# May 2009
## Distinguished Lecturers Program
**Dr. James R. Huddle, Chair**

All AESS Chapters and IEEE Sections are encouraged to take advantage of the AESS Distinguished Lecturers Program for their regular or special meetings. We have selected an outstanding list of speakers who are experts in their fields. The AES Society will cover up to $1000 of the speaker’s expenses for travel in North America, with any remaining amount normally covered by the AESS Chapter or Section or by the speaker’s organization. For travel outside North America, the AES Society will cover half of the speaker’s expenses per trip, up to a maximum of $2500. An additional allowance of up to $500 is available for each additional lecture location within any region. The procedure for obtaining a speaker is as follows: If a Chapter or Section has an interest in inviting one of the speakers, it should first contact the speaker directly in order to obtain his agreement to give the lecture on a particular date. After this is accomplished, and if the Chapter or Section wishes to request financial support from the AESS, it should contact James R. Huddle on (818) 715-3264, F (818) 715-3976, j.huddle@ieee.org at least 30 days before the planned meeting, in order to obtain approval for the financial support. The list of distinguished speakers who have expressed their willingness to speak to Chapters or Sections, along with their organization, topics, and telephone numbers, is given below.

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<tr>
<th>Title</th>
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<tr>
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<td>Back-Side Lunar Observatories</td>
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<td>One Hundred Years of Inertial Navigation Practitioner’s View of</td>
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<td>Century</td>
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<td>Sensors</td>
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Suggestions for other subjects should be forwarded to: James Huddle, VP-Technical Operations.
INS/GPS Technology Trends

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Abstract
This paper focuses on accuracy and other technology trends for inertial sensors, Global Positioning Systems (GPS), and integrated Inertial Navigation System (INS)/GPS systems, including considerations of interference, that will lead to better than 1 meter accuracy navigation systems of the future. For inertial sensors, trend-setting sensor technologies will be described. A vision of the inertial sensor instrument field and strapdown inertial systems for the future is given. Planned accuracy improvements for GPS are described. The trend towards deep integration of INS/GPS is described, and the synergistic benefits are explored. Some examples of the effects of interference are described, and expected technology trends to improve system robustness are presented.

1.0 Introduction
Inertial navigation systems have progressed from the crude electromechanical devices that guided the early V-2 rockets (Figure 1a) to the current solid-state devices that are in many modern vehicles. The impetus for this significant progress came during the ballistic missile programs of the 1960s, in which the need for high accuracy at ranges of thousands of kilometers using autonomous navigation systems was apparent. By “autonomous” it is meant that no man-made signals from outside the vehicle are required to perform navigation. If no external man-made signals are required, then an enemy cannot jam them.

One of the early leaders in inertial navigation was the Massachusetts Institute of Technology (MIT) Instrumentation Laboratory (now Draper Laboratory), which was asked by the Air Force to develop inertial systems for the Thor and Titan missiles and by the Navy to develop an inertial system for the Polaris missile. This request was made after the Laboratory had demonstrated in 1953 the feasibility of autonomous all-inertial navigation for aircraft in a series of flight tests with a system called SPIRE (Space Inertial Reference Equipment), Figure 1b. This system had gimbals, was 5 feet in diameter and weighed 2700 pounds. The notable success of those early programs led to further application in aircraft, ships, missiles, and spacecraft such that inertial systems are now almost standard equipment in military and civilian navigation applications.
Inertial navigation systems do not indicate position perfectly because of errors in components (the gyroscopes and accelerometers) and errors in the model of the gravity field that the INS implements. Those errors cause the error in indicated position to grow with time. For vehicles with short flight times, such errors might be acceptable. For longer-duration missions, it is usually necessary to provide periodic updates to the navigation system such that the errors caused by the inertial system are reset as close to zero as possible. Because GPS offers world-wide, highly accurate position information at very low cost, it has rapidly become the primary aid to be used in updating inertial systems, at the penalty of using an aid that is vulnerable to interference. Clearly, the ideal situation would be low-cost but highly accurate INS that can do all, or almost all, of the mission without using GPS.

The military has had access to a specified accuracy of 21 m (95-percent probability) from the GPS Precise Positioning Service (PPS). This capability provides impressive worldwide navigation performance, especially when multiple GPS measurements are combined in a Kalman filter to update an INS on a military platform or a weapon. The Kalman filter provides an opportunity to calibrate some of the GPS errors, such as satellite clock and ephemeris errors, as well as several of the inertial system errors, and when properly implemented, Circular Error Probables (CEPs) better than 5m have been observed. In the near term, accuracies in the integrated navigation solution are predicted to improve to the 1 meter level. These accuracies will need to be available in the face of intentional interference of GPS, and the inertial system will provide autonomous navigation information during periods of GPS outage.

The following sections describe:

- The expected technology trends for inertial sensors and strapdown (no gimbals) systems that can support autonomous operation at low cost. Expectations are for strapdown INS/GPS systems that are smaller than 3 in\(^3\) and weigh less than a pound, and possibly cost under $1000.
- Expected accuracy improvements and implementations for GPS.
- Issues and benefits of INS/GPS integration, particularly in an environment with interference.

The combination of a robust, antijam GPS receiver and an accurate, low-cost inertial system will provide the global precision navigation system of the future. Figure 2 depicts the “roadmap” to meeting this objective.

![Figure 2. Roadmap to precision navigation for multiple applications.](image-url)
2.0 Inertial Sensor Trends

The major error sources in the inertial navigation system are due to gyro and accelerometer inertial sensor imperfections, incorrect navigation system initialization, and imperfections in the gravity model used in the computations. But, in nearly all inertial navigation systems, the largest errors are due to the inertial sensors.

Whether the inertial sensor error is caused by internal mechanical imperfections, electronics errors, or other sources, the effect is to cause errors in the indicated outputs of these devices. For the gyros, the major errors are in measuring angular rates. For the accelerometers, the major errors are in measuring acceleration. For both instruments, the largest errors are usually a bias instability (measured in deg/hr for gyro bias drift, or micro g (μg) for the accelerometer bias), and scale-factor stability (which is usually measured in parts per million (ppm) of the sensed inertial quantity). The smaller the inertial sensor errors, the better the quality of the instruments, the improved accuracy of the resulting navigation solution, and the higher the cost of the system. As a “rule-of-thumb,” an inertial navigation system equipped with gyros whose bias stability is 0.01 deg/hr will see its navigation error grow at a rate of 1 nmi/hr of operation. The navigation performance requirements placed on the navigation system lead directly to the selection of specific inertial instruments in order to meet the mission requirements.

Figure 3, “Current Gyro Technology Applications,” gives a comprehensive view of the gyro bias and scale-factor stability requirements for various mission applications and what type of gyro is likely to be used in current applications (Figures 3 – 9 are revised versions of the figures in Ref. [1]).

![Figure 3. Current gyro technology applications.](image-url)

Solid-state inertial sensors, such as Microelectromechanical System (MEMS) devices, have potentially significant cost, size, and weight advantages, which has resulted in a proliferation of the applications where such devices can be used in systems. While there are many conventional military applications, there are also many newer applications that will emerge with the low cost and very small size inherent in such sensors, particularly at the lower performance end of the spectrum. A vision of the gyro inertial instrument field for relevant military applications for the near-term is shown in Figure 4. Strapdown systems will also dominate.

The MEMS and Interferometric Fiber-Optic (IFOG) technologies are expected to replace many of the current systems using Ring Laser Gyros (RLGs) and mechanical instruments. However, one particular area where
The RLG is expected to retain its superiority over the IFOG in applications requiring extremely high scale-factor stability. The change to all-MEMS technology hinges primarily on MEMS gyro development. The performance of MEMS instruments is continually improving, and they are currently being developed for many applications. This low cost can only be attained by leveraging off the consumer industry, which will provide the infrastructure for supplying the MEMS sensors in extremely large quantities (millions). The use of these techniques will result in low-cost, high-reliability, small-size, and lightweight inertial sensors and the systems into which they are integrated. The tactical (lower) performance end of the application spectrum will likely be dominated by micromechanical inertial sensors. The military market will push the development of these sensors for applications such as “competent” and “smart” munitions, aircraft and missile autopilots, short-time-of-flight tactical missile guidance, fire control systems, radar antenna motion compensation, “smart skins” using embedded inertial sensors, multiple intelligent small projectiles such as flechettes or even “bullets,” and wafer-scale INS/GPS systems.

Figure 4. Near-term gyro technology applications.

Figure 5 shows how the gyro technology may possibly be applied to new applications in the far term. The figure shows that the MEMS and integrated-optics (IO) systems technology will dominate the entire low- and medium-performance range. The rationale behind this projection is based on two premises. The first is that gains in performance in the MEMS devices will continue with similar progression to the dramatic 3 to 4 orders-of-magnitude improvement that has already been accomplished in the last decade. That further improvements are likely is not unreasonable since the designers are beginning to understand the effects of geometry, size, electronics, and packaging on performance and reliability. Second, efforts are already underway to put all six sensors on one (or two) chips, which is the only way to reach a possible cost goal of less than $1000 per INS/GPS system. In addition, since many of the MEMS devices are vibrating structures with a capacitive readout, this may restrict the performance gains. It is in this area that the integrated optics technology is most likely to be required to provide a true solid-state micromechanical gyro with optical readout. At this time, the technology to make a very small, accurate gyro does not exist, but advances in integrated optics are already under development in the communications industry. For the strategic application, the IFOG could become the dominant gyro. Work is underway now to develop radiation-hard IFOGs as well as super-high-performance IFOGs.
A potentially promising technology, which is in its infancy stages, is inertial sensing based upon atom interferometry (sometimes known as cold atom sensors). A typical atom de Broglie wavelength is $10^{-11}$ times smaller than an optical wavelength, and because atoms have mass and internal structure, atom interferometers are extremely sensitive. (Ref. [16]) Accelerations, rotations, electromagnetic fields, and interactions with other atoms change the atom interferometric fringes. This means that atom interferometers could make the most accurate gyroscopes, accelerometers, gravity gradiometers, and precision clocks, by orders of magnitude. If this far-term technology can be developed, then it could result in a 2 to 5-meter/hour navigation system without GPS, in which the accelerometers are also measuring gravity gradients.

![Diagram](image)

**Figure 5.** Far-term gyro technology applications.

Figure 6, “Current Accelerometer Technology Applications,” gives a comprehensive view of the accelerometer bias and scale-factor stability requirements for various mission applications and what type of accelerometer is likely to be used in current applications. “Mechanical Instruments” refers to the use of a Pendulous Integrating Gyro Assembly (PIGA) which is a mass unbalanced spinning gyroscope used to measure specific force.

![Diagram](image)

**Figure 6.** Current accelerometer technology applications.
Current applications are still dominated by electromechanical sensors, not only because they are generally low-cost for the performance required, but also because no challenging alternative technology has succeeded, except for quartz resonators, which are used in the lower-grade tactical and commercial applications. MEMS inertial sensors have not yet seriously broached the market, although they are on the verge of so doing, especially in consumer applications.

In the near-term (Figure 7), it is expected that the tactical (lower) performance end of the accelerometer application spectrum will be dominated by micromechanical accelerometers. As in the case for gyros, the military market will push the development of these sensors for applications such as “competent” and “smart” munitions, aircraft and missile autopilots, short-time-of-flight tactical missile guidance, fire control systems, radar antenna motion compensation, “smart skins” using embedded inertial sensors, multiple intelligent small projectiles such as flechettes or even “bullets,” and wafer-scale INS/GPS systems. Higher performance applications will continue to use mechanical accelerometers and possibly resonant accelerometers based on quartz or silicon. Quartz resonant accelerometers have proliferated widely into tactical and commercial (e.g., factory automation) applications. Silicon micromechanical resonator accelerometers are also being developed. Both of these technologies have possible performance improvements.

![Figure 7. Near-term accelerometer technology applications.](image)

Figure 8 shows how the accelerometer technology may be applied to new applications in the far term. As in the case of gyro projections for the future, the figure shows that the MEMS and integrated optics technology will dominate the entire low- and medium-performance range. The rationale behind this projection is based on exactly the same two premises as for the gyros. However, it is likely that the far-term accelerometer technology projections will be realized years sooner than the gyro.

Figure 9 shows INS or INS/GPS relative strapdown system cost “projections” as a function of inertial instrument technology and performance. The cost of a GPS receiver is likely to be so small that it will be insignificant. The systems are classified as: laser gyro or IFOG systems containing various types of accelerometer technologies; quartz systems with both quartz gyros and quartz accelerometers; and MEMS/integrated optics systems. The solid line indicates the range of approximate costs expected. Clearly, the quantity of systems produced affects the cost; large production quantities would be at the lower end of the cost range. The IFOG systems have the potential for lower cost than laser gyro systems because the IFOG should be well below the cost of an RLG. However, this has not happened to date, primarily because the RLG is in relatively large-volume production in well-facilitated factories and the IFOG is not yet manufactured in similar production quantities. Clearly, the MEMS/integrated optics INS/GPS systems offer the lowest cost. The ultimate low cost only becomes feasible in quantities of millions. This can be achieved only with multi-axis instrument clusters and on-chip or adjacent-chip electronics and batch packaging.
The ability of silicon-based MEMS devices to withstand high “g” forces has been demonstrated recently in a series of firings in artillery shells where the g forces reached over 6500 g. These small MEMS-based systems, illustrated in Figure 10, have provided proof-of-principal that highly integrated INS/GPS systems can be developed and led to a recent program where the goal was a system on the order of 3 in$^3$, or 2 in$^3$ for the INS alone (Ref. [2]). Unfortunately, the goals were not met. The current status of a typical MEMS INS is represented by the Honeywell HG1900 with a weight <1 lb., volume <20 cubic inches, power <3 watts, gyro bias of 1 to 30 °/hr, and gyro angle random walk of 0.1 °/√hr. This system is in production. Another is the HG1930 which has a volume of <4 cubic inches, a gyro bias of 20 °/hr and a gyro random walk of 0.15 deg/√hr (Ref. [3]). The volumes compare with tactical grade RLG and IFOG systems with a volume of about 34in$^3$. These systems also represent 4 orders of magnitude improvement in weight and volume over the gimbal system SPIRE. If micromechanical instrument performance improvements can be made, they will come to dominate the entire inertial instrument application spectrum.
3.0 GPS Accuracy and Other Improvements

The accuracy specification that is currently applicable to the GPS results in a precise positioning (PPS) of a GPS receiver operating with the military P(Y) code of approximately 10 m (CEP) in the WGS-84 coordinate system. Recent advances and programs to improve GPS accuracy have contributed to the real possibility of developing INS/GPS systems with smaller than 1-m CEP in the near term. This section will discuss these items.

The accuracy of the GPS PPS provides impressive navigation performance, especially when multiple GPS measurements are combined in a Kalman filter to update an INS. The Kalman filter provides an opportunity to calibrate the GPS errors, as well as the inertial errors, and when properly implemented, CEPs better than either system are achievable.

In assessing GPS accuracy in the mid 1990’s, the largest error sources were in the space and control segment. The space segment dominant errors are: ionospheric errors, tropospheric errors, satellite clock errors, and satellite ephemeris with the latter two errors being dominant. The ionospheric errors can be reduced by using a two-frequency receiver (L₁ and L₂) and tropospheric errors can be reduced by using a deterministic compensation model. Table 1 gives a typical 1995 absolute GPS error budget (Ref. [4], p. 105). Horizontal Dilution of Precision (HDOP) is a geometrical factor that is a function of the geometry between the GPS receiver and the tracked satellites. For tracking four satellites, HDOP is typically 1.5. Then with a user equivalent range error (UERE) of 3.8m, and applying the approximate formula, CEP = (0.83)(HDOP)(UERE), the resulting CEP is 4.7 m.
Table 1. “Typical” absolute GPS error budget. (circa 1995)

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<tr>
<th>GPS Noise-Like Range Errors</th>
<th>1σ Values (m)</th>
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<tr>
<td>Multipath</td>
<td>0.6</td>
</tr>
<tr>
<td>Receiver noise</td>
<td>0.3</td>
</tr>
<tr>
<td>RMS noise-like error</td>
<td>0.7</td>
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<table>
<thead>
<tr>
<th>GPS Bias-Like Range Errors</th>
<th>1σ Values (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Satellite ephemeris</td>
<td>1.4</td>
</tr>
<tr>
<td>Satellite clock</td>
<td>3.4</td>
</tr>
<tr>
<td>Atmospheric residual</td>
<td>0.2</td>
</tr>
<tr>
<td>RMS bias-like error</td>
<td>3.7</td>
</tr>
</tbody>
</table>

User equivalent range error (UERE) = (0.7² + 3.7²)⁺²=3.8m
CEP = (0.83) (UERE) (HDOP) = 4.7m if HDOP = 1.5

Beginning in the mid 1990’s various accuracy improvement programs were begun (Refs. [4] – [7]) to reduce the clock and ephemeris errors listed in Table 1. These errors can be reduced by sending more accurate and more frequent ephemeris and clock updates to the satellites from the control segment. In addition, if pseudorange corrections for all satellites are uploaded in each scheduled, individual satellite upload, then a PPS receiver can decode the messages from all satellites it is tracking and apply the most recent correction set. Increasing the upload frequency to three uploads per day for each satellite is expected to improve the combined error contribution of clock and ephemeris for PPS users by 50% by substantially decreasing the average latency of 11.5 hours in the data broadcast by the satellites.

In another phase of the program called the Accuracy Improvement Initiatives, the data from six National Geospatial Agency (NGA) GPS monitoring sites were integrated with data from the six existing Air Force monitoring sites in the operational control segment (OCS). By including additional data from the NGA sites, which are located at higher latitudes than the Air Force sites, an additional 15-percent improvement in combined clock and ephemeris accuracy is predicted. Improvements to the Kalman filter that is used in the ground control segment to process all the satellite tracking information can further reduce the errors by 15 percent. In addition, by incorporating better dynamical models in the filter, another 5-percent improvement may be anticipated. Table 2 summarizes these predicted accuracy improvements (Ref. [4], p. 102).
Table 2. Planned reduction of combined clock and ephemeris errors.

<table>
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<tr>
<th>Enhancement</th>
<th>Anticipated Combined Clock and Ephemeris Error Improvement over Existing Combined Error of 3.7 m (1σ)</th>
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<tr>
<td>Correction Updates (50% reduction)</td>
<td>1.8 m</td>
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<tr>
<td>Additional Monitor Stations (additional 15% reduction)</td>
<td>1.5 m</td>
</tr>
<tr>
<td>Non partitioned Kalman Filter (additional 15% reduction)</td>
<td>1.3 m</td>
</tr>
<tr>
<td>Improved Dynamic Model (additional 5% reduction)</td>
<td>1.2 m</td>
</tr>
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</table>

Figure 11 shows the additional six NGA sites added in the initial stages of the Accuracy Improvement Initiative. The final five NGA sites included were at even higher latitudes to provide even more tracking data and additionally provide triple ground station usability of every GPS satellite.

![Figure 11. OCS and NGA Tracking Stations](image.png)

Improvements in the GPS Master Station Control Segment software such as implementing a non-partitioned Kalman filter and improved dynamic models are presented in Figure 12.
After all of these improvements, a ranging error on the order of 1.4 m is a reasonable possibility with the atmospheric residual unchanged. With all-in-view tracking (HDOP approximately 1.0), CEPs on the order of 1 m appear quite possible in the near term. CEP=(0.83) (1.0) (1.4) = 1.1 m. If then, multiple GPS measurements are combined with an inertial system and Kalman Filter, better than 1 m accuracy should result.

To illustrate the benefits of the various GPS improvements, a simulation was conducted with an error model for a typical INS whose errors would result in 1.0 nmi/h error growth rate without GPS aiding. After 30 minutes of air vehicle flight including GPS updates every second, with all of the GPS accuracy improvements included, less than 1 meter CEP is obtained as shown in Table 3.

Table 3. Tightly coupled INS/GPS System-Air Vehicle Trajectory (@30 min).

<table>
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<tr>
<th>CLOCK AND EPHEMERIS ERROR (1σ) ALL IN VIEW TRACKING</th>
<th>CEP (m) 8 SATELLITES</th>
</tr>
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<tbody>
<tr>
<td>Current Model</td>
<td>2.97 m</td>
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<tr>
<td>Correction Updates</td>
<td>1.46 m</td>
</tr>
<tr>
<td>Additional Monitor Stations</td>
<td>1.22 m</td>
</tr>
<tr>
<td>Non-partitioned Kalman Filter</td>
<td>1.06 m</td>
</tr>
<tr>
<td>Improved Dynamic Model</td>
<td>0.98 m</td>
</tr>
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</table>
Another significant improvement in GPS for military systems will be the introduction of the M-code in GPS III, which is designed to be more secure and have better jamming resistance than the current Y code (Ref. [7]). The system is being designed such that a higher power signal (+20 dBW over current signal levels) will be available for localized coverage over an area of operations to boost signal jamming resistance. This significant improvement (M-code spot beam) is scheduled for the GPS-III phase of the GPS modernization process.

4.0 INS/GPS Integration

Many military inertial navigation systems could be replaced with less accurate inertial systems if it were guaranteed that GPS would be continuously available to update the inertial system to limit its error growth. A less accurate inertial system usually means a less costly system. However, given the uncertainty in the continuous availability of GPS in most military scenarios, an alternate way to reduce the avionics system cost is to attack the cost issue directly by developing lower-cost inertial sensors while improving their accuracy and low noise levels, as described in the “Inertial Sensor Trends” section. For applications without an interference threat, in the future, GPS updating is expected to provide better than 1-m navigation accuracy (CEP) when used in conjunction with an INS. The benefits and issues in using INS augmented with GPS updates, including a discussion of interference issues, have been presented in many references. Systems currently in use tend to be classified as either “the loosely coupled approach” or “the tightly coupled approach” (Figures 13 and 14 and Ref. [8]).

Figure 13. Loosely coupled approach.

Figure 14. Tightly coupled approach.
The most recent research activity is a different approach called “deep integration” (Figure 15, Refs. ([9] and [10]). In this approach, the problem is formulated directly as an estimation problem in which the optimum (minimum-variance) solution is sought for each component of the multidimensional navigation state vector. By formulating the problem in this manner, the navigation algorithms are derived directly from the assumed dynamical models, measurement models, and noise models. The solutions that are obtained are not based on the usual notions of tracking loops and operational modes (e.g., State 3, State 5, etc.). Rather, the solution employs a nonlinear filter that operates efficiently at all jammer/signal (J/S) levels and is a significant departure from traditional extended Kalman filter designs. The navigator includes adaptive algorithms for estimating postcorrelation signal and noise power using the full correlator bank. Filter gains continuously adapt to changes in the J/S environment, and the error covariance propagation is driven directly by measurements to enhance robustness under high jamming conditions.

Extended-range correlation may be included optionally to increase the code tracking loss-of-lock threshold under high jamming and high dynamic scenarios. If excessively high jamming levels are encountered (e.g., beyond 70-75 dB J/S at the receiver input for P(Y) code tracking), the GPS measurements may become so noisy that optimal weights given to the GPS measurements become negligible. In this situation, navigation error behavior is essentially governed by current velocity errors and the characteristics of any additional navigation sensors that are employed, such as an INS. Code tracking is maintained as long as the line-of-sight delay error remains within the maximum allowed by the correlator bank. If there is a subsequent reduction in J/S so that the optimal weights become significant, optimum code tracking performance is maintained without the need for reacquisition. Detector shapes for each correlator depend on the correlator lag and rms line-of-sight delay error.

Experiments have shown an improvement in code tracking of about 10 to 15 dB in wideband A/J capability for this architecture. Another 5 dB might be possible with data stripping to support extended predetection integration. Given that the implementation is done in software, it would be expect to be used in many future INS/GPS implementations. “Deep integration” is trademarked by the C.S. Draper Laboratory, Inc.

Figure 15. INS/GPS deep integration.
5.0 INS/GPS Interference Issues

Interference to the reception of GPS signals can be due to many causes such as telecommunication devices, local interference from signals or oscillators on the same platform, or possibly radar signals in nearby frequency bands. Attenuation of the GPS signal can be caused by trees, buildings, or antenna orientation, and result in reduced signal/noise ratio even without interference. This loss of signal can result in an increase in effective jammer/signal (J/S) level even without intentional jamming or interference. The minimum received signal power at the surface of the Earth is about -155dBW, a level easily overcome by a jammer source.

Military receivers are at risk due to intentional jamming. Jammers as small as 1 W located at 100 km from the receiver can possibly prevent a military receiver from acquiring the satellite signals and “locking-on” to C/A code. Representative jammers are shown in Figure 16. Larger jammers are good targets to find and to attack because of their large radiated power. Smaller jammers, which are hard to find, need to be defended against by improved anti-jam (A/J) technologies within the receiver, improved antennas, or by integration with an inertial navigation system. Proponents of high-accuracy inertial systems will generally argue that a high anti-jam GPS receiver is not required, while receiver proponents will argue that using a higher A/J receiver will substantially reduce inertial system accuracy requirements and cost. Both arguments depend entirely on the usually ill-defined mission and jamming scenario.

What has generally become accepted is that the GPS is remarkably vulnerable to jamming during the C/A code acquisition phase where conventional receiver technology has only limited jammer tolerability (J/S - 27 dB) (Refs. [10], [11], [12]). A 1-W (ERP) jammer located at 100 km from the GPS antenna terminals could prevent acquisition of the C/A code. Figure 17 is very useful in determining trade-offs between required A/J margin and jammer power. A 1-W jammer is “cheap” and potentially the size of a hockey puck. Furthermore, the C/A code can be spoofed by an even smaller power jammer. So generally, a GPS receiver cannot be expected to acquire the C/A code in a hostile environment.

Figure 16. Jammer possibilities.
For long-range cruise missile type applications, the C/A code could be acquired outside hostile territory and then the receiver would transition to P(Y) code lock, which has a higher level of jamming immunity. A 1-kW (ERP) jammer at about 100 km would now be required to break inertially-aided receiver code lock at 54 to 57 dB. As the weapon approaches the jammer, jammer power levels of about 10 W would be effective in breaking P(Y) code lock at 10 km (see Figure 18).

As previously mentioned, the “deep integration” architecture for combining INS and GPS may allow for tracking GPS satellites up to 70 – 75 dB J/S, an improvement of 15 to 20 dB above conventional P(Y) code tracking of 54 to 57 dB. If future increases of 20 dB in broadcast satellite power using the M-code spot beam (M spot) are also achieved, nearly 40 dB of additional performance margin would be achieved, so a jammer of nearly 100 kW would be required to break lock at 10 km. Furthermore, new receiver technology with advanced algorithms and space-time adaptive or nulling antenna technologies might also be incorporated into the system, further increasing its A/J capability significantly.
Recently (Ref. [3]) Honeywell and Rockwell Collins created a joint venture, Integrated Guidance Systems LLC, to market and produce a series of deep integration guidance systems. The IGS-200, for example, is G-hardened for artillery applications (15,750G), has a volume 16.5 in³, weighs < 1.25 lb., is based on the 1930G Honeywell MEMS IMU, and with deep integration and 2-channel digital nulling, the system supposedly has 80 – 90 dB J/S against a single jammer.

If A/J performance is increased significantly, then the jammer power must also be increased significantly. A large jammer would present an inviting target to an antiradiation, homing missile. In the terminal area of flight against a target, the jammer located at the target will eventually jam the receiver, and the vehicle will have to depend on inertial-only guidance or the use of a target sensor. Thus, it is important to ensure that accurate guidance and navigation capability is provided to meet military mission requirements against adversaries who are willing to invest in electronic countermeasures (ECM). This fact is true today and is expected to remain so in the foreseeable future. Figure 19 summarizes electronic counter-countermeasures (ECCM) techniques.

- **Lower Cost, High-Accuracy IMU’s**
- **Improve Signals in Space**
  - Increased Accuracy
  - Mcode and Mspot
- **Improved Receivers**
  - Deep Integration With IMU
  - Anti-Spoof Techniques
  - Higher A/J Electronic
- **Direct P (Y) Code Acquisition, Lock-on Before Launch**
  - Improved Aircraft Interface To Munitions
  - Miniature On-board Clock
  - Multiple Correlators
- **Higher Performance, Lower Cost Adaptive Antennas**
  - Digital Beamforming
  - Modern Algorithms

**Figure 19.** Valuable ECCM technologies and techniques.

### 6.0 Concluding Remarks
Recent progress in INS/GPS technology has accelerated the potential use of these integrated systems, while awareness has also increased concerning GPS vulnerabilities to interference. Accuracy in the broadcast GPS signals will allow 1 meter INS/GPS accuracy. Many uses will be found for this high accuracy. In parallel, lower-cost inertial components will be developed and they will also have improved accuracy. Highly integrated A/J architectures for INS/GPS systems will become common, replacing avionics architectures based on functional black boxes where receivers and inertial systems are treated as stand-alone systems.

For future military and civilian applications, it is expected that the use of INS/GPS systems will proliferate and ultimately result in worldwide navigation accuracy better than 1 m, which will need to be maintained under all conditions. It can be expected that applications such as personal navigation systems, micro air vehicles (MAV), artillery shells, and automobiles will be quite common, see Figure 20. Other applications will certainly include spacecraft, aircraft, missiles, commercial vehicles, and consumer items.
Acknowledgments/Additional References
Thanks to Neil Barbour for assistance with the section on Inertial Sensor Trends. A history of inertial navigation is given in Ref. [14] and a history of the GPS program is given in Ref. [15].

References


The interested reader is directed to books

**Bibliography of selected NATO AGARD and NATO RTO Reports**
INS/GPS Integration Architecture Performance Comparisons

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Abstract

Performance comparisons between the three major INS/GPS system architectures for various mission scenarios will be presented in order to understand the benefits of each. The loosely coupled and tightly coupled systems will be compared in several scenarios including aircraft flying against jammers and a helicopter flying a scout mission. The tightly coupled and deeply integrated architectures will be compared for several jamming scenarios including that of a precision guided munition.

1.0 Introduction

In Reference 1, INS/GPS integration architectures defined as loosely coupled, tightly coupled, and deeply integrated configurations were described. The advantages and disadvantages of each level of integration were listed. Examples of current and future systems were cited. In this paper, performance comparisons between the three major INS/GPS system architectures for various mission scenarios will be presented in order to understand the benefits of each. The loosely coupled and tightly coupled systems will be compared in several scenarios including aircraft flying against jammers and a helicopter flying a scout mission. The tightly coupled and deeply integrated architectures will be compared for several jamming scenarios including that of a precision guided munition.

2.0 Loosely Coupled vs. Tightly Coupled Performance Comparison

This section shows the jamming-related performance of loosely coupled and tightly coupled INS/GPS navigation systems in several hypothetical situations. In addition to comparing navigation architectures, the performance of inertial systems of varying quality was evaluated. The analysis considered only the performance of the combined INS/GPS solution and is thus appropriate to either of the loosely coupled architectures as they share the same INS/GPS solution. This particular example of an INS/GPS loosely coupled system has been the subject of numerous published studies [e.g., Ref. 2]. The tightly coupled system did not necessarily correspond to any particular existing system.

Several jamming scenarios were considered. The first scenario was designed to simply show the behavior of INS/GPS systems when GPS satellites are lost and reacquired one at a time. That is, there will be four satellites in track, then three, two, one, and finally zero. Then they were reacquired one at a time. For one of the scenarios, the navigation system was augmented with a Doppler ground speed measuring device.

2.1 Loosely Coupled System Definition

The loosely coupled GPS system consisted of a GPS receiver, an inertial navigation system, and an integration filter. The PVA solution from a typical receiver like the MAGR was used as the input to the INS/GPS integration Kalman filter. In order to avoid the problem of dealing with correlated measurements, the integration filter only used the position from the PVA solution, and this only once every 10 s and only if the Expected Horizontal Error (EHE), a receiver output and measure of horizontal navigation quality, was less than 100 m. The receiver did not compute a solution if there were fewer than four satellites in track. The state elements for the GPS receiver are shown in Table 2.1.
Table 2.1. State Elements for the Unaided GPS Receiver.

<table>
<thead>
<tr>
<th>State Element</th>
<th>Components</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position</td>
<td>3</td>
</tr>
<tr>
<td>Velocity</td>
<td>3</td>
</tr>
<tr>
<td>Acceleration</td>
<td>3</td>
</tr>
<tr>
<td>User clock bias</td>
<td>1</td>
</tr>
<tr>
<td>User clock drift</td>
<td>1</td>
</tr>
<tr>
<td>Altimeter bias</td>
<td>1</td>
</tr>
<tr>
<td>Total</td>
<td>12</td>
</tr>
</tbody>
</table>

Since the GPS receiver solution is the result of a (Kalman) filter, the velocity is correlated with the position, and both position and velocity are correlated in time. Process noise, which allows the filter to track changing acceleration, also decorrelates the output. The process noise is of such a magnitude that position solutions separated by 10-s intervals are not significantly correlated. The state elements of the integration filter that processes these measurements is shown in Table 2.2.

Table 2.2. State Elements for the Loosely Coupled Integration Filter.

<table>
<thead>
<tr>
<th>Error State Element</th>
<th>Components</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position</td>
<td>3</td>
</tr>
<tr>
<td>Velocity</td>
<td>3</td>
</tr>
<tr>
<td>Misalignment</td>
<td>3</td>
</tr>
<tr>
<td>Gyro drift</td>
<td>3</td>
</tr>
<tr>
<td>Gyro scale factor</td>
<td>3</td>
</tr>
<tr>
<td>Accel. bias</td>
<td>3</td>
</tr>
<tr>
<td>Accel. scale factor</td>
<td>3</td>
</tr>
<tr>
<td>Altimeter bias</td>
<td>1</td>
</tr>
<tr>
<td>Total</td>
<td>22</td>
</tr>
</tbody>
</table>

Most Kalman filters are suboptimal estimators. Some are less near optimal than others. The cascaded filter architecture of loosely coupled systems is certainly far from optimal. These systems are particularly sensitive to the procedure known as “tuning,” in which the process noise is added and measurements are down-weighted or omitted. A considerable effort went into tuning the loosely coupled INS/GPS system such that it could be compared fairly with the tightly coupled systems.

2.2 Tightly Coupled System Definition

The tightly coupled system consists of a receiver, inertial instruments, and an integration filter. The integration filter accepts measurements of pseudo-range and pseudo-range rate from each satellite at a 1-Hz rate. The filter state is extrapolated forward in time using inertial measurements and a model for the earth’s gravity field. The state elements for this most straightforward approach are shown in Table 2.3. These same states appear in the cascaded filters of the loosely coupled system.
Table 2.3. State Elements for the Tightly Coupled Integration Filter.

<table>
<thead>
<tr>
<th>Error State Element</th>
<th>Components</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position</td>
<td>3</td>
</tr>
<tr>
<td>Velocity</td>
<td>3</td>
</tr>
<tr>
<td>User clock bias</td>
<td>1</td>
</tr>
<tr>
<td>User clock drift</td>
<td>1</td>
</tr>
<tr>
<td>Misalignment</td>
<td>3</td>
</tr>
<tr>
<td>Gyro drift</td>
<td>3</td>
</tr>
<tr>
<td>Gyro scale factor</td>
<td>3</td>
</tr>
<tr>
<td>Accel. bias</td>
<td>3</td>
</tr>
<tr>
<td>Accel. scale factor</td>
<td>3</td>
</tr>
<tr>
<td>Altimeter bias</td>
<td>1</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>24</strong></td>
</tr>
</tbody>
</table>

2.3 Initial Errors, Modeling Errors, and Instrument Errors

These error sources influence the performance of the navigation system, some more than others. The initial errors in position, velocity, and misalignment in fact have very little effect on the performance of the system as long as it operates for a significant time. They are set to levels that are consistent with some kind of ground calibration and alignment mode, but are poor enough to show improvement as in-flight alignment progresses -- with either system architecture. Other errors can have significant effect on navigation system performance. Those errors that are independent of INS quality are given in Table 2.4. The Markov processes in this table are characterized by two numbers, a standard deviation, and a distance constant.

Table 2.4. Error Values for INS Independent Models.

<table>
<thead>
<tr>
<th>Bias Errors</th>
<th>Modeled Value (1σ)</th>
<th>No. Components</th>
</tr>
</thead>
<tbody>
<tr>
<td>Initial position</td>
<td>16 m (vertical) 600 m (horizontal)</td>
<td>1</td>
</tr>
<tr>
<td>Initial velocity</td>
<td>0.3 m/s</td>
<td>3</td>
</tr>
<tr>
<td>GPS user clock</td>
<td>5000 ms 10^{-2} ppm 10^{-3} ppm/g</td>
<td>3</td>
</tr>
<tr>
<td>GPS pseudo-range</td>
<td>3.0 m</td>
<td>4</td>
</tr>
<tr>
<td>GPS range rate</td>
<td>0.003 m/s</td>
<td>4</td>
</tr>
<tr>
<td>Gravity (Markov)</td>
<td>35 µg/37 km</td>
<td>3</td>
</tr>
<tr>
<td>Barometer (Markov)</td>
<td>150 m/460 km</td>
<td>1</td>
</tr>
<tr>
<td>Noise errors</td>
<td></td>
<td></td>
</tr>
<tr>
<td>GPS pseudo-range</td>
<td>from receiver tracking</td>
<td>4</td>
</tr>
<tr>
<td>GPS range rate</td>
<td>loop simulation</td>
<td>4</td>
</tr>
<tr>
<td>Barometer</td>
<td>3 m</td>
<td>1</td>
</tr>
</tbody>
</table>

The performance of four different IMU qualities were analyzed. The four IMUs were characterized by their navigation error after 1 h of unaided (inertial-only) operation. The error characteristics of actual inertial instruments whose performance was close to 10, 1, 0.5 and 0.2 nmi/h were scaled proportionally to yield those exact values.
The error values for each of these hypothetical instruments are shown in Table 2.5.

Table 2.5. IMU Error Sources.

<table>
<thead>
<tr>
<th>Error Source</th>
<th>10 nmi/h</th>
<th>1.0 nmi/h</th>
<th>0.5 nmi/h</th>
<th>0.2 nmi/h</th>
</tr>
</thead>
<tbody>
<tr>
<td>Accel. bias</td>
<td>223 µg</td>
<td>37 µg</td>
<td>19 µg</td>
<td>4.2 µg</td>
</tr>
<tr>
<td>Accel. scale factor</td>
<td>223 ppm</td>
<td>179 ppm</td>
<td>90 ppm</td>
<td>21 ppm</td>
</tr>
<tr>
<td>Input axis misalign.</td>
<td>22 arcsec</td>
<td>3 arcsec</td>
<td>1.5 arcsec</td>
<td>0.4 arcsec</td>
</tr>
<tr>
<td>Random walk</td>
<td>56 µg/√Hz</td>
<td>15 µg/√Hz</td>
<td>7.5 µg/√Hz</td>
<td>4.2 µg/√Hz</td>
</tr>
<tr>
<td>Gyro bias</td>
<td>0.11 deg/h</td>
<td>0.0045 deg/h</td>
<td>0.0022 deg/h</td>
<td>0.00084 deg/h</td>
</tr>
<tr>
<td>Gyro scale factor</td>
<td>112 ppm</td>
<td>7.5 ppm</td>
<td>3.5 ppm</td>
<td>1.67 ppm</td>
</tr>
<tr>
<td>Input axis misalign.</td>
<td>22 arcsec</td>
<td>2.2 arcsec</td>
<td>1.1 arcsec</td>
<td>0.4 arcsec</td>
</tr>
<tr>
<td>Random walk</td>
<td>4.7 deg/h/√Hz</td>
<td>0.13 deg/h/√Hz</td>
<td>0.066 deg/h/√Hz</td>
<td>0.03 deg/h/√Hz</td>
</tr>
</tbody>
</table>

Initial misalignment error was derived from “gyrocompassing” each of the inertial units so it is instrument-dependent. Its values are not critical for the analysis because improvements in alignment due to in-flight maneuvers soon dominate the navigation results.

2.5 GPS Receiver Bandwidth, Loss of Lock and Reacquisition

For the loosely coupled receiver, the noise bandwidths of the code and carrier loop are fixed. The carrier was a third-order loop with bandwidth of 5.83 Hz. The code loop band is first order, but is aided by either the carrier loop if the carrier loop is in lock or by the INS if the carrier loop is not in lock. During carrier loop aiding, the code loop bandwidth is 1.5 Hz. During inertial aiding, the bandwidth is 0.5 Hz.

The bandwidths for the tightly coupled receiver were set appropriate to the quality of inertial instruments. These bandwidths are determined by the requirement that the loops stay in lock for a 10-g/s jerk, which lasts for 0.6 s. (The carrier tracking bandwidth was actually set for this study by requiring that the phase error be less than 90 deg for a 6-g acceleration step. This is a slightly more stringent requirement, but is easier to analyze.) The next several paragraphs present the method used for setting the tracking loop bandwidths. We took maximum advantage of knowing the inertial instrument performance. Closely tuning the tracking loops to the inertial performance in this way may not always be practical for actual receivers.

The phase error in a third-order loop following an acceleration step is shown in the following equation. (Note that distance has been converted to phase error in degrees using the code length of 300 m.)

\[
\Delta \Phi = \frac{R}{\omega_0^2} \left[ e^{-\omega_0 t} + e^{-\omega_0^2 t^2} \left( \sin \omega_0 t \frac{\sqrt{3}}{2} - \sqrt{3} \cos \omega_0 t \frac{\sqrt{3}}{2} \right) \right]
\]

where:  
\( R \) is the step magnitude (deg/s²)  
\( \omega_0 \) is the filter natural frequency (rad/s)  
\( \Delta \Phi \) is the phase error in degrees

The natural frequency should be selected to keep the peak phase error less than 90 deg. The graph in Figure 2.1 shows the response, \( \Delta \Phi \), for a natural frequency of 17.67 rad/s, the maximum error is 90 deg.
Figure 2.1. Error in third-order loop response to a 6-g step in acceleration.

With inertial aiding, the tracking loop will not be affected by the full magnitude of the step in acceleration. Only a residual part of the acceleration step due to imperfect inertial instruments will affect the tracking loop. The error, $\Delta \Phi$, is proportional to the step magnitude and inversely proportional to the square of the natural frequency. To maintain a 90-deg peak error, the natural frequency can be scaled by the square root of the ratio of aided to unaided step magnitude.

$$\omega_{0,\text{aided}} = \sqrt{\frac{R_{\text{aided}}}{R_{\text{unaided}}}} \omega_{0,\text{unaided}}$$ \hspace{1cm} (2.1)

The residual error (post-calibration) accelerometer scale factor and IMU misalignment cause a residual acceleration step to be seen by the tracking loop. Lag in the inertial aiding would also add to the acceleration seen by the tracking loop. This lag was assumed to be negligible in this tightly coupled situation.

The residual acceleration seen by the tracking loop due to scale factor error is shown below.

$$\delta a_{sf} = \begin{bmatrix} sf_1 & 0 & 0 \\ 0 & sf_2 & 0 \\ 0 & 0 & sf_3 \end{bmatrix} a$$

IMU misalignment causes acceleration to be rotated incorrectly. The error in acceleration due to misalignment is shown below.

$$\delta a_{mis} = \begin{bmatrix} 0 & -\delta \theta_3 & \delta \theta_2 \\ \delta \theta_3 & 0 & -\delta \theta_1 \\ -\delta \theta_2 & \delta \theta_1 & 0 \end{bmatrix} a$$

The net error caused by scale factor and misalignment due to a unit acceleration step is thus:

$$\delta a = \delta a_{sf} + \delta a_{mis}$$
The covariance of residual acceleration error due to a unit acceleration step is shown below.

\[
\langle \delta a \delta a^T \rangle = \begin{bmatrix}
1 & 0 & 0 & -1 & 1 \\
0 & 1 & 0 & 1 & 0 \\
0 & 0 & 1 & -1 & 1 \\
\end{bmatrix} \begin{bmatrix}
\sigma_{sf}^2 & c_{sf-mis} & c_{sf-mis}^2 \\
\end{bmatrix} \begin{bmatrix}
1 & 0 & 0 \\
0 & 1 & 0 \\
0 & 0 & 1 \\
-1 & 0 & 1 \\
1 & -1 & 1 \\
\end{bmatrix}
\]

The quantities \(\sigma_{sf}\) and \(\sigma_{mis}\) are the scale factor and misalignment standard deviations. The quantity \(c_{sf-mis}\) is the covariance of these two quantities. This scale factor/misalignment matrix (the middle factor on the right) was taken from a covariance analysis after the aircraft had performed in-flight calibration maneuvers. If the scale factor is expressed as a fraction and the misalignment is in radians, the acceleration variance (on the left) will be the variance in acceleration seen by the tracking loop for a unit acceleration step. The radius of the sphere that enclosed 90% of these acceleration errors was taken to be the acceleration magnitude to which the tracking loops were tuned. Although a 90% level may not seem very robust, it should be remembered that tracking loop errors greater than 90 deg do not necessarily cause loss of lock.

For the four qualities of IMU studied, the radius of the acceleration sphere and the corresponding bandwidths are shown in Table 2.6. The error due to a unit acceleration step is given in parts per million.

Table 2.6. The Residual Acceleration Error and Corresponding Bandwidth for the Carrier Tracking Loop for Four IMU Qualities.

<table>
<thead>
<tr>
<th>IMU</th>
<th>Residual Error (90%) Due to Scale Factor and Misalignment (ppm)</th>
<th>Acceleration Seen by Tracking Loop Due to 6-g Acceleration Step</th>
<th>Required Carrier Tracking Loop Bandwidth</th>
</tr>
</thead>
<tbody>
<tr>
<td>No inertial aiding</td>
<td>Not applicable</td>
<td>6 g</td>
<td>18 rad/s</td>
</tr>
<tr>
<td>10 nmi/h</td>
<td>1880</td>
<td>0.011 g</td>
<td>0.77</td>
</tr>
<tr>
<td>1 nmi/h</td>
<td>431</td>
<td>0.0026 g</td>
<td>0.37</td>
</tr>
<tr>
<td>0.5 nmi/h</td>
<td>311</td>
<td>0.0019 g</td>
<td>0.32</td>
</tr>
<tr>
<td>0.2 nmi/h</td>
<td>225</td>
<td>0.0014 g</td>
<td>0.27</td>
</tr>
</tbody>
</table>

The quantity in the second column is the radius of the sphere that encloses 90% of the errors. The quantities in the third column are that error times the 6-g acceleration step, and the quantities in the third column are the required bandwidth as determined by Eq. 2.1. For example, the bandwidth for the 10-nmi/h system was computed as shown below.
For the purposes of the analysis, it was declared that the receiver had lost lock if the carrier phase error exceeded 90 deg or if the signal-to-noise ratio dropped below 19 dB. Loss of lock for the code phase was declared if the tracking error was greater then 1/2 chip (50 ns) or if the signal-to-noise ratio dropped below 18 dB. Conversely, reacquisition was dependent on achieving a signal-to-noise ratio of at least 21 dB for the code and 22 dB for the carrier for a required amount of time. The required time depends on the uncertainty in the range and range rate to each satellite and the rate at which each code phase and frequency combination could be searched.

At the given signal level, a 20-ms integration period should be adequate for accumulating signal energy. The size of the phase shift between 20-ms search intervals was 36 deg corresponding to 30 m. The size of the frequency bandwidth was 50 Hz corresponding to 10 m/s. The approximate time required to search over this position-frequency space ($\pm \sigma$) is given below.

$$\Delta T = 0.020 \left[ \frac{2\sigma_{\text{range}}}{30} \right] \times \left[ \frac{2\sigma_{\text{range rate}}}{10} \right]$$

Some additional time must be added to allow for receiver moding. That is, the search process must be halted, and the receiver tracking loops cycled several times with an adequate signal-to-noise /jammer ratio.

### 2.6 Navigation Performance for Four Missions

Four missions were studied. The purpose of each of these mission scenarios was to observe the effects of jamming on loosely and tightly coupled INS/GPS systems and to observe the effect of IMU quality on tightly coupled systems. For the first scenario, the loss of lock and reacquisition for each of four satellites was spaced out so that the behavior of the navigation solution could be observed for extended periods of time between each loss. The other missions consisted of: 1) an aircraft flying past a jammer so that it loses lock then reacquires satellites as the jammer recedes into the distance, 2) an aircraft approaching a jammer head on, and 3) a helicopter operating in the vicinity of a jammer.

#### 2.6.1 Sequential Outage

In this scenario, the loss of lock on satellite carrier and code phase was forced at 3-min intervals. Loss of code lock began after the fourth carrier tracking loop lost lock. Thus, the sequence began with loss of lock on a single carrier signal and ended with the loss of lock of the fourth code loop. Two variations in the study were considered at this point. In one of these, the signals were reacquired in inverse order after a total outage of 20 min. In the other variation, the mission was continued for 84 min using inertial measurements without the aid of GPS.

The behavior of the horizontal velocity error for the loosely and tightly coupled 1 nmi/h systems is shown in Figures 2.2 and 2.3.

The first loss of carrier tracking occurs at 360 s. The first loss of code at 1020 s. One feature of this loosely coupled system is that it does not form a navigation solution if fewer than four satellites are in lock. Thus, the immediate rise in velocity error begins at this point in the bottom graph (loosely coupled system). In contrast, the increase in velocity error for the tightly coupled system is somewhat delayed. The sequence of code reacquisition begins at 2760 s. Since the tightly coupled system makes immediate use of the first code measurement, the step improvement in velocity is seen at that time. The correlations between position and velocity in the Kalman filter cause the decrease in velocity error, even though it is a range measurement that has been made. Each successive code loop reacquisition causes a step improvement in the velocity accuracy. In contrast, the loosely coupled system does not get the benefit of the recently reacquired code phase until four satellites are in lock. At this point (in the lower graph), the improvement in velocity accuracy is recognized easily.
With four satellites in lock, the loosely coupled system yields perfectly acceptable navigation performance. The response to jamming the tightly coupled system is somewhat better. The maximum horizontal position error for each system studied is shown in Figure 2.4.
If the navigation system is denied, GPS measurements for 84 min, the horizontal position errors grow to the levels shown in Figure 2.5.

![Position Error (m)](image)

**Figure 2.5.** Horizontal position errors 1 Schuler period after loss of last satellite.

In addition to the obvious correlation of navigation error to IMU quality, we can make the following general observations about these results. The tightly coupled 1.0-nmi/h system does perform better than the loosely coupled system. This is due to two factors: 1) the tightly coupled system makes use of measurements even when fewer than four satellites are in lock, and 2) the calibration of the inertial instruments is somewhat better with the tightly coupled system. This performance difference diminishes with time. A very long time after the last GPS measurement, the performance of the tightly coupled and loosely coupled systems would be identical -- that of a 1-nmi/h system. The performance of the 1.0-nmi/h system is about 10 times better than that of the 10.0-nmi/h system at the end of the 20-min blackout interval. However, at the end of 84 min, the 1.0-nmi/h system has only drifted to a 1131 m error. The 10-nmi/h system has drifted to close to 18000 m. This simply reflects the fact that the major source of error for the 10-nmi/h system is uncalibratable random errors.

### 2.6.2 Jammer Flyby

For this scenario, an aircraft flies by a jammer, thus losing and regaining lock in a somewhat more realistic fashion. A jammer was placed on the ground near the midpoint of the trajectory to cause an approximate 20-min outage. In contrast to the previous situation in which the period of the outage was specified, in this scenario, the actual loss of lock will be determined by the signal-to-noise ratio for each satellite. Reacquisition will be determined by the growing uncertainty in range and range rate to each satellite. The period of outage will also be a function of the bandwidth of the carrier tracking loops for each of the receivers. We will, again, observe position and velocity error growth as GPS measurements are lost.

Table 2.7 shows the number of range intervals to be searched for each satellite at the time the signal-to-noise threshold rose above 21 dB. In no case did the error growth in velocity cause the range rate uncertainty to any satellite to be greater than 10 m/s (one Doppler shift interval).
Table 2.7. Search for Range Phase as a Function of IMU Quality.

<table>
<thead>
<tr>
<th>Navigation System</th>
<th>Range (Code Phase Intervals to be Searched)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Satellite Identification</td>
</tr>
<tr>
<td></td>
<td>3</td>
</tr>
<tr>
<td>Tightly coupled 10 nmi/h</td>
<td>73</td>
</tr>
<tr>
<td>Loosely coupled 1.0 nmi/h</td>
<td>8</td>
</tr>
<tr>
<td>Tightly coupled 1.0 nmi/h</td>
<td>7</td>
</tr>
<tr>
<td>Tightly coupled 0.5 nmi/h</td>
<td>6</td>
</tr>
<tr>
<td>Tightly coupled 0.2 nmi/h</td>
<td>6</td>
</tr>
</tbody>
</table>

For a search time of 20 ms/chip (code phase interval), the better IMUs hold the search time to 0.2 s. The 10-nmi/h inertial system holds the search time to about 0.8 to 1.6 s. These numbers only reflect the error growth in position (and velocity) uncertainty and assume that the entire $1\sigma$ search area must be searched before lock on is achieved. Sometimes the signal will be found sooner, and of course, 32% of the time, it will be outside the $1\sigma$ bounds and require a longer search. Unfortunately, these results cannot be generalized. The placement of the jammer, its signal strength, the antenna orientation, and gain pattern are unique to the scenario and can only be considered typical.

The blackout period as a function of IMU architecture and IMU quality is shown in Figure 2.6.

![Figure 2.6. GPS loss of lock as a function of IMU architecture and quality.](image)

(Note the vertical scale does not begin at zero. The difference is not so striking, as the graph seems to indicate.) For this particular scenario, the performance difference is due to better calibration of the inertial instruments rather than jamming resistance. Even for the best IMU, the blackout time is reduced by only about 2 min from the 20-min blackout experienced by the loosely coupled receiver with full (unaided) bandwidth. Once again, it is difficult to generalize from these results.

For this jammer flyby scenario, the horizontal position errors (root-sum-squared (rss) $1\sigma$) just prior to reacquisition are shown in Figure 2.7.
The ratio of error level between the 10-nmi/h and the 1-nmi/h tightly coupled systems (3300:260) is greater than the 10 to 1 ratio implied by their characterization. Noise is a big error source in the 10-nmi/h system and cannot be calibrated by the GPS measurements as can biases and scale-factor errors. Thus, the better IMUs perform better yet when they are calibrated continuously by in-flight GPS measurements. The tightly coupled 1-nm/h system benefits somewhat more by the GPS in-flight calibration than the loosely coupled system.

**2.6.3 Head-on Approach to Jammer**

This scenario is meant to simulate the navigation performance of a fighter-bomber mission in which there is a jammer at the target. After take-off, the aircraft climbs to 40,000 ft, dodges a surface-to-air missile, then dives down to 200 ft to get below radar detection and to avoid GPS jamming. On approaching the target area, the aircraft then climbs to a few thousand feet to locate the target, then releases the bomb. The jammer at the target overwhelms all variations of IMU quality and architecture as soon as the aircraft climbs above its horizon. The time interval between loss of lock and bomb release is about 159 s for the loosely coupled system and 153 s for the tightly coupled systems. Figure 2.8 shows the position error at bomb release for the five navigation systems.

![Figure 2.7. Horizontal position error just prior to reacquisition.](image)

![Figure 2.8. Position error at bomb release after about 2.7 min of free inertial navigation.](image)
In contrast to the previous scenarios, the IMU performance shortly after loss of lock is shown here. The tightly coupled 10 nmi/h system has better performance than this particular loosely coupled system when GPS measurements are available. After this short a time, 2.7 min, this advantage has not yet been lost. Another contributor to the difference is that the tightly coupled systems resisted the jamming for about 6 s longer than did the loosely coupled system.

2.6.4 Helicopter Performance in Jammer Vicinity

This scenario is meant to depict a helicopter on a scouting mission. The helicopter closely follows the terrain in order to avoid detection. The resulting flight profile has high levels of acceleration and jerk, which caused occasional momentary loss of carrier lock. No effect on mission performance can be seen.

The jamming scenario was simplified for this mission. GPS measurements were available until on-board estimates of IMU calibration and alignment had reached steady state. At that point, GPS was assumed to be jammed. The mission continued for another 19 min. In a variation from the previous scenarios, the navigation system of the helicopter was augmented with ground speed Doppler measurements. These Doppler measurements yield velocity in body coordinates. It will be seen that these measurements make a considerable difference in navigation performance after GPS is lost. The error model for the Doppler measurements is given in Table 2.8.

Table 2.8. Error Model for Doppler Ground Speed Measurements.

<table>
<thead>
<tr>
<th>Error Source</th>
<th>Vertical (1σ)</th>
<th>Horizontal (1σ) (two components)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bias</td>
<td>0.05 nmi/h</td>
<td>0.1 nmi/h</td>
</tr>
<tr>
<td>Scale factor</td>
<td>0.1%</td>
<td>0.25%</td>
</tr>
<tr>
<td>Misalignment</td>
<td>2.0 mrad</td>
<td>2.0 mrad</td>
</tr>
</tbody>
</table>

At the end of the mission, the task of the helicopter is to define coordinates of a target at some distance (8 km) from its own position. The error in target coordinates, \( \delta r_{tgt} \), is thus due to a combination of helicopter location error, \( \delta r_{helicopter} \), and IMU misalignment, \( \delta \alpha \).

\[
\delta r_{tgt} = \delta r_{helicopter} + \delta \alpha \times r
\]

where \( r \) is the vector from helicopter to target.

Figure 2.9 shows the error in helicopter position and target location as a function of two IMU qualities when no ground-speed Doppler measurements are included in the navigation solution.

As seen in an earlier scenario, the ratio of the errors between the 10 nmi/h and the 1 nmi/h navigation system, 2750:192 in this case, is greater than the characterization ratio, 10:1. The pointing error is negligible compared with the position error so that the target location errors and the aircraft position errors are essentially the same.

Figure 2.10 shows the same errors when the navigation solution is aided with ground-speed Doppler measurements. Results for both an INS/GPS system and for an INS/GPS system supplemented with Doppler ground-speed measurements are shown.
As expected, the Doppler ground-speed measurements slow the error growth that is seen with the free inertial system. These errors in these velocity measurements integrate into growing position errors so they are not equivalent to GPS, which provides position as well as velocity. But they provide much better results than the inertial instruments whose measurements must be integrated twice before yielding position. The improvement with the Doppler ground-speed sensor is dramatic. Note that when aided by these measurements, the performance of the 10-nmi/h system is nearly the same (50% greater target location errors) as that of the 1-nmi/h system.
2.8 Summary of Comparison Results – Loosely Coupled vs. Tightly Coupled

This comparison has illustrated several features of INS/GPS systems that are true in many cases, but there is some danger in drawing general conclusions because the flight profiles and jamming scenarios are quite specific. A particular flight profile may allow more or less in-flight calibration, depending on aircraft maneuvers. These differences can be minimized by including maneuvers whose specific purpose is in-flight calibration and alignment. Jamming scenarios, however, are more difficult to characterize in a general way. Jammers can be on the ground, in which case, they are shadowed by the terrain for low-altitude approaches. They could also be airborne, in which case, their effective range will be greater, but for which their signal strength will grow with closing distance uncomplicated by shadowing considerations. Furthermore, there may be focused jammers and as a countermeasure to jamming, receiver antennas whose gain can be made a function of direction. All these variables make it difficult to generalize about how much longer a tightly coupled system will be able to maintain lock on the GPS signals. Perhaps the most general statement that can be made is to state the improvement in decibels in signal-to-noise (jammer) ratio that inertially aided receivers achieve.

The in-flight calibration and alignment of the tightly and loosely coupled receivers is simpler to assess. As in this study, loosely and tightly coupled architectures can be proposed. The resulting performance after loss of lock can then be assessed by either Monte Carlo techniques or, as in this study, by linearized covariance analysis.

After doing the analysis and observing simulation results, the following cautious assertions can be made:

1. When GPS is available, its measurements dominate navigation performance. The steady-state navigation error will be reduced by inertial aiding, which simply considered, allows GPS measurement noise to be “averaged out.” Improvement of steady-state error with improving inertial quality is not as dramatic.

2. Tight coupling is superior to loose coupling for maintaining lock in a jamming environment, but the gain is hard to quantify, except by improvement in the signal-to-noise (jammer) ratio.

3. Better inertial instruments gain more from in-flight alignment and perform better after GPS is lost. This is because poorer instruments in general have larger proportions of uncalibratable noise.

4. For short time intervals after GPS loss, coupling architectures can make a difference in performance (because they affect calibration and alignment quality).

5. In the long run, basic IMU quality will dominate navigation accuracy due to instrument noise and loss of calibration accuracy.

Finally, it was shown that there is a dramatic difference in jammed performance if Doppler ground-speed measurements were available.

3.0 Deep Integration vs. Tightly Coupled Performance Comparison

The inertial sensor error model used was representative of a particular Microelectromechanical System (MEMS) IMU capability. Rms accelerometer errors were characterized by 1-mg bias stability, 100-ppm scale-factor stability and 1-cm/s/√h random walk. Rms gyro errors were 10-deg/h bias stability and 0.03-deg/√h random walk. All stability errors were modeled as first-order Markov processes with a time constant of 5 min, which is representative of expected in-flight error characteristics. Accelerometer and gyro input axis nonorthogonalities of 1 mrad, rms were assumed and were treated as fixed biases.

A full 6-degree-of-freedom simulation was used with four satellites continuously in view. The navigation state vector consisted of 3 components of position, velocity, inertial sensor stability errors (bias, scale factor, and misalignment for both accelerometers and gyros), user clock, and clock rate. Clock errors were treated as biases. Satellite ephemeris errors were accounted for by including an unestimated range bias of two meters, rms.

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1 The material in this section is from References 3 and 4. “Deep integration” is trademarked by Draper Laboratory.
The navigation algorithms are nonlinear. Thus, it was not possible to perform an accurate error covariance analysis based on linearization. As a consequence, the results in this section are based on Monte Carlo analysis.

The measurements from all correlators were processed simultaneously, while the measurements from each satellite were processed sequentially. An ideal correlation function was assumed in the navigation algorithms, with \( R_c(\tau) = 1 - |\tau| \) for \(|\tau| \leq 1\) and \( R_c(\tau) = 0 \) for \(|\tau| > 1\). A correlator spacing of \( \frac{1}{2} \) chip was used throughout. Two types of jammers were assumed: 1) wideband Gaussian jammer with a 20 MHz bandwidth, and 2) narrowband jammer with a 1 kHz bandwidth. Jammer outputs were generated using a first-order Markov process driven by pseudorandom Gaussian noise.

In order to assess performance relative to conventional systems, a tightly coupled INS/GPS system was also simulated. The simulation model assumed a GPS receiver capable of calculating pseudo-range and delta-range. The receiver outputs and the simulated MEMS sensor outputs were fed to an INS/GPS integration filter. This filter was mechanized as a standard extended Kalman filter and used the same navigation state vector as the deeply integrated system. The receiver was velocity aided using the velocity components of the state vector estimate. The tightly coupled receiver filter bandwidth was 0.1 Hz while in State 3 tracking. Code loop loss-of-lock was assumed to occur at J/S = 54 dB. Above this threshold, GPS data were not used and free inertial navigation was assumed.

3.1 Constant Wide-Band Gaussian Jamming

Performance comparisons between the deeply integrated and tightly coupled systems were first conducted for two scenarios in which the navigator is subject to a constant wide-band Gaussian jamming. This allows the assessment of the additional loss-of-lock capability due to deep integration in a relatively straightforward manner. Dynamic maneuvers were simulated: body-frame specific force was pulsed between 0 and \( \pm 1 \) g along all axes; pulsewidth was 10 s with a period of 90 s.

Results for the first scenario are shown in Figure 3.1 using 27 Monte Carlo runs. J/S was maintained at 30 dB over the first 60 s of flight and then instantaneously switched to a higher value and held there for the remainder of the 5-min flight. Initial rms navigation errors were 30 m and 1 m/s along each axis. Initial rms clock errors were 20 m and 2 m/s. J/S levels used were 40 to 80 dB in increments of 5 dB, as shown outside the right edge of the figures. The deeply integrated system was able to maintain code lock up to 65 dB J/S, an improvement of approximately 15 dB over the tightly coupled system. Note also that the deeply integrated system achieves lower rss error during the 60-s initialization phase.

![Figure 3.1. Navigation performance comparison: first scenario.](image-url)
The second scenario used a constant value of J/S over a 7-min flight. Initial rms position and velocity errors were 2 m and 1 m/s along all axes. The rms clock errors were 1 m and 0.1 m/s. J/S was varied between 50 and 80 dB in 5-dB increments. The results are shown in Figure 3.2, using 35 Monte Carlo runs and for both 10-deg/h and 1-deg/h INS error models. The results again indicate a significant improvement in loss-of-lock capability due to deep integration. This improvement is shown quantitatively in Figure 3.3, in which the results of Figure 3.2 are replotted vs. antijam (A/J) improvement. A/J improvement is in excess of 10 dB at all values of rss error and exceeds 15 dB for rss error greater than 35 m. Note that the A/J improvement also increases when higher quality inertial sensors are employed. This is primarily due to the enhanced in-flight error calibration capability of the deeply integrated system.

Figure 3.2. Navigation performance comparison: second scenario.

Figure 3.3. A/J improvement due to deep integration.
3.3 Precision Guided Munition Scenario

The performance of the deeply integrated navigation system was evaluated for a precision guided munition (PGM) scenario in which the target was at a range of 63 nmi. The altitude profile is plotted in Figure 3.4. A single wideband Gaussian jammer was placed 5 nmi in front of the target in an attempt to simulate a worst-case scenario for a single jammer. This placement gives maximum J/S prior to final target approach with a resultant loss of navigation system performance just prior to target impact. The J/S history for a 100 W jammer is shown in Figure 3.5.

![Figure 3.4. PGM altitude profile.](image)

![Figure 3.5. PGM scenario: J/S vs. time.](image)

Performance was evaluated by varying the jammer power from 1 W to 10 kW. A total of 25 Monte Carlo runs was made at each power level. Initial rms navigation errors were 10 m and 0.2 m/s per axis; initial rms clock errors were 10 m and 0.2 m/s. The CEP at target impact is plotted vs. jammer power for wideband jamming in Figure 3.6 and for narrowband jamming in Figure 3.7. Comparing the results in the figures, it can be seen that the deeply integrated system offers significant improvement over the traditional tightly coupled system for both wideband and narrowband jamming. As an example, a 100-W wideband jammer results in a CEP of 11 m for the deeply integrated system, compared with a CEP of 120 m for the tightly coupled system. If the jammer power is reduced to 10 W, the CEP values are 2.6 m for the deeply integrated system and 71 m for the tightly coupled system.

![Figure 3.6. CEP vs. jammer power: wideband jammer.](image)

![Figure 3.7. CEP vs. jammer power: narrowband jammer.](image)

A/J improvement capability may be quantified by comparing jammer power at a constant value of CEP. The resulting improvement in A/J capability due to deep integration can be seen in Figure 3.8. For wideband jamming, improvements of at least 15 dB are seen for CEP values ranging from 6 to 120 m. For narrowband
jamming, improvements of at least 15 dB are seen for CEP values ranging from 4 to 80 m. Improvement is seen to decrease as the CEP decreases below 10 m. In this case, the decrease in CEP results from a decrease in jammer power, and the tightly coupled system tends to maintain lock with higher probability as the jammer power decreases. In the limit as the jammer power approaches zero, the tightly coupled system approaches efficient operation, and both systems give comparable performance. The improvement is also seen to decrease as the CEP increases beyond 100 m. In this case, the increase in CEP results from an increase in jammer power and the tracking quality of the deeply integrated system begins to degrade. In the limit as the jammer power increases without bound, the deeply integrated system can no longer maintain lock, and both systems are operating in a free inertial mode where the CEP is determined solely by initial navigation errors and inertial sensor errors.

![Figure 3.8. A/J improvement due to deep integration.](image)

3.3 Final Comment on Deep Integration Comparison

As shown in Reference 5, every 20 dB increase in A/J improvement in the INS/GPS system requires that the jammer increase its power by a factor of 100 to maintain the same effectiveness as a jammer. Yet the increase in jammer power makes its detection and attack much more probable. Thus, the potential benefits of deep integration are substantial in both comparisons reported here. Additional comparisons can be found in Gustafson and Dowdle, “Deeply Integrated Code Tracking,” ION GPS/GNSS Portland, OR, 2003.

4.0 Concluding Remarks

This paper has presented several options for the integration of INS and GPS systems in order to benefit from the advantages of each system. As been shown, if the integration level between the two systems increases, the benefits also generally increase. The comparison of deeply integrated vs. closely coupled indicates that the deeply integrated approach will likely be dominant in the future.

References:


Abstract
An inertial navigation system (INS) exhibits relatively low noise from second to second, but tends to drift over time. Typical aircraft inertial navigation errors grow at rates between 1 and 10 nmi/h (1.8 to 18 km/h) of operation. In contrast, Global Positioning System (GPS) errors are relatively noisy from second to second, but exhibit no long-term drift. Using both of these systems is superior to using either alone. Integrating the information from each sensor results in a navigation system that operates like a drift-free INS. There are further benefits to be gained depending on the level at which the information is combined. This presentation will focus on integration architectures, including "loosely coupled," "tightly coupled," and "deeply integrated" configurations. (Deep integration is trademarked by Draper Laboratory.) The advantages and disadvantages of each level of integration will be listed. Examples of current and future systems will be cited. Examples of current and future systems will be cited.

1.0 Introduction
INS/GPS integration is not a new concept [Refs. 1, 2, 3, 4]. Measurements of noninertial quantities have long been incorporated into inertial navigation systems to limit error growth. Examples shown in Figure 1.1 are barometric “altitude” measurements, Doppler ground speed measurements, Doppler measurements to communications satellites, and range measurements to Omega stations.

Although GPS provides a deterministic solution for both position and velocity, it has its own shortcomings. Among them are: low data rate (typically 1 Hz), susceptibility to jamming (even unintentional interference), and lack of precision attitude information.

GPS and inertial measurements are complementary for two reasons. Their error characteristics are different and they are measures of different quantities. GPS provides measures of position and velocity. An
accelerometer measures specific force. The gyroscopes provide a measure of attitude rate, and after initial alignment, they allow the accelerometer measurements to be resolved into a known coordinate frame.

GPS position measurement accuracy is limited due to a combination of low signal strength, the length of the pseudo-random code, which is about 300 m, and errors in the code tracking loop. Multipath, the phenomenon whereby several delayed copies of the signal arrive at the antenna after being reflected from nearby surfaces, is a source of correlated noise, especially for a moving vehicle. GPS position measurements also have constant or slowly changing biases due to satellite ephemeris and clock errors. These biases are bounded and are not integrated since they are already at the position level.

GPS velocity (position difference) measurements are also noisy, again due to variations in signal strength, the effects of changing multipath, and user clock instability.

In contrast, the accelerometers in an inertial navigation system measure specific force. They have relatively low-noise characteristics when compared with GPS measurements. The signals must be compensated for gravity and integrated twice before providing position estimates. This fundamental difference in radio navigation measurements and inertial measurements is a clue to the difference in the behavior of INS and GPS navigators.

Figure 1.2 shows accelerometer noise (and its first two integrals). The noise level was specified at 56 $\mu$g/$\sqrt{\text{Hz}}$, typical of a 10-nmi/h inertial system. The accelerometer noise itself is shown in the top graph.

![Figure 1.2. Accelerometer noise and its first two integrals.](image)

In these graphs, the accelerometer is read every 20 ms for 20 s. The integral of acceleration, the middle graph, shows the familiar “random walk” behavior of the integral of random noise. The dotted lines are the $1\sigma$ expected errors in the random walk. The second integral, the bottom graph, corresponds to position. (Units are metric: m, m/s and m/s$^2$)

GPS receivers typically produce solutions at 1 Hz or 10 Hz. The data bit rate of 50 Hz sets a “natural” minimum of 20 ms between position and velocity determinations. The middle graph in Figure 1.3 shows random noise in a set of measurements. The standard deviation of the velocity measurement is 0.01 m/s, typical of a good GPS receiver and strong signals in a benign environment.
Back differencing these measurements to match the 50-Hz accelerometer output results in the noisy acceleration measurements as shown in the top graph of the figure. (Again, units are metric: m, m/s and m/s²) The bottom graph of Figure 1.3 shows the first integral of the velocity over 1-s intervals as it might be used for carrier track smoothing of the GPS position measurement. The circles show the value of the integral after each 1-s interval. Thus, they indicate the error in the position difference from one GPS measurement (at 1 Hz) to the next. It is considerably smaller than the measurement error in the position measurement itself, thus the impetus for carrier track smoothing. The position measurement keeps the integral of the carrier track from diverging in the same “random walk” fashion as the integral of accelerometer noise. Users will, quite naturally, want the features of both systems -- the high bandwidth and autonomy of inertial systems, and the long-term accuracy of GPS.

Table 1.1 summarizes the features and shortcomings of inertial and GPS navigation systems.

The goal of INS/GPS integration, besides providing the redundancy of two systems, is to take advantage of the synergy outlined as follows:

1. The conventional approach to aiding the receiver’s carrier and code tracking loops with inertial sensor information allows the effective bandwidth of these loops to be reduced, even in the presence of severe vehicle maneuvers, thereby improving the ability of the receiver to track signals in a noisy environment such as caused by a jammer. The more accurate the inertial information, the narrower the bandwidth of the loops that can be designed. In a jamming environment, this allows the vehicle to more closely approach a jammer-protected target before losing GPS tracking.² A minimum of a factor of 3 to 4 improvement in approach distance is typical. A “deeply-integrated” approach to aiding will be shown to be even more robust. Outside a jamming environment, INS data provide high bandwidth accurate navigation and control information and allow a long series of GPS measurements to play a role in the recursive navigation solution. They also provide an accurate navigation solution in situations where

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1 It is not necessary to break the velocity measurement into 20-ms intervals. As suggested by Cox et al. [Ref. 1] it is possible to track the carrier phase continuously from satellite rise to satellite set. Another method for extracting a less noisy velocity would be to recognize that the error at the beginning of one interval is the negative of the error at the end of the preceding interval (if carrier tracking is continuous across the data bit).

2 Representative jammers are given in Reference 4.
“GPS only” navigation would be subject to “natural” short-term outages caused by signal blockage and antenna shading.

Table 1.1. Inertial and GPS Attributes and Shortcomings.

<table>
<thead>
<tr>
<th>Attributes</th>
<th>Shortcomings</th>
</tr>
</thead>
<tbody>
<tr>
<td>GPS</td>
<td>Low data rate</td>
</tr>
<tr>
<td></td>
<td>No attitude information</td>
</tr>
<tr>
<td></td>
<td>Susceptible to jamming (intentional and unintentional)</td>
</tr>
<tr>
<td>INS</td>
<td>Unbounded errors</td>
</tr>
<tr>
<td></td>
<td>Requires knowledge of gravity</td>
</tr>
<tr>
<td>Errors are bounded</td>
<td></td>
</tr>
<tr>
<td>High data rate</td>
<td></td>
</tr>
<tr>
<td>Both translational and rotational information</td>
<td></td>
</tr>
<tr>
<td>Self-contained (not susceptible to jamming)</td>
<td></td>
</tr>
</tbody>
</table>

2. The inertial system provides the only navigation information when the GPS signal is not available. Then inertial position and velocity information can reduce the search time required to reacquire the GPS signals after an outage and to enable direct P(Y) code reacquisition in a jamming environment.

3. Low-noise inertial sensors can have their bias errors calibrated during the mission by using GPS measurements in an integrated navigation filter that combines inertial system and GPS measurements to further improve the benefits listed under (1) and (2). The accuracy achieved by the combined INS/GPS system should exceed the specified accuracy of GPS alone. The synergistic benefits of combining inertial data with GPS data as described in the previous paragraph are notionally shown in Figure 1.4.

4. Having inertial instruments at the core of the navigation system allows any number of satellites to play a role in the solution.

The accuracy of the solution, the resistance to jamming, and the ability to calibrate the biases in low-noise inertial system components depend on the avionics system architecture. There have been many different system architectures that have been commonly implemented to combine the GPS receiver outputs and the
INS information, thus obtaining inertial sensor calibration, to estimate the vehicle state. Different INS/GPS architectures and benefits will be discussed in the following section.

2.0 Alternate INS/GPS Architectures

Four architectures will be discussed in this paper: separate systems, loosely coupled, tightly coupled, and deeply integrated systems. Several variations of loosely coupled and tightly coupled systems will be shown.

2.1 Separate Systems

The simplest way to get the features of both systems is to simply have both navigation systems integrated only in the mind of the user. Only slightly more complex would be to simply add a correction from the GPS to the inertial navigation solution. Figure 2.1 illustrates such a system.

![Figure 2.1. Separate GPS and INS systems with a possible INS reset.](image)

This mode of operation or coupling has the advantage of leaving the two systems independent and redundant. But as the Inertial Measurement Unit (IMU) drifts, the inertial solution becomes practically useless.

By using a GPS “reset” or correction, the inertial system errors are kept bounded, but after the first reset, the INS solution is no longer independent of the GPS system. Of course, the corrections could be monitored for reasonableness to prevent the contamination of the inertial solution with grossly incorrect GPS measurements should they occur. Even if not independent, the systems do remain redundant in the sense that they both still have dedicated displays, power supplies, etc., so that the failure of one does not affect the other or leave the vehicle with no navigator.

Inertial system resets provided the first mechanization for the U.S. Space Shuttle GPS integration. The Space Shuttle has a ground uplink capability in which the position and velocity are simply set to the uplinked quantities. For minimum change to the software, the GPS system simply provides a pseudo ground

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“Coupled” here refers to combining data from the GPS and INS systems into a single navigation solution. When retrofitting older aircraft with new navigation systems, there is often a problem with space and with power and data connections. For these reasons, it can be desirable to include GPS in the same box with the inertial navigator. This repackaging will be referred to as “embedding.”
uplink. To make a minimal impact on existing software and hardware is a common rationale for the more loosely coupled systems.

In summary, this architecture offers redundancy, bounded position and velocity estimates, attitude and attitude rate information, high data rates for both translational and rotational information suitable for guidance and control functions, and (for existing systems) minimum impact on hardware and software.

2.2 Loosely Coupled
Most often, discussions of INS/GPS integration focus on systems that are more tightly coupled than the system described in the previous section. This will be true of the remaining architectures. Redundancy and solution independence can be maintained, but we will see more benefits from coupling than the simple sum of inertial and GPS navigation features. New software will be required, an integration filter for example.

2.2.1 Loosely Coupled - Conservative Approach
Figure 2.2 shows one version of a loosely coupled system. In this system, the functional division could correspond to the physical division with the GPS in a box, the INS in a box, and the computer that combines the navigation solutions in yet another box. Only low rates are required for data links between the boxes. Of course, the three functions could be packaged together if desired.

![Figure 2.2. A loosely coupled INS/GPS navigation system.](image)

Simplified diagrams for each of the functions are shown. The following paragraphs consist of a high-level description of the operation of a receiver and inertial navigator. It is assumed that the reader has some familiarity with these sensors; thus, the discussion is intended to serve more as a reminder of pertinent features rather than a tutorial.

The receiver diagram shows signals coming into the radio frequency “front end” of the receiver. They are down converted to baseband and fed into the correlators. Meanwhile, a duplicate of the signal is generated internally in the receiver. In fact, three (or more) copies are generated. One of these copies is supposedly time synchronized so that it arrives at the “prompt” correlator at exactly the same time as the signal from the antenna. The other copies are intentionally either a little early or a little late compared with what is expected from the satellite. These copies are sent to the early and late correlators. The magnitude of the early and late correlations, indicated by [+,-] in the figure, is given to the code tracking function. The difference in these magnitudes is an indication of the timing error (and thus range error). This error signal is fed back into the code generator, which makes a correction to the code phase timing. This process is repeated as long as the signal is present. At some point, the phase error will be driven down to an acceptable
level, and the code will be declared “in lock.” While “in lock,” the time difference between the broadcast of the signal and the receipt of the signal are a measure of the pseudo-range.

Similarly, the in-phase and quadrature signals are fed into the carrier tracking function. The arctangent of these two signals is a measure of the carrier tracking error. This signal is fed back to the numerically-controlled oscillator (NCO), and its frequency is adjusted accordingly. It might be noted that the carrier tracking loop is typically of third order, allowing it to “perfectly track” signals with constant range acceleration. Note that the carrier loop (when it is “in lock”) “aids” the code loop as indicated by the arrow labeled $\Delta p$. In this mode, the code tracking loop can be of first order.

For this architecture, the receiver only uses INS data for the purpose of aiding in acquisition. Knowing the position and velocity of the vehicle enables the code generator and oscillator to make good initial guesses of the frequency and code phase of the incoming signal. The search time during acquisition can be reduced significantly depending on the accuracy of these estimates.

The output of the two tracking loops is an estimate of the range and range rate between the vehicle and the satellite. Range and range rate estimates from four satellites are sufficient to resolve the vehicle position, velocity, receiver clock bias, and receiver clock drift rate. For some receivers, these deterministic quantities are the ultimate receiver output. However, receivers that are expected to operate in a dynamic environment use a polynomial Kalman filter to estimate position, velocity, and acceleration, and clock bias and clock drift rate.

A (strapdown) INS diagram is shown at the bottom of the figure. Raw measurements from the accelerometers and gyroscopes are compensated using a priori values, or values derived from another mode of operation (e.g., a calibration and alignment mode). The gyroscope output is used to maintain the rotational state of the vehicle. Angular rates are integrated into either a quaternion or matrix, which relates the vehicle attitude to some reference coordinate system (e.g., local level). Corrected $\Delta V$'s are rotated into this coordinate system and integrated to maintain the translational state: position and velocity.

The INS/GPS integration function is shown in the middle diagram of the figure. It receives corrected inertial measurements, $\Delta \Theta$' and $\Delta V$' from the INS and position and velocity measurements from the GPS receiver. The 1-Hz GPS measurements, coming from a Kalman filter, are highly correlated. The second Kalman filter in this “cascaded” architecture handles this problem by only incorporating these measurements every 10 s. The 10-s interval allows each position/velocity measurement to be more or less independent of the previous measurements. A performance comparison between this loosely coupled architecture and a tightly coupled architecture is given in Reference 5. Note that the integration Kalman filter includes calibration and alignment estimates that provide in-flight improvement of the INS calibration and alignment. This conservative approach to coupling yielded surprisingly good results in estimating these gyro and accelerometer parameters.

Table 2.1 summarizes the functions of the three components of the system. Table 2.2 lists the attributes of the system.

<table>
<thead>
<tr>
<th>Component</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>GPS</td>
<td>The Kalman filter estimates: Position, velocity, acceleration Clock bias, clock drift</td>
</tr>
<tr>
<td>INS</td>
<td>The INS provides: Position, velocity, acceleration Attitude, attitude rate</td>
</tr>
<tr>
<td>Integration Filter</td>
<td>The integration filter estimates: Position, velocity Attitude corrections, instrument corrections</td>
</tr>
</tbody>
</table>
We distinguish between jamming resistance and mitigation against jamming. By the latter term, we simply mean that the inertial bias and scale-factor parameters will be better calibrated so that if the GPS signal is lost, the INS/GPS solution (receiving only inertial data) will be accurate for longer than otherwise.

### 2.2.2 Loosely Coupled-Aggressive Approach

Figure 2.3 shows possible variations in what may still be considered a loosely coupled architecture. Inertial aiding of tracking loops has not yet been introduced, and the integration filter still uses position and velocity data rather than pseudo-range and range rate. Additional data transfer beyond that of the previous architecture is indicated by heavy lines. Either one or the other or both data transfers are viable options.

The first of these data transfers is of the corrected velocity increment $\Delta V'$ from the INS/GPS module to be used in the GPS module to propagate the solution between measurements. This provides a vast improvement in dynamic situations. Otherwise, the propagation must be done using the acceleration...
estimate from the GPS Kalman filter itself. This acceleration, although a component of the filter state, is derived by back differencing the velocity. Figure 1.3 showed the level of acceleration noise inherent in this operation. It is true that the filter offers some “smoothing.” However, it cannot offer much due to the process noise, which must be added in the dynamic aircraft environment. There is a requirement by the U.S. GPS Joint Program Office that the receiver be able to maintain track at a jerk level of 10 g/s for 0.6 s. Although this requirement is on the tracking loops, it most certainly has implications for the process noise that must be added to the acceleration covariance term in the GPS Kalman filter. There is no substitute for using the measured acceleration.

The other optional data transfer is that of the in-flight calibration and alignment corrections from the INS/GPS estimator to the INS. This helps keep the INS in closer agreement with the INS/GPS solution. Of course, the independence of the two solutions is lost.

In summary, we have improved the navigation accuracy of the combined GPS and the INS at the cost of independence in their solutions. We have maintained redundant systems.

2.2.3 Loosely Coupled - Rockwell's MAGR Approach

This approach might actually be characterized somewhere between loosely and tightly coupled. Figure 2.4 shows the GPS and INS functions and interfaces between them. The MAGR (Military Airborne GPS Receiver) has an INS mode and a PVA (Position, Velocity, and Acceleration) mode. The latter is a stand-alone mode independent of inertial measurements.

In the INS mode, inertial measurements are used to aid the code tracking loop when the carrier loop is out of lock and unable to provide aiding. The GPS uses the inertial measurements to extrapolate the position and velocity between GPS measurements rather than estimating acceleration in a polynomial filter. The GPS estimates attitude corrections for the IMU. The MAGR (in the INS mode) thus has some of the features of a tightly coupled system. Table 2.3 lists the filter state elements for the PVA and INS mode of operation.

Table 2.3. Filter states for the MAGR.

<table>
<thead>
<tr>
<th>PVA Mode</th>
<th>INS Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position</td>
<td>Position</td>
</tr>
<tr>
<td>Velocity</td>
<td>Velocity</td>
</tr>
<tr>
<td>Acceleration</td>
<td>Attitude</td>
</tr>
<tr>
<td>Clock bias</td>
<td>corrections</td>
</tr>
<tr>
<td>Clock drift</td>
<td>Clock bias</td>
</tr>
<tr>
<td>Barometer bias</td>
<td>Clock drift</td>
</tr>
<tr>
<td></td>
<td>Barometer bias</td>
</tr>
</tbody>
</table>
2.3 Tightly Coupled
Finally, the two changes that define a tightly coupled system are introduced. The GPS range and delta range measurements are incorporated directly into the navigation estimate, and the position and velocity from the inertial system are used by the GPS receiver to reduce the tracking loop bandwidths even in the presence of high dynamics.\(^4\) First, a straightforward system that provides a single combined INS/GPS solution will be presented. Then a system that also maintains independent and redundant GPS and INS solutions will be presented.

2.3.1 Tightly Coupled - Combined INS/GPS Only
Figure 2.5 shows the architecture for a tightly coupled INS/GPS navigation system that offers a single navigation solution. The INS and GPS modules have been truncated. The inertial “system” now simply provides raw measurements. The GPS receiver does not have its own Kalman filter, but it does still have independent tracking loops that provide the values for pseudo-range and range rate. Although it has not been shown in any of the figures, it is of course understood that the pseudo-range and range rate to at least four satellites are required for a position and velocity determination. The GPS functions shown in the upper diagram of Figure 2.5 are duplicated for each satellite by having multiple “channels” in a receiver - only one of which is shown in the diagram.

Figure 2.5. A tightly coupled INS/GPS navigation system offering only one combined solution.

The tracking loops in the receiver are aided by data from the INS/GPS state estimator. These data are required at a high rate, thus the propagation from one measurement epoch to another is broken into many subintervals for the purpose of tracking loop aiding. The goal is to make these tracking loops “think” the receiver is sitting still. The quantities being estimated by the Kalman filter are position and velocity, whereas the data required by the tracking loops are code phase (range) and Doppler frequency shift (range rate). The estimated position and velocity and the satellite ephemerides are used to calculate the code phase and frequency shift. The diagrams in this paper will show the transfer of r, v, and delta range and range rate, implying that these calculations are done in the receiver. They could as well be done in the “State Estimator” box. The bandwidth of the tracking loops must only accommodate the errors in the measured

\(^4\) The definitions of tightly coupled are not universally agreed upon. The first round of “EGI” receivers were considered to be tightly coupled by some but they did not have inertial aiding of the carrier tracking loops.
acceleration rather than the whole acceleration. These errors are many orders of magnitude less than the acceleration itself, depending on the quality of the inertial system and its calibration.

The tightly coupled navigation systems are more accurate. This can be seen in Reference 5, where tightly and loosely coupled systems are compared. We still have the gains or attributes of the loosely coupled systems except for the loss of redundancy. The bandwidth of the tracking loops can be reduced, thus increasing jamming resistance. The integration filter can make optimal use of any and all satellites that are being tracked, even if there are less than four of them. It should be said that GPS-only solutions can be maintained with either three or two satellites if one or two or both of the following assumptions are made: 1) the clock bias is constant and 2) the altitude is constant or is known by some other means (e.g., a baroaltimeter).

Only the redundancy offered by three complete systems is lost for this architecture. A summary of the benefits accrued by coupling will be given at the end of Section 2.3.2.

2.3.2 Tightly Coupled - Redundant Solutions

Figure 2.6 illustrates a tightly coupled architecture that also offers redundant navigation solutions from both the GPS and INS. This figure most closely resembles the Figure 2.3 for the loosely coupled architecture. The changes with reference to that earlier figure are inertial aiding of the tracking loops from the INS/GPS solution and the use of pseudo-range and range rate measurements rather than position and velocity in the integration filter.

Figure 2.6. Tightly coupled architecture with redundant GPS and INS-only solutions.

This more elaborate system requires more software. This is the price of the redundancy unless the software is already present in existing INSs and GPS. This can indeed be the case and was the case in the U.S. GPS Joint Program Office's Embedded GPS Inertial (EGI) program. The concept of the EGI program was to obtain a navigation system with GPS and inertial attributes at minimum cost. Specifications for such a (nondevelopmental) system were published. Several vendors have produced such embedded systems, among them are LN-100G [Ref. 6] and the H-764G [Ref. 7] combinations of GPS with ring laser gyroscopes. The U.S. Advanced Research Projects Administration also sponsored a tightly coupled and embedded combination, the GPS Guidance Package, using fiber-optic gyroscopes.

Embedding the receivers allows the data transfer rates required for tight coupling. EGI specifications state that separate and independent inertial-only and GPS-only solutions are to be maintained. Although they do not specify the two characteristics we have used to define tight coupling, they do state that INS aiding of the
tracking loops is allowed [Ref. 8]. This potentially makes the GPS solution dependent on the INS. Mathematical independence is maintained if the tracking loops have adequate signal strength to work with and can maintain lock such that the error in range rate (for example) is independent of the aiding value. If the error in the tracking loops is independent of the aiding, the GPS and INS/GPS solutions will be independent. Logic in the receivers attempts to recognize when lock is lost and not incorporate the resulting “bad” measurements into the GPS solution. This precaution also (arguably) keeps the GPS solution mathematically independent of the other solutions.

The tightly coupled receiver offers elevated jamming resistance. It offers the ability to continue operation when GPS is intermittent due to wing shadowing, foliage, or other natural or man-made obstructions. Table 2.4 summarizes the benefits that have been gained by coupling GPS with INS. The benefits are cumulative. That is, the benefits for each level also include those for the previous level. (The exception is loss of redundancy and independence for the simpler of the tight coupling architectures.)

Table 2.4. Cumulative Benefits of Increasingly Tight Coupling.

<table>
<thead>
<tr>
<th>Coupling Level</th>
<th>Benefit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Uncoupled/reset INS to GPS (Sum of system attributes)</td>
<td>Position, velocity, acceleration, attitude, and attitude rate information</td>
</tr>
<tr>
<td></td>
<td>Redundant systems</td>
</tr>
<tr>
<td></td>
<td>- A drift-free GPS</td>
</tr>
<tr>
<td></td>
<td>- A high-bandwidth INS</td>
</tr>
<tr>
<td>Loosely coupled</td>
<td>More rapid GPS acquisition</td>
</tr>
<tr>
<td></td>
<td>In-flight calibration and alignment</td>
</tr>
<tr>
<td></td>
<td>Better inertial instrument calibration and alignment</td>
</tr>
<tr>
<td></td>
<td>- Better attitude estimates</td>
</tr>
<tr>
<td></td>
<td>- Longer operation after jamming</td>
</tr>
<tr>
<td>Tightly coupled</td>
<td>Better navigation performance</td>
</tr>
<tr>
<td></td>
<td>Better instrument calibration</td>
</tr>
<tr>
<td></td>
<td>Reliable tracking under high dynamics</td>
</tr>
<tr>
<td></td>
<td>Reduced tracking loop bandwidth (jamming resistance)</td>
</tr>
<tr>
<td></td>
<td>Optimum use of however many SVs available</td>
</tr>
</tbody>
</table>

2.4 Deeply Integrated

Figure 2.7 shows the architecture of a deeply integrated INS/GPS navigation system. This figure compares most closely with the first tightly coupled architecture shown in Figure 2.4. In the deeply integrated concept, independent tracking loops for the code and carrier have been eliminated.

In the deeply integrated approach, the problem is formulated directly as an estimation problem in which the optimum (minimum-variance) solution is sought for each component of the multidimensional navigation state vector. By formulating the problem in this manner, the navigation algorithms are derived directly from the assumed dynamical models, measurement models, and noise models. The solutions that are obtained are not based on the usual notions of tracking loops and operational modes (e.g., State 3, State 5, etc.). Rather, the solution employs a nonlinear filter that operates efficiently at all jammer/signal (J/S) levels and is a significant departure from traditional extended Kalman filter designs. The navigator includes adaptive algorithms for estimating pos-correlation signal and noise power using the full correlator bank.

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5 The material in this section is from References 9 and 10. “Deep integration” is trademarked by Draper Laboratory.
Filter gains continuously adapt to changes in the J/S environment, and the error covariance propagation is driven directly by measurements to enhance robustness under high jamming conditions (see Figure 2.8).

Figure 2.7. Deeply integrated INS/GPS systems feature a single estimator for both detection and navigation.

Figure 2.8. INS/GPS deep integration.

In this system, individual satellite phase detectors and tracking loop filters are eliminated. Measurements from all available satellites are processed sequentially and independently, and correlation among the line-of-sight distances to all satellites in view is fully accounted for. This minimizes problems associated with unmodeled satellite signal or ephemeris variations and allows for full Receiver Autonomous Integrity Monitoring (RAIM) capability.

The design offers several significant benefits at high J/S levels. The effects of measurement nonlinearities, which are significant at high J/S levels, are accounted for in an efficient manner. The estimator produces near-optimal state vector estimates as well as estimates of the state error covariance matrix. The estimator operates in real time using recursive algorithms for both state vector and error covariance matrix estimation.
The J/S levels are estimated adaptively in real time to facilitate seamless transitions between course tracking and tight tracking without the use of artificial moding.

Extended-range correlation may be included optionally to increase the code tracking loss-of-lock threshold under high jamming and high dynamic scenarios. If excessively high jamming levels are encountered (e.g., beyond 70 to 75 dB J/S at the receiver input for P(Y) code tracking), the GPS measurements may become so noisy that optimal weights given to the GPS measurements become negligible. In this situation, navigation error behavior is essentially governed by current velocity errors and the characteristics of any additional navigation sensors that are employed. Code tracking is maintained as long as the line-of-sight delay error remains within the maximum allowed by the correlator bank. If there is a subsequent reduction in J/S so that the optimal weights become significant, optimum code tracking performance is maintained without the need for reacquisition. Detector shapes for each correlator depend on the correlator lag and rms line-of-sight delay error.

For navigators using GPS only, navigation errors will be reduced significantly by using algorithms that approximate the minimum-variance solutions at high J/S. For navigators employing other sensors, a fully integrated system will allow simpler, smaller, cheaper hardware to be employed. Superior sensor calibration capability will reduce sensor performance requirements, allowing lower-cost sensors to be used.

Figure 2.9 shows the information flow between the principal elements of the navigation system. The data from each satellite in view are processed sequentially; the figure illustrates processing for a single satellite. The GPS receiver front end performs filtering, carrier wipeoff and sampling to produce I/Q data. These data are processed by each correlator to produce the 50-Hz samples $I_{50}(j,k)$ and $Q_{50}(j,k)$ for the $k^{th}$ correlator at the $j^{th}$ time point. Square law detection and summation is then used to obtain $Z_n(n)$; currently, summation is over five samples so that $Z_n(n)$ is 10-Hz data. The processor uses inputs $Z_n(n)$ to calculate the navigation state estimate $\hat{x}(n)$. The state estimate is propagated to measurement update time using an assumed dynamical model. As shown in the figure (dashed lines), two types of sensors may be optionally added to the GPS-based navigator. Inertial sensor data may be incorporated during propagation to reduce the error bandwidth during periods of high dynamics and retard error growth if code lock is lost. If inertial sensors are used, the processor accepts raw sensor data (e.g., body frame specific force and angular rates for a strapdown configuration) and time-correlated sensor error states may be included in the navigation state vector in order to perform in-flight calibration of significant error sources. At measurement update time, the state is updated using the measurements $\{Z_k(n); k = -m, ..., m\}$ from $2m + 1$ correlators, satellite ephemeris data, and (optionally) measurements from other sensors (e.g., radars, altimeters, etc.). The estimated time delay $\hat{\tau}(n)$, which is a function of the state estimate and satellite ephemeris, is fed to the code NCO, which controls correlator code phase in order to maintain the mean code tracking error close to zero.
The navigator includes adaptive algorithms for estimating postcorrelation signal power (S) and noise power (N). Noise statistics are assumed to be the same for all correlators. Although the 50-Hz noises are uncorrelated over time, the noise in adjacent correlators is correlated.

3.0 Summary

This paper has described INS/GPS integration architectures including loosely coupled, tightly coupled, and deeply integrated configurations. The advantages and disadvantages of each level of integration were listed. Examples of current and futures systems were cited. In a companion paper, Reference 5, performance comparisons between the three major INS/GPS system architectures for various mission scenarios will be presented in order to understand the benefits of each. The loosely coupled and tightly coupled systems will be compared in several scenarios including aircraft flying against jammers and a helicopter flying a scout mission. The tightly coupled and deeply integrated architectures will be compared for several jamming scenarios including that of a precision guided munition.

References:


