurate GPS-aided IMU system on the ship can transfer alignment and calibrate out navigation initialization errors for the missile before launch. As a result, a low cost Micro Electro Mechanical System (MEMS) IMU from the Army Common Guidance development program is shown to be adequate for the missile. The study determines that the more accurate DIGNU-3 IMU can meet all missile requirements even under conservative navigation error assumptions. If required, GPS in-flight navigation updates could be used to allow successful missile operation with much less accurate IMUs.

I. Introduction

Self defense of Naval ships against small surface craft is a difficult problem for a number of reasons. These include the wide variety of threat platforms, weapons, and tactics that may be encountered, and (in most situations) very restrictive rules-of-engagement coupled with a confined battlespace. In addition, current shipboard sensors do not provide fire control quality tracks of individual targets in closely spaced groups, and do not provide needed target identification information at ranges beyond a few nautical miles. Existing shipboard sensors – principally radar, since electro-optical sensors are deployed in limited numbers – generally provide good range data and bearing or azimuthal data of varying quality. In addition, such sensors can track a group of targets to some level of accuracy (by tracking the "centroid" or "leading edge" of the group).

The high-level challenges in ship self defense include a number of technologies in the weapon, sensor and command and control areas. Sensor system requirements include shipboard surveillance, detection, identification, tracking, and damage assessment capability adequate to support a desired maximum engagement range. Since shipboard sensor capability will likely not be substantially improved without the addition of new and expensive sensors, the technical challenge is to find system solutions that are compatible with current sensor capabilities.

In the present paper, the focus will be on design of a self defense missile to prosecute engagement of a swarm of surface targets, and it will be tacitly assumed that the sensor suites to support such an engagement are available. The paper will describe the guidance and navigation system features that allow a guided missile with strapdown sensors to achieve the small requisite miss distances and desired impact angles at intercept over the full battlespace.

The concept uses an interceptor with a midcourse phase employing inertial guidance, and a terminal phase with a strapdown infrared (IR) seeker based guidance. The goal of the missile design was to define a small, low-cost, precision guided weapon with adequate range and lethality for a wide spectrum of potential threats. The result of initial trade studies focused on a small guided rocket, 3 inches in diameter, and weighing about 40 lbs. The missile guidance and navigation system was required to achieve a terminal accuracy of 1-2 meter CEP, using a body-fixed,

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low-cost seeker and a low-cost Inertial Measuring Unit (IMU). The interceptor is tail-controlled and has mid-body wings designed to yield three g's of lateral maneuverability at an angle of attack of 7 degrees. This value was selected to allow the missile to engage a 1 g surface target maneuver while keeping the target in the body fixed seeker's field of view (FOV).

The rocket motor is a boost-sustain configuration designed to achieve a nearly constant speed of Mach 0.8 out to its maximum range. The average speed was selected because the system must engage the target swarm in a series of salvos. The number of simultaneous interceptors in flight is limited to ensure that only one interceptor engages each target Because of the large number of targets, a series of salvos are required to engage the entire raid. This average speed allowed a sufficient number of salvos in the available engagement timeline without a significant interceptor weight increase.

During terminal homing, the missile's IR sensor is used for tracking of targets within the sensor FOV. The concept requires terminal guidance algorithms that manage the seeker FOV pointing, as well as perform the traditional guidance functions. In addition, achieving favorable lethality against the expected targets using a small sized warhead required the use of trajectory shaping to control the aspect angle relative to the target at intercept. Trajectory shaping was also desired to accommodate low cloud ceilings and to minimize solar glint effects on the IR seeker.

The baseline missile design is illustrated in Figure 1.

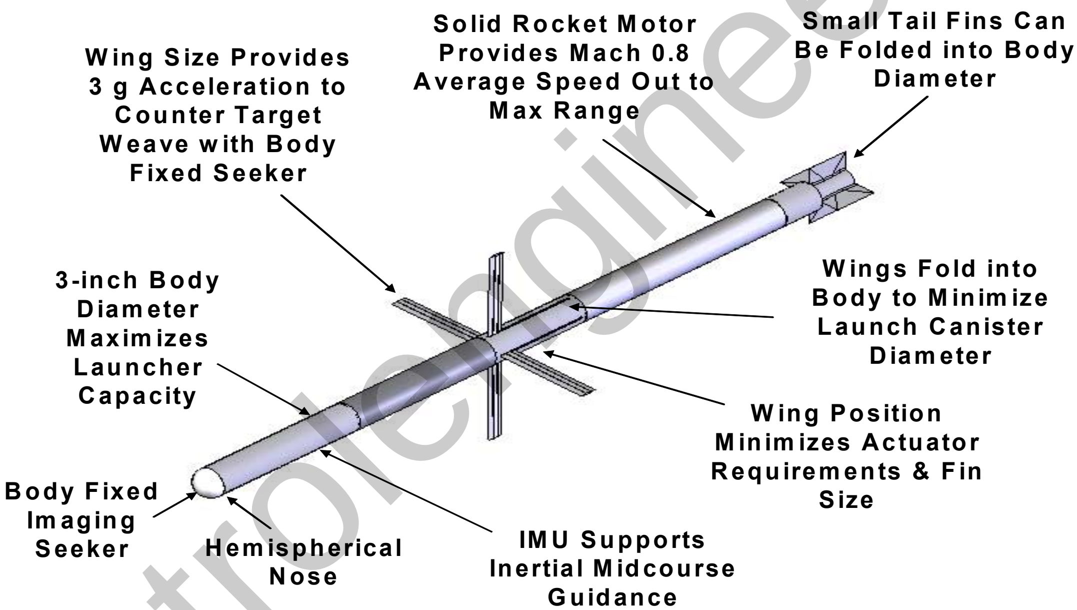

Figure 1. Ship Self Defense Missile Baseline Design

II. Guidance System Design and Performance

The guidance scheme is designed to allow the engagement of an unresolved swarm of targets and to minimize the possibility of multiple interceptors attacking the same target. Expected ship-based sensor performance is not adequate to allow individual tracks of the closely spaced targets. In the swarm defense guidance approach, the interceptors will be launched in a series of salvos. The number of missiles in each salvo is determined by the desire to minimize multiple engagements of the same target. The interceptors contain a downlink that can be used during the mission to provide the ship fire control system with additional information beyond that provided by the ship sensors. The snapshots from the IR seekers on the missiles of the incoming swarm will be downlinked to the ship to provide a raid count and additional information on the extent of the swarm dimensions. However, analysis has indicated that the gridlock between the interceptors is insufficient to allow a full correlation of the multiple images and fusion into a common view of the swarm components for targeting purposes.

The raid will be engaged in the initial salvo as a single entity with a width and depth across the sea surface. The missiles will be inertially guided to aimpoints distributed across the width and depth of the swarm. To prevent multiple missiles from selecting and homing on the same target during the terminal guidance phase, each missile is directed into a basket compatible with its FOV and centered about the inertial aimpoint. The basket is sized based on the estimated 3σ uncertainties in the missile navigation solution. The number of simultaneous interceptors that may be launched in a single salvo is limited by the number of projected uncertainty baskets that fit within the swarm's estimated perimeter. This constraint ensures that each interceptor has a unique surface patch within which it may search for targets.

Should more than one target appear in an interceptor's search basket, the target closest to the inertial aimpoint is selected as the target for engagement. Any unengaged targets in the search basket would be dealt with in subsequent salvos. Should no targets appear in the interceptor's search basket, then the search pattern would be increased in a step-wise fashion until a target is found. In this situation, the possibility of two interceptors engaging the same target is considered preferable to allowing an interceptor to go to waste. If the targets have bunched themselves in a non-uniform distribution so that some interceptors' baskets are empty, the expanding basket strategy will help migrate interceptors toward the concentrated regions of the swarm. Conflict between the expanded and nominal search interceptors is mitigated because each selects the target closest to its inertial aimpoint. As successive salvos are launched against the swarm, the interior of the swarm will become more sparse and the expanded search basket logic increasingly important. At some point the sparseness of the swarm would allow the ship system to resolve the raid into a set of smaller swarms or individual targets.

To prevent re-engagement of destroyed but still visible targets, sufficient time delay is added between salvos to allow any destroyed targets to fall out of the FOV of the seeker when looking at the swarm. Thus, only targets with sufficient functionality to keep up with the swarm will be engaged. To prevent enemy tactics from exploiting this logic, any secondary swarms created by a purposeful slowdown that retain sufficient functionality to attack the ship would be tracked by the ship and classified as their own independent swarm.

Design requirements placed on the seeker look angle at acquisition and on the missile-target aspect angle at intercept dictated that trajectory shaping be implemented during the missile's flight. To accomplish this, Generalized Explicit Guidance (GENEX) [1, 2] was adopted as the missile guidance law for the midcourse phase of the trajectory. This guidance law has the ability to simultaneously achieve small miss distance and a specified final missile-target relative orientation. The terminal angle specification may be used to enhance the performance of warheads whose effectiveness is influenced by the terminal encounter geometry. The GENEX guidance law is parameterized in terms of a design coefficient n, which determines the degree of curvature in the trajectory. For the present analysis, a GENEX guidance parameter value of $n = 1$ was used, although this parameter will be optimized in future work. Further application of the GENEX technique is provided in [3].

The guidance analysis first examined the ability to effectively engage a threat at the minimum range goal of 800 m while achieving a required 30 degree impact angle with the dual phase guidance approach (inertial midcourse to terminal). A low speed surface target was assumed, and system errors were initially neglected. The transition from midcourse to terminal homing was made when the missile was 960 meters away from the target. This range was determined assuming a 1 sec delay for target acquisition and 10 missile time constants of terminal homing with a 0.3 sec autopilot time constant. When the missile transitioned from midcourse to terminal homing, it was required that the look angle to the target be within the IR seeker FOV of $\pm$ 7 degrees.

The minimum range for intercept of a stationary target using the GENEX guidance routine was found to be 2.2 km with a time of flight of 14.5 seconds. The launch angle was 85 degrees and the missile terminal angle at intercept was 30 degrees as required. Figure 2 shows the trajectory of the missile and its velocity history for the minimum range engagement. In Figure 3, the missile's commanded normal acceleration and angle of attack are shown versus time. The line-of-sight (LOS) angle to the target is illustrated in Figure 4, and demonstrates that at the transition to terminal homing, the target lies within the ± 7 deg seeker FOV.

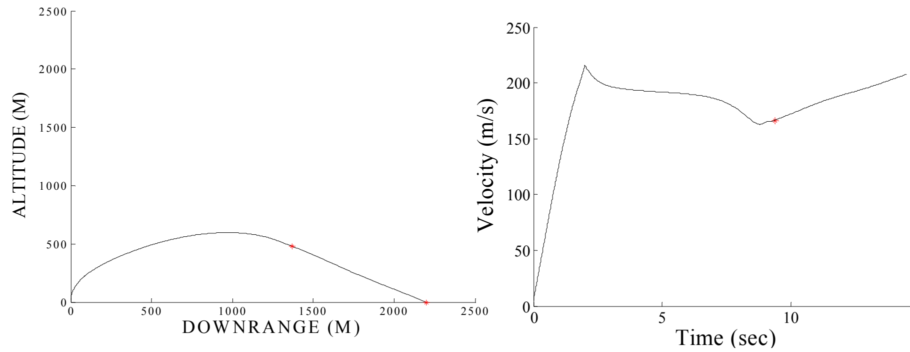

Figure 2. Minimum Range Missile Trajectory: Position and Velocity vs Time

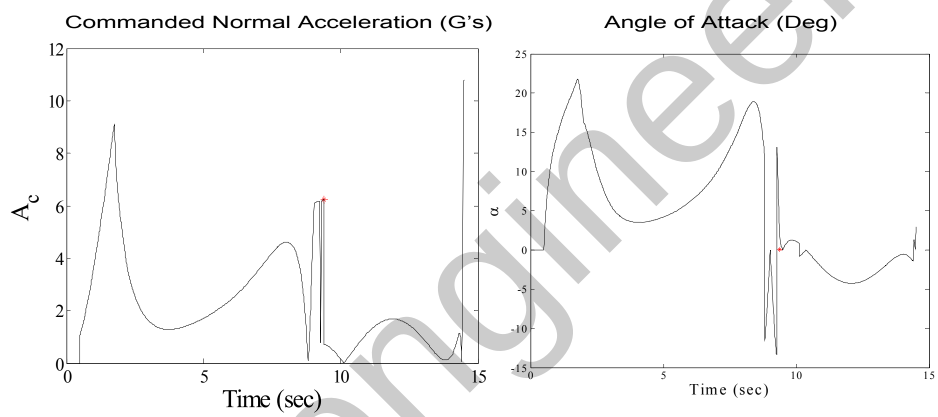

Figure 3. Minimum Range Missile Trajectory: Commanded Acceleration and Angle of Attack vs Time

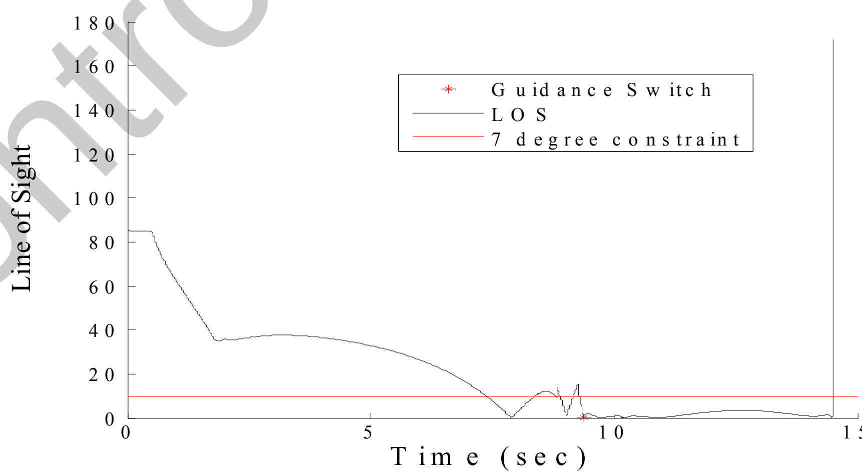

Figure 4. Line of Sight Angle vs. Time for Minimum Range Engagement

The maximum effective range using GENEX guidance was determined to be 17 km for a slowly moving surface target. Figure 5 shows the flight trajectory and velocity history of the missile for this maximum range case. Figure 6 provides the commanded normal acceleration and angle-of-attack versus time. For the maximum range trajectory, the LOS angle satisfied the FOV requirement at acquisition, as shown in Figure 7. During the terminal guidance phase (using proportional navigation), the look angle was just barely contained within the FOV, and will require further optimization. At intermediate ranges, GENEX guidance was able to easily satisfy both the acquisition angle and the terminal impact angle requirements.

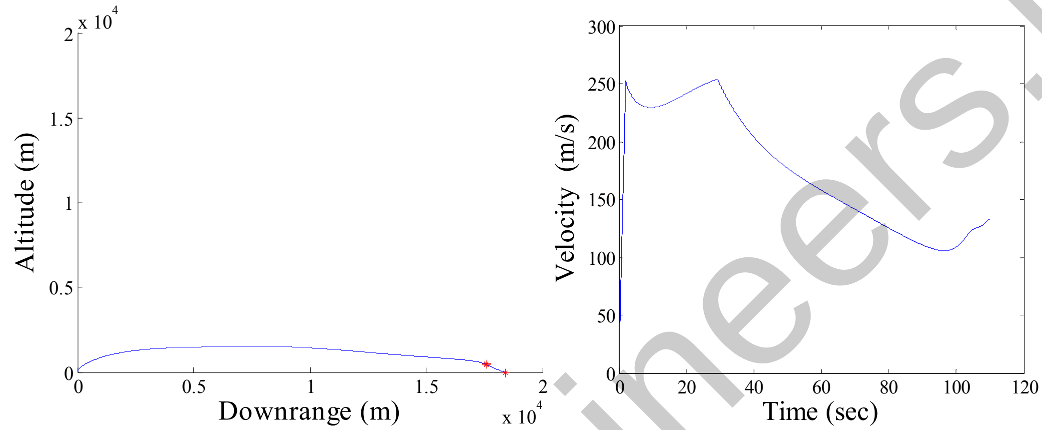

Figure 5. Maximum Range Missile Trajectory: Position and Velocity vs Time.

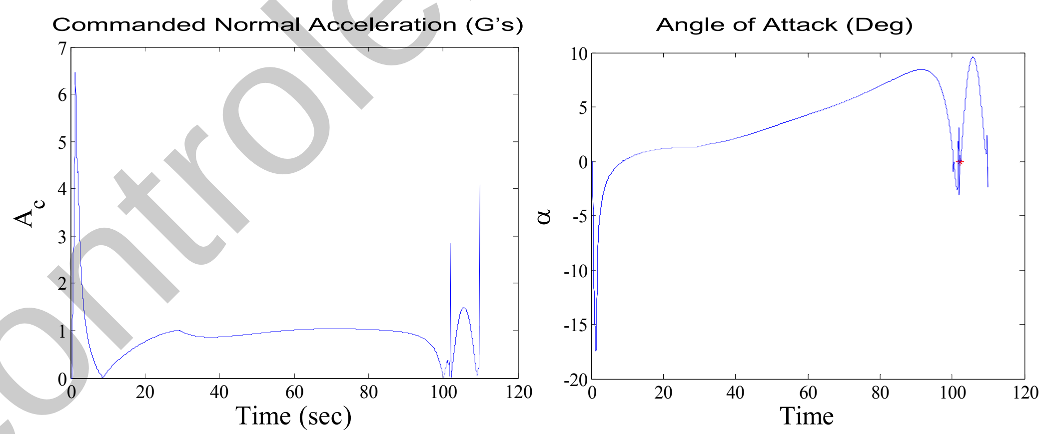

Figure 6. Maximum Range Missile Trajectory: Commanded Acceleration and Angle of Attack vs Time.

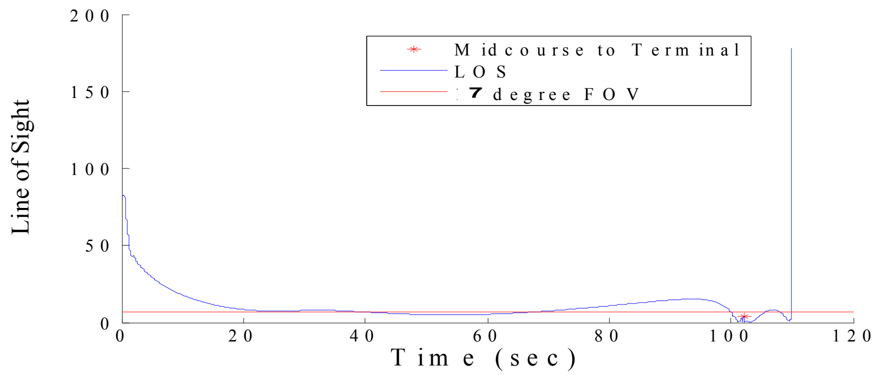

Figure 7. Line-of-Sight Angle vs Time for Maximum Range Engagement.

III. Navigation System Design & Performance

In early studies, the ability to perform pre-launch alignment and calibration of the IMU in the missile was evaluated. The approach assumed, first, that a high quality GPS/IMU system was available on the ship's missile launcher and that it could be accurately calibrated while underway using GPS measurements combined with intentional ship maneuvers to aid the alignment process. Secondly, it was assumed that the accurately aligned master IMU system could transfer its initialization information to the IMU on the missile, using a technique similar to that described in [4]. The assumed result was that the missile IMU inherited the accuracy of the master IMU with respect to initial position, velocity and attitude errors. Preliminary studies using simplified models indicated that this pre-launch calibration strategy appeared feasible.

Although well-aligned, the missile IMU still possessed residual IMU errors. A remaining question was whether these residual errors might cause unacceptable growth in the inertial navigation system (INS) errors (position, velocity and attitude) during the missile flyout. If excessive, these errors might hinder seeker acquisition of the target.

To answer this question, a parametric performance study was carried out. Typical missile flyout trajectories (see Figure 8) were used which spanned the anticipated engagement ranges. A detailed (58 state) navigation error covariance model (see Figure 9), based on the methodology in [5], was used to propagate the initial INS errors to the seeker acquisition point. Then, the total alignment and navigation errors were compared to the available seeker FOV to determine if acquisition of the target were possible. As a check on the earlier results determined with an approximate model, the pre-launch errors for the master GPS/IMU system (used to initialize the missile IMU) were re-computed using the more detailed covariance model

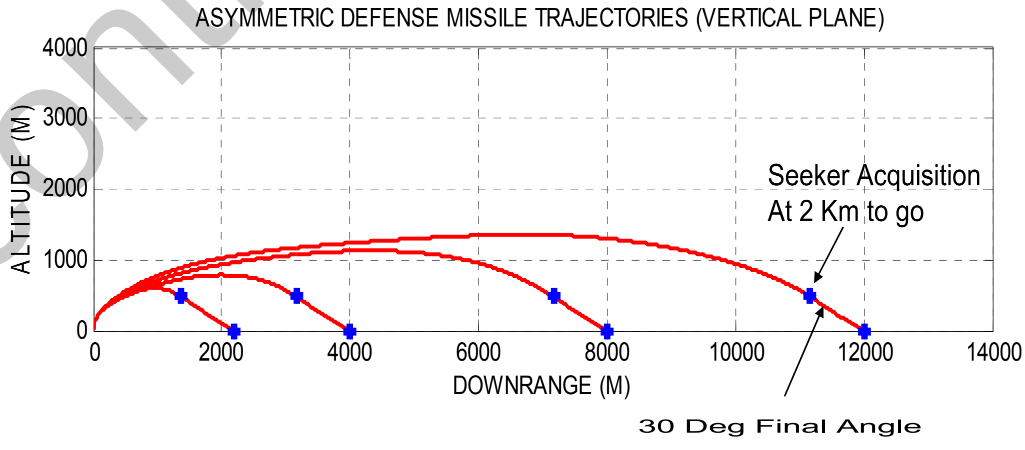

Figure 8. Typical Missile Flyout Trajectories

Error States	Number of States	Coordinate Frame	Truth Model	Filter Model
INS Position	3	ECEF	√	√
INS Velocity	3	ECEF	√	√
INS Attitude	3	ECEF	√	√
Gyro Bias	3	IMU	√	√
Accel Bias	3	IMU	√	√
Receiver Clock Bias & Drift	2	GPS	√	√
Gyro Scale Factor & Misalignments	9	IMU	√	
Gyro G-Sensitive Drift	9	IMU	√	
Gyro G^2-Sensitive Drift	6	IMU	√	
Accel Scale Factor & Misalignments	9	IMU	√	
Pseudo Range Measurement Biases	8	GPS	√	
	58			

Figure 9. Detailed 58-State Covariance Model Used in Navigation Study

The master system assumed an Ashtech μZ dual-frequency GPS receiver and a Litton LN-200 IMU with accuracy specifications shown in Figure 10. During the pre-launch phase, the ship was assumed to be moving on a circular path with a speed of 20 knots and a steady turn rate of 2 deg/sec under Sea State 0 conditions. The values of turn rate and sea state were reduced from those assumed in the earlier study (4 deg/sec and Sea State 5) with no apparent degradation in performance. Some typical histories of the navigation error reduction during the pre-launch alignment phase are shown in Figure 11. The navigation and IMU errors existing in the master system at the end of the pre-launch calibration process are shown in Figure 12. The GPS updating was able to align the launcher IMU system to under 5 mrad (3σ) in azimuth and to under 1 mrad (3σ) in level north-east axes. Position and velocity errors were reduced to very low levels, less than 5 m and 0.02 m/s (3σ), respectively.

- **Ashtech μZ GPS Receiver**
  - 12 channel, dual frequency receiver
  - Pseudo range measurement accuracy = 50 cm (1σ)
  - Delta range measurement accuracy = .01 m/s (1σ)
  - GPS Updates taken every 1 sec
- **Litton LN-200 IMU**
  - Gyro drift = .35 deg/hr (1σ)
  - Accelerometer Bias = .05 mg (1σ)
  - Gyro random walk = .07 deg/√hr
  - Accel random walk = 50 μg/√Hz

Figure 10. Launcher GPS & IMU Characteristics

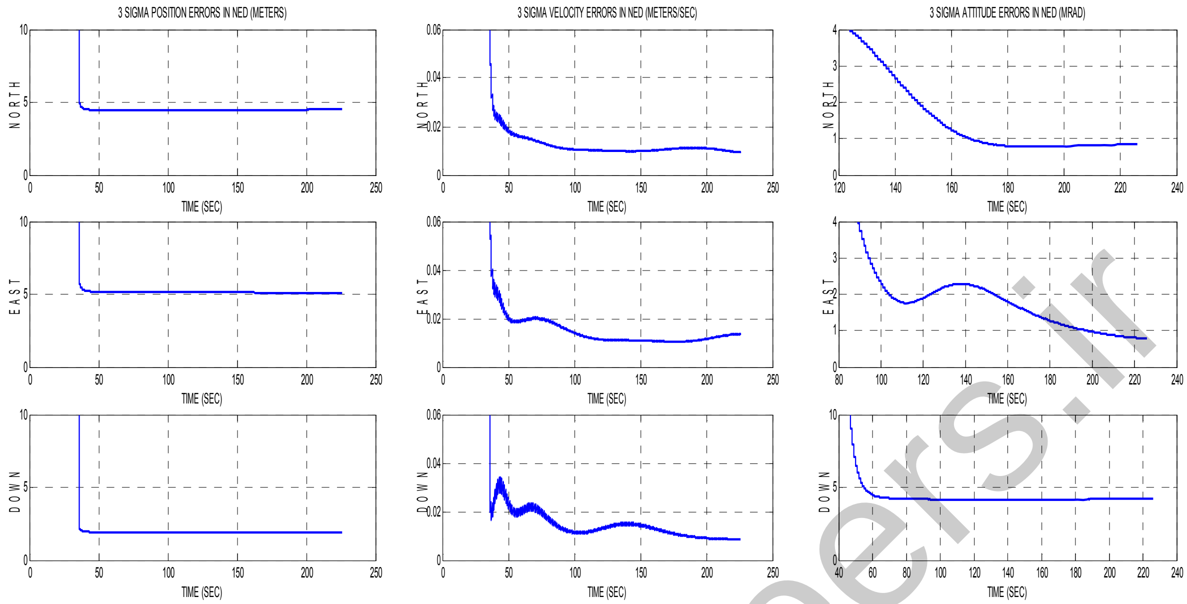

Figure 11. 3σ Position, Velocity and Attitude Error Histories During Pre-Launch Alignment Phase

Final 3σ Errors After GPS IMU Calibration

Error Source	Units	3σ Error North	3σ Error East	3σ Error Down
Position	m	4.503	5.103	1.887
Velocity	m/s	0.009	0.013	0.008
Attitude	mrad	0.845	0.782	4.223
Gyro Bias	deg/hr	1.050	1.050	0.995
Accel Bias	mg	0.111	0.105	0.022

Figure 12. Errors in Master GPS/INS at End of Pre-launch Calibration Phase

A primary objective of the navigation system analysis was to select an IMU of low cost but adequate to meet mission requirements. The candidate IMUs assumed for the missile system were based on the Micro Electro Mechanical Systems (MEMS) being developed under the Army's Common Guidance program [6, 7]. The Deeply Integrated Guidance & Navigation Unit (DIGNU) is an advanced GPS/INS system based on the Honeywell HG1910 series IMU and the Rockwell Collins NavStrike GPS receiver. The DIGNU IMUs are being developed in time-phased realizations known as DIGNU-1 through DIGNU-3, with each successive version having an increased accuracy and a reduced packaging volume (see Figure 13). The assumed error characteristics of the DIGNU IMU series are shown in Figure 14. DIGNU-3 has been taken as the baseline for the self defense missile system. However, because the availability of DIGNU-3 in the expected missile deployment timeframe is not certain, the current study also considered excursions involving the lower accuracy DIGNU-1 and DIGNU-2.

Phase 1	Phase 2	Phase 3
		
Features: • < 8.0 in^3 IMU (2.69" dia x 1.3" tall)	***Features:*** • < 4 in^3 IMU (2.0" dia x 1.27" tall)	***Features:*** • < 2 in^3 IMU (2.0" dia x 0.64" tall)

Figure 13. IMU Candidates from Army Common Guidance Program

3-σ Error Sources	Units	DIGNU-1	DIGNU-2	DIGNU-3
Gyro bias	deg/hr	225	60	3
Gyro scale factor	ppm	2250	1050	450
Gyro misalignment	mrad	3.6	2.1	0.6
Gyro g-drift	deg/hr/g	30	3	1.5
Gyro g^2-drift	deg/hr/g^2	0.9	0.3	0.15
Gyro random walk	deg/√hr	1.5	0.9	0.36
Accel bias	mg	27	12	3
Accel scale factor	ppm	4500	2100	900
Accel misalignment	mrad	3.6	1.8	0.6
Accel random walk	m/s/√hr	0.99	0.36	0.21

Figure 14. Assumed Errors for Missile IMU Candidates

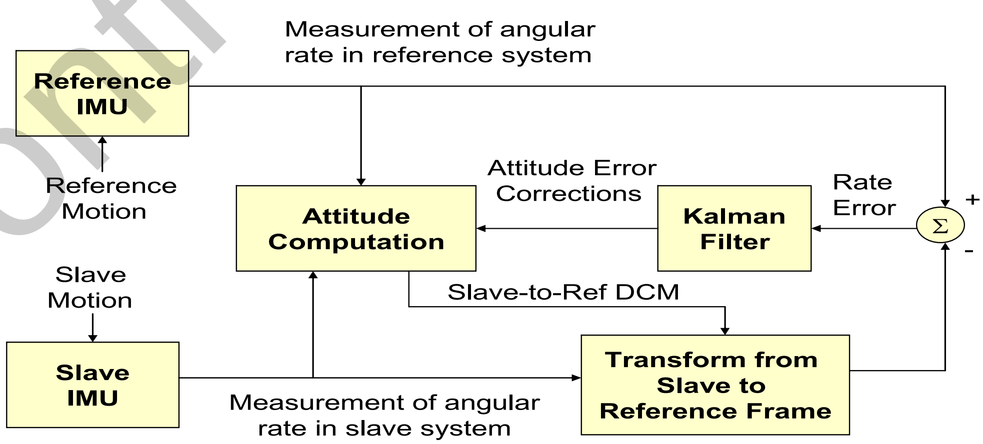

Figure 15. IMU Angular Rate Matching Transfer Alignment Scheme

In order to accurately align the missile IMU and calibrate its gyro and accelerometer errors, it is necessary that an IMU-to-IMU data matching procedure be performed between the master and slave IMUs. This process is conceptually illustrated in Figure 15 [4]. To accomplish this calibration procedure, the following steps must be performed prior to launch:

- Power must be applied to the missile and its flight software activated.
- A data interface must exist so the master IMU system can provide its data to the missile system.
- The missile must accept this data and run its internal Kalman filter to estimate and remove attitude errors and IMU biases.
- Only static (but not time-dependant) biases can be removed in this way from the missile IMU.
- While this process is being done, the ship must perform some deliberate motions to aid the error identification.
- When complete, the data interface must be disconnected and the missile power source removed, if not ready for launch.
- This process must be repeated if a significant time passes before the missile is launched. This is because of growth in the missile alignment and IMU errors due to subsequent ship motion.

Because the steps above will place an additional burden on the ship and missile pre-launch operations, the navigation study was performed parametrically, assuming several sets of initial conditions for the missile IMU and INS. These are represented by Cases 1 through 3 described below:

- **Case 1** – Assumes more conservative initial conditions than those shown in Figure 12. These allow for the possibility of interrupted GPS coverage at sea, the fact that there are unmodeled errors in the analysis model, and that the pre-launch IMU-to-IMU calibration process may at times be impractical due to operational constraints. The assumed missile initial errors for this case were:
  - » 30 m (3 σ) position errors
  - » 0.3 m/s (3σ) velocity errors
  - » 6 mrad (3σ) attitude error
  - » full DIGNU-2 and DIGNU-3 IMU errors (Figure 14)
- **Case 2** – Assumes that position, velocity, and attitude errors consistent with Figure 12 can be passed from the launcher GPS/IMU system to the missile IMU before launch. However, no calibration of the missile IMU's gyros and accelerometers takes place, and the IMU errors of Figure 14 apply.
- **Case 3** – Assumes position, velocity and attitude errors from Figure 12 **PLUS** the launcher system is able to remove the static gyro and accelerometer biases from the missile IMU prior to launch (the remaining IMU errors are present).

The sensitivity of the missile flyout navigation errors will be considered for each of these cases.

Figures 16 and 17 show the worst case attitude errors at acquisition for each of the cases, assuming the DIGNU-2 and DIGNU-3 IMUs. In these and subsequent figures, the worst case errors were calculated by performing an eigenvalue decomposition (diagonalization) of the respective 3 x 3 error covariance matrix, and taking 3 times the square root of the maximum eigenvalue as the indicated 3σ error. In these figures, each range point represents a trajectory flown to that intercept range, with the 3σ attitude error at acquisition displayed. The acquisition limit is taken as half the FOV or 7 degrees (122 mrad). For both DIGNU-2 and DIGNU-3, the accumulated flyout attitude errors are well below the acquisition limit for all ranges and for all of the cases.

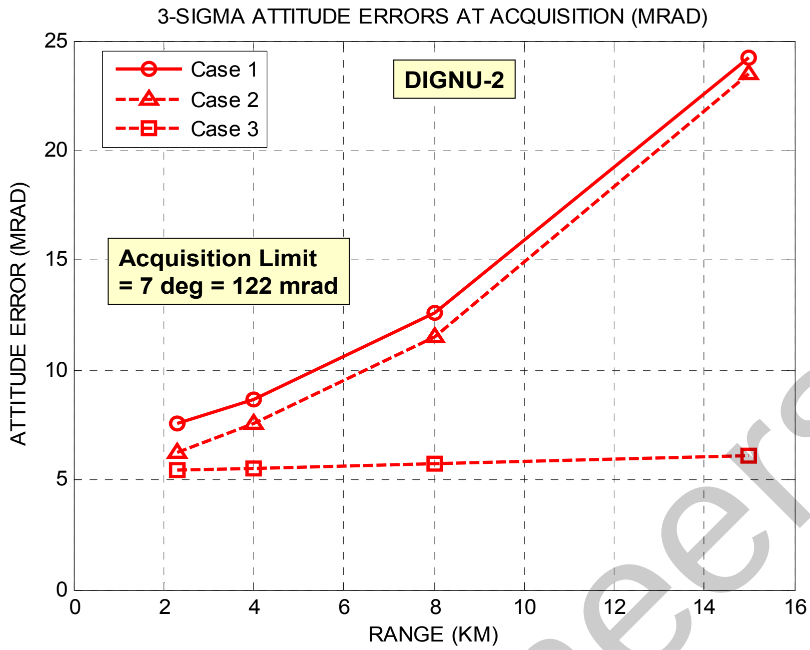

Figure 16. Effect of Initial Conditions on DIGNU-2 Attitude Errors

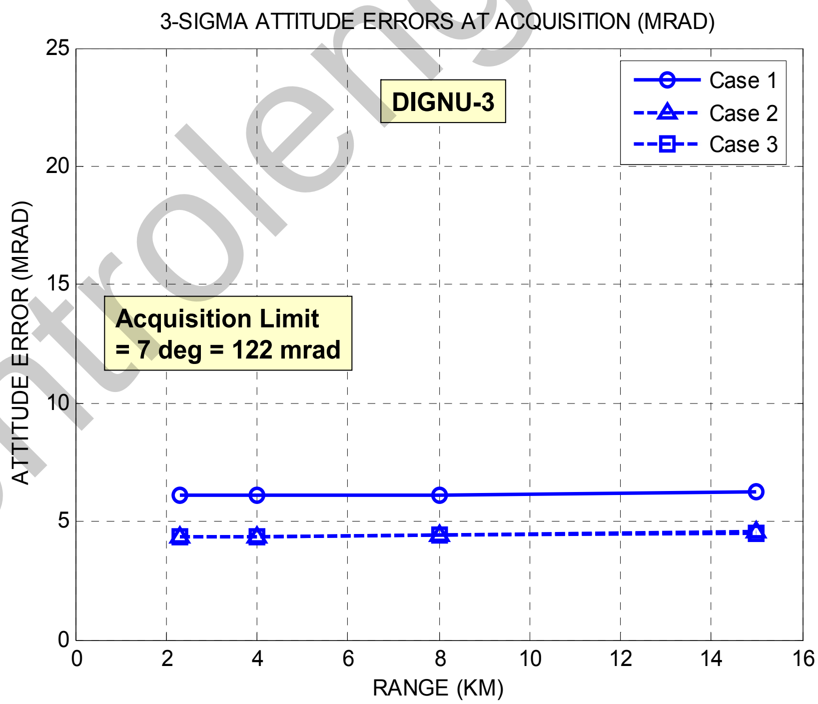

Figure 17. Effect of Initial Conditions on DIGNU-3 Attitude Errors

Similar results are shown in Figures 18 and 19 for the 3σ position error at acquisition. In this case the acquisition limit is given by a nominal acquisition range of 2000 m times the tangent of half the FOV or 245 m. For DIGNU-2, Cases 1 and 2 violate the acquisition limit for ranges slightly greater than 10 km. For DIGNU-3, the position errors are below the limit for all three cases.

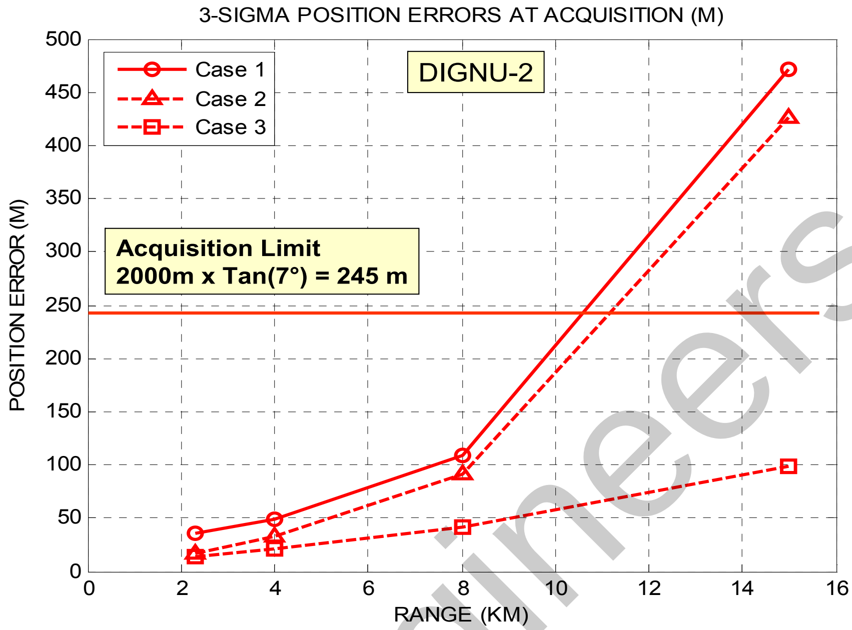

Figure 18. Effect of Initial Conditions on DIGNU-2 Position Errors

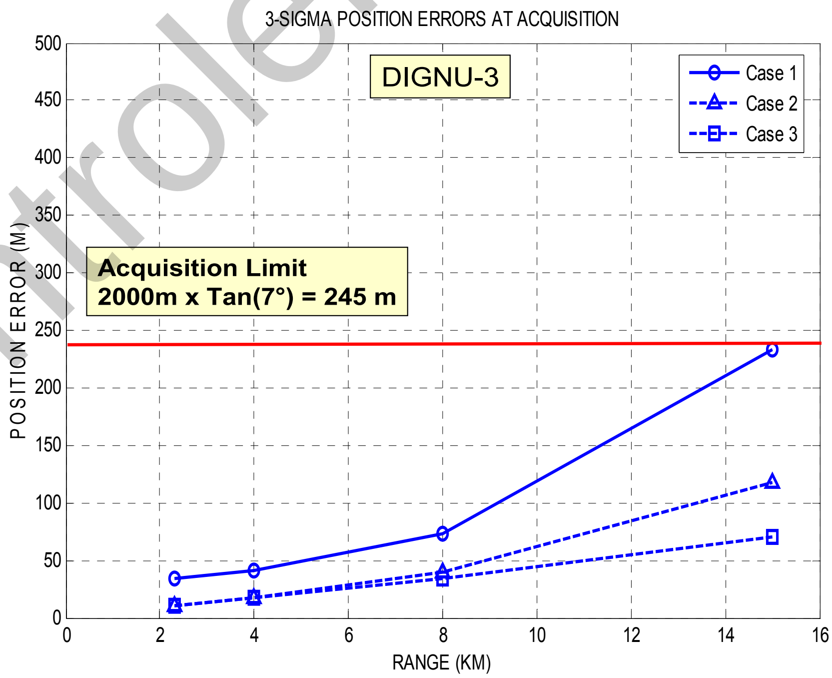

Figure 19. Effect of Initial Conditions on DIGNU-3 Position Errors

In Figures 20 and 21, the 3σ velocity errors at acquisition are shown for the three cases. Here, the acquisition limit on lateral velocity error is determined by multiplying the missile average velocity (Mach 0.8 at sea level, or 272 m/s) by the tangent of half the FOV to obtain 33 m/s. Both DIGNU-2 and DIGNU-3 easily satisfy the acquisition limit for all three cases.

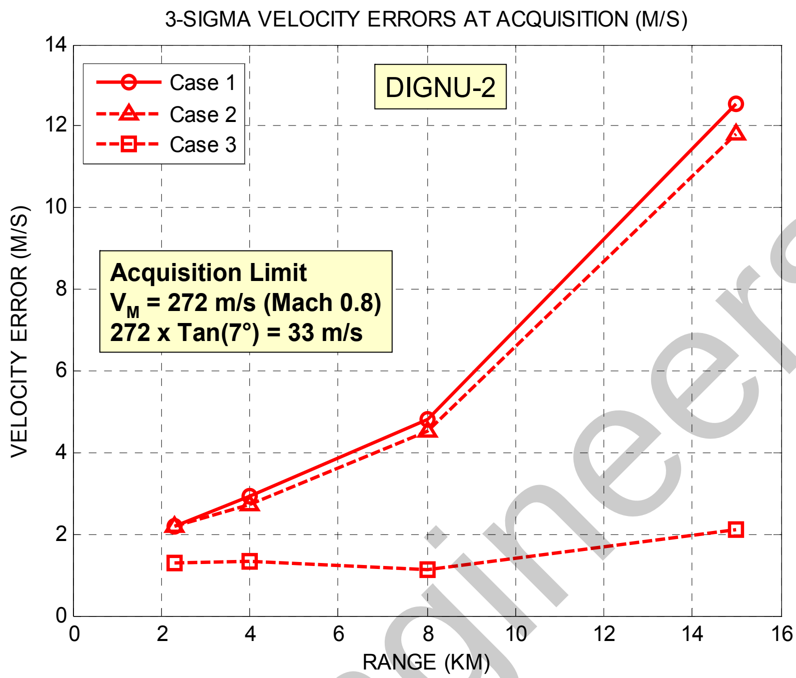

Figure 20. Effect of Initial Conditions on DIGNU-2 Velocity Errors

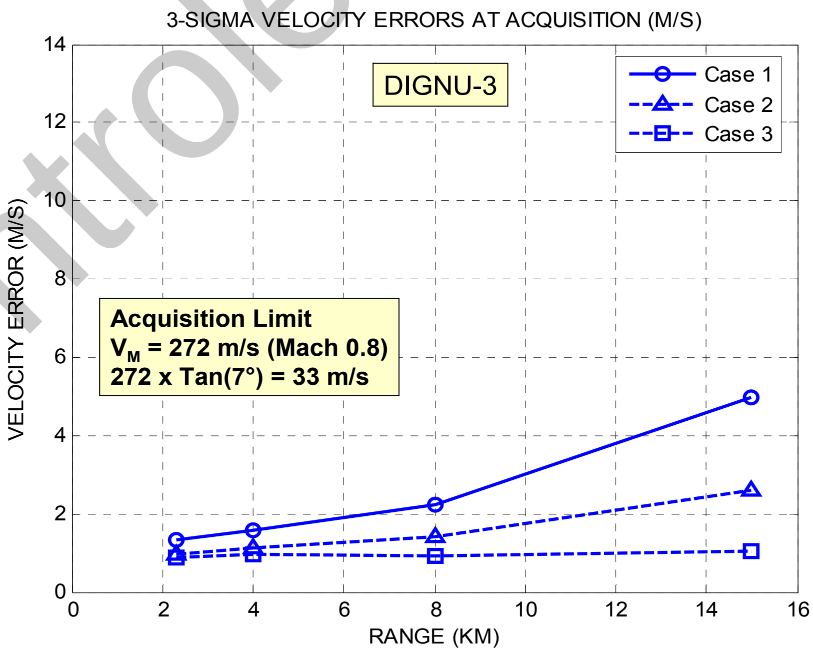

Figure 21. Effect of Initial Conditions on DIGNU-3 Velocity Errors

Figure 22 shows some sample results for the histories of the position, velocity and attitude errors versus time for a 15 km trajectory with DIGNU-3.

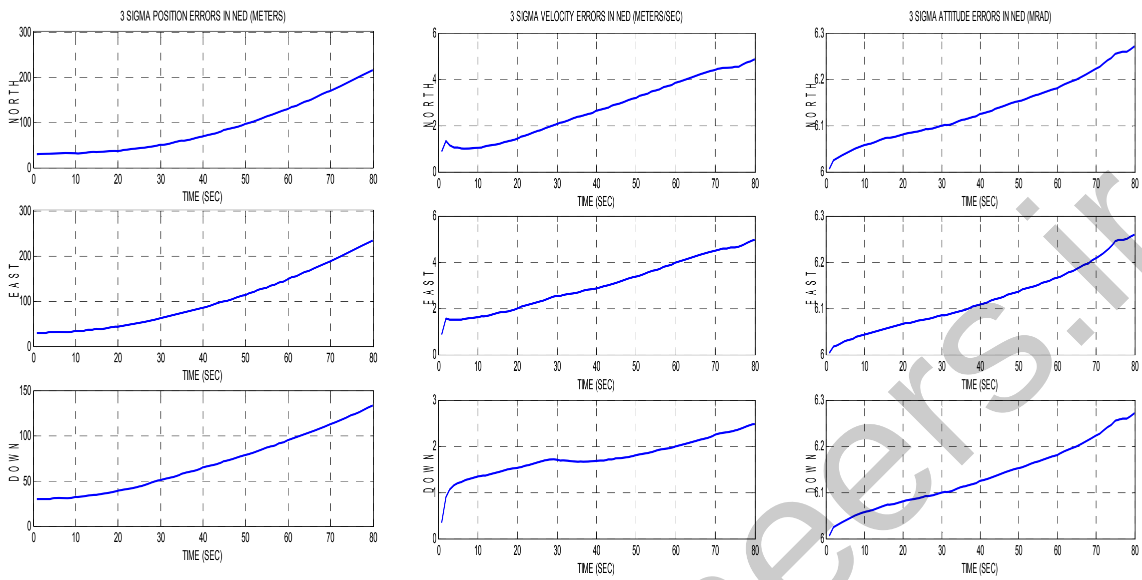

Figure 22. 3σ Position, Velocity and Attitude Errors Versus Time For 15 Km Trajectory with DIGNU-3

Based on the previous results, an excursion study was done to investigate whether GPS aiding could allow use of DIGNU-1 or substantially benefit DIGNU-2 performance. The analysis considered only the 15 km trajectory, and it was assumed (conservatively) that the first GPS updates (position and velocity) occurred near apogee at about 40 sec into the flight. Results are shown in Figures 23 for DIGNU-2 and in Figures 24 for DIGNU-1. After the first GPS update, there is a dramatic reduction in the errors that have built up during the time before apogee. Use of GPS would thus allow the lower grade IMUs (DIGNU-1 or DIGNU-2) to be safely used at the longer ranges.

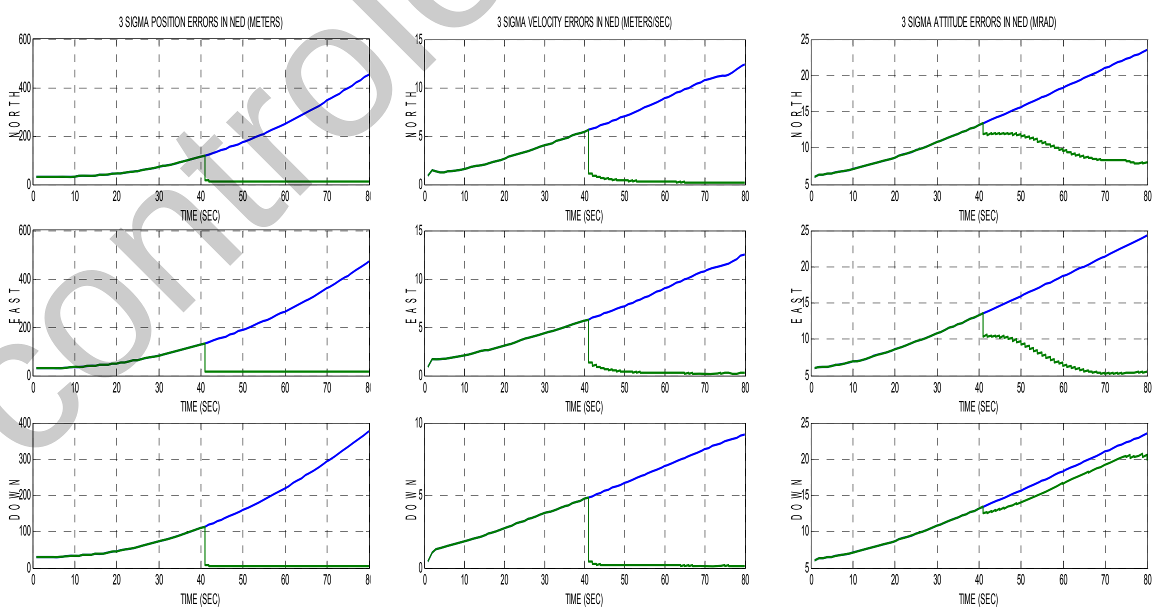

Figure 23. 3σ Position, Velocity and Attitude Errors versus Time along 15 Km Trajectory for DIGNU-2: Effect of GPS Updates Starting at Apogee

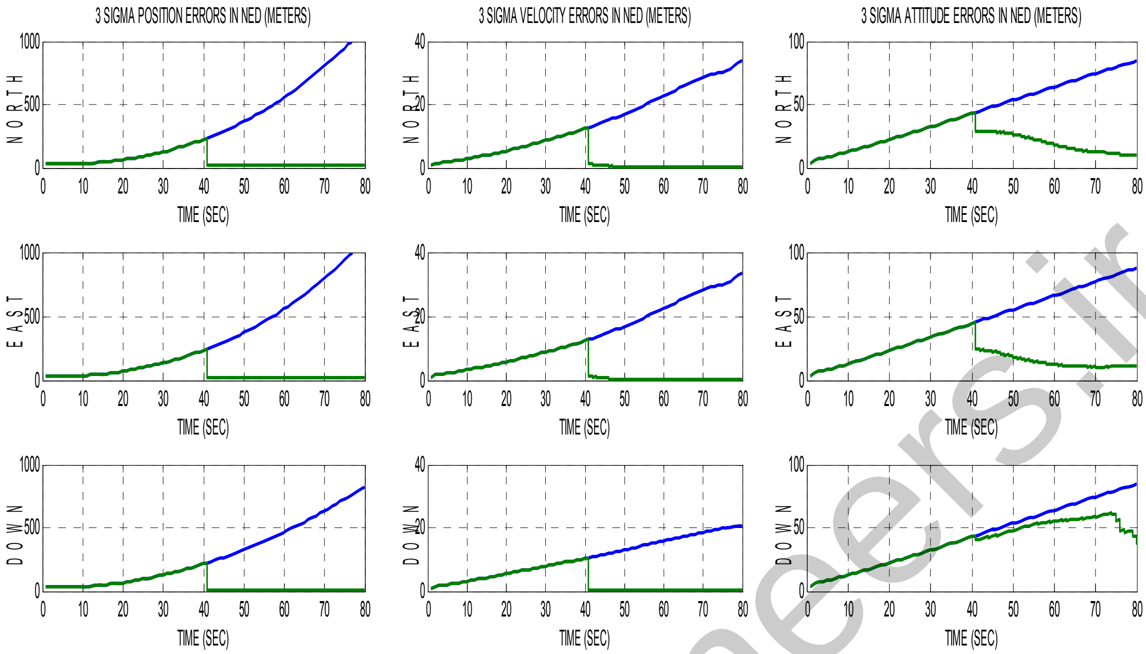

Figure 24. 3σ Position, Velocity and Attitude Errors versus Time along 15 Km Trajectory for DIGNU-1: Effect of GPS Updates Starting at Apogee

IV. Navigation Study Conclusions

Given below are some conclusions from the navigation error sensitivity study:

- For all the initial condition cases, the IMU-induced growth of alignment errors during flyout is not excessive. The maximum 3σ alignment error is about 25 mrad for DIGNU-2 and about 6 mrad for DIGNU-3. These are much less than the seeker FOV of $\pm$ 7 degrees (122 mrad).

- In all cases, the velocity error growth results in heading errors much less than the 33 m/s acquisition limit.

- Position error growth is always below the 245 m acquisition limit for DIGNU-3. For DIGNU-2, the position errors exceed the acquisition limit at ranges slightly greater than 10 km for Cases 1 and 2. The Case 3 position errors are always below the limit.

- Thus, even with conservative initial errors (Case 1), for DIGNU-2 or DIGNU-3 and ranges out to 10 km, performance is very good using free inertial guidance. For these IMUs and conditions, there appears to be no need for in-flight GPS updates.

- Going from Case 1 to Case 2 had a relatively small effect on DIGNU-2, but was more pronounced for DIGNU-3.

- If the launcher GPS/IMU system were able to calibrate out the DIGNU IMU static biases prior to launch (Case 3), a substantial improvement in the errors at acquisition would result. However, achieving Case 3 initial conditions may be operationally difficult (for reasons cited previously). In any case, achieving Case 3 conditions appears unnecessary given the very good performance for Case 1 or 2.

V. Summary

This paper has reviewed the design and evaluation of a ship self defense missile. The operational concept of defending against a swarm attack of small boats, and a particular guidance strategy designed to counter the swarm threat was described. The basic missile airframe and guidance law requirements were discussed and a design achieving the objectives of small miss distance and a required terminal impact angle was demonstrated. The IMU and navigation system requirements necessary to guarantee a high probability of target acquisition by a strapdown, narrow field-of-view seeker were determined. It was shown that an accurate GPS-aided IMU system on the ship could transfer alignment and calibrate out navigation initialization errors for the missile before launch. In this case, a low cost MEMS IMU from the Army Common Guidance IMU development program was shown to be adequate for the missile. The study determined that the more accurate DIGNU-3 IMU could meet all missile requirements even under conservative navigation error assumptions. If required, GPS in-flight navigation updates could be used to allow successful missile operation with the much less accurate DIGNU-1 or DIGNU-2 IMUs.

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Acknowledgment

The authors would like to thank Mr Gil Graff of the Office of Naval Research for his support of this task.

References

[1]Ohlmeyer, E. J., "Control of Terminal Engagement Geometry Using Generalized Vector Explicit Guidance," *American Control Conference*, Volume 1, IEEE, Piscataway, NJ, 2003.

[2]Ohlmeyer, Ernest J. and Phillips, Craig A., "Generalized Vector Explicit Guidance," *AIAA Journal of Guidance, Control, and Dynamics*, Vol. 29, No. 2, March-April, 2006, pp 261-268.

[3]Phillips, Craig A., Ohlmeyer, Ernest J., and Sorenson, Shane, "Optimized Trajectory Shaping Guidance for an Air-to-Ground Missile Launched from a Gunship," *NDIA 40th Annual Armament Systems: Guns ˜ Ammunition ˜ Rockets ˜ Missiles Conference*, 25-28 April 2005, http://www.dtic.mil/ndia/index.html.

[4]D. H. Titter ton and J. L. Weston, *Strapdown Inertial Navigation Technology*, Peter Peregrines Ltd., London, UK, 1997, Section 9.5, "Alignment at Sea."

[5]Ohlmeyer, E. J., Pepitone, T. R., Miller, B. L., "Assessment of Integrated GPS/INS for the EX-171 Extended Range Guided Munition,", *AIAA Guidance, Navigation, and Control Conference, Boston, MA*, Aug. 10-12, 1998, pp. 1374-1389.

[6]Panhorst, David and LeFevre, Vicki, "MEMS in Weapons," *NDIA International Armaments Technology Symposium*, 14-16 June, 2004, http://www.dtic.mil/ndia/index.html.

[8]LeFevre, Vicki; Rider, Lynne; and Panhorst, David, "MEMS IMU – Common Guidance," *NDIA 40th Annual Armament Systems: Guns ˜ Ammunition ˜ Rockets ˜ Missiles Conference*, 25-28 April 2005, http://www.dtic.mil/ndia/index.html.