Inertial Navigation for Guided Missile Systems

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An accurate inertial reference based on measurements of missile angular velocity and acceleration is needed for all of the major guidance and control functions of a guided missile. For example, intercept of a target would not be possible without a good inertial reference system to stabilize target line-of-sight measurements for the computation of missile guidance commands. This article provides an overview of inertial navigation for guided missiles. Missile navigation data (position, velocity, and attitude) are needed for missile guidance and control. Furthermore, this article describes in-flight alignment techniques that can be used to increase the accuracy of the missile navigation-system data by incorporating external non-inertial navigation-aiding data using a navigation Kalman filter. For guided missile systems, this aiding often is provided by an external radar track of the missile and/or Global Positioning System (GPS) receiver measurements. This article also discusses more recent advances in navigation for guided missiles. APL has been a major contributor to the development of many of the advanced guided missile navigation systems in use today.

INTRODUCTION

Inertial navigation has been a key element of missile system design since the 1950s. Traditionally, the focus has been on strategic- and precision-strike systems. In these applications, terminal-position accuracy is the primary objective of the navigation system. In guided missile systems in which a terminal seeker is used to sense and track an air or ballistic missile threat, a critical function of the inertial navigation system (INS) is to provide accurate seeker-attitude information and, therefore, allow accurate pointing of the seeker for acquisition of a target. In addition, the navigation system provides essential data for guidance and flight-control functions.

This article also discusses more recent advances in navigation for guided missiles. These advances have been motivated by several factors. The historical use of a semi-active RF seeker with a wide field of view placed less demand on the accuracy of the navigation system for pointing information. The use of wide-field-of-view seekers also was consistent with the fact that accurate navigation systems were high in cost, heavy in weight, large in volume, and, therefore, not suitable for tactical guided missiles. However, as lower-cost, smaller, and more reliable inertial measurement units (IMUs) have become readily available, missile systems have been able to employ higher-accuracy, smaller-field-of-view seekers such as infrared or high-frequency RF technology without the need to perform an angle search. The use of advanced seeker technology naturally leads to better overall performance against more stressing targets. A second consideration is that targeting information may be improved by the use of multiple sensors. As sensor-alignment errors and target-track errors are taken into account, it is desirable to minimize the alignment error between the missile seeker and the targeting reference. A third consideration is the missile guidance system configuration before seeker acquisition. Typically, a missile is guided by uplinks that are based on filtered radar measurements of both the missile and the target. In an alternative approach, called inertial midcourse guidance, the tracking radar still provides filtered targeting data and unfiltered missile-position measurement data, but the missile itself computes the guidance commands. This latter approach places greater reliance on the missile navigation and guidance systems in an attempt to improve overall system performance.

This article provides a general overview of inertial navigation and describes the basic navigation-system design approach. Also included is a discussion of the navigation functions of a guided missile system during the various phases of flight. Finally, we present a summary of the future of advances in inertial navigation for guided missile systems.

Unaided Inertial Navigation

Most commonly, an INS is used to determine the position, velocity, and orientation of a vehicle moving relative to the Earth's surface. The INS computations are based on gyroscope measurements of inertial angular velocity to determine the orientation of a triad of accelerometers. The accelerometer measurements, in turn, are integrated to estimate vehicle velocity and position.

There are two fundamental approaches to INS mechanization. Because of the dynamic ranges and error sensitivities of earlier gyro technologies and computer limitations, platform systems were the most common mechanization approach before the 1990s. In these systems, the inertial instruments are placed on a stabilized platform that is gimbaled with respect to the host vehicle, making the measurements insensitive to rotational motion. Although platform mechanization still is used today in many applications, such as aircraft, cruise missiles, and ships, it is not suitable for tactical missiles because of the cost, volume, and weight.

During the 1970s, gyroscopes with lower error sensitivity to angular rate were developed. Concurrent advances in computer technology led to interest in strapdown systems in which the inertial instruments are rigidly attached to the host vehicle. Here, the sensor measurements are mathematically transformed to a stabilized reference frame to remove the effects of vehicle motion. Although the computations associated with a strapdown INS are conceptually simple, the mechanization can be quite complex because of the multiplicity of rotating coordinate frames involved.[1–5] As shown in Fig. 1, the strapdown INS measures angular velocity and acceleration of the missile body relative to inertial coordinates, but these measurements are sensed in the rotating frame of the missile body denoted by the coordinates of the inertial measurement unit case, $(\mathbf{i}_b, \mathbf{j}_b, \mathbf{k}_b)$. Moreover, the desired navigation solution typically is formed relative to a second, rotating Earth-centered Earth-fixed (ECEF) coordinate frame, $(\mathbf{i}_e, \mathbf{j}_e, \mathbf{k}_e)$, having angular velocity $\bar{\omega}_{ie}$, relative to the inertial frame. If the position vector of the missile, $\bar{\mathbf{r}}$, over the Earth's surface is desired (latitude, longitude, and altitude), then a model for the ellipsoidal shape of the Earth's surface must be used. For the guided missile problem, one of the most critical quantities is the orientation of the $(\mathbf{i}_b, \mathbf{j}_b, \mathbf{k}_b)$ coordinate frame relative to some reference frame.

A common choice for the reference frame where strapdown computations are performed is a local-level navigation frame that is tangent to the Earth's surface and perpendicular to the local gravity vector acting on the missile. This local-level frame moves across the surface of the Earth with the translational motions of the

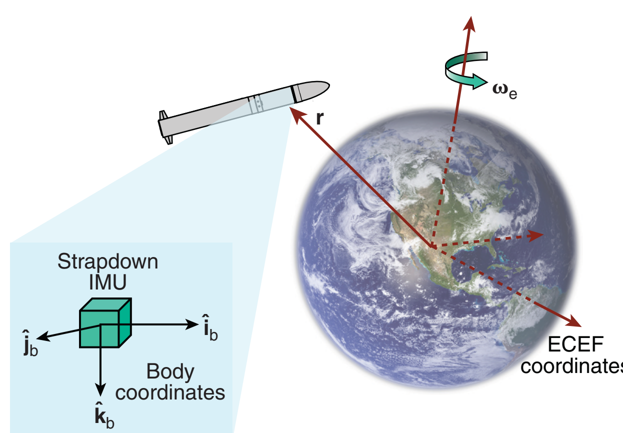

Figure 1. The inertial navigation problem.

missile. The coordinate axes of the local-level frame contained in the tangent plane are further defined so that the angular velocity of the local-level coordinate frame along the vertical direction is zero (Fig. 2). The resulting coordinate frame is commonly called the "wander azimuth frame," which is defined so as to avoid high-azimuth angular velocity near the poles. The wander angle is the angle of the $\mathbf{j}_n$ axis relative to the north direction.

To illustrate the nature of the navigation computations, we first show the velocity equation written in the navigation frame. Define $\bar{\mathrm{v}}$ as the missile velocity relative to the Earth-fixed frame. Then, the vector time derivative of $\bar{\mathrm{v}}$ ($D_n\bar{\mathrm{v}}$) as observed in the rotating navigation frame is given by

$$D_n\bar{\mathrm{v}} = \bar{\mathrm{f}} + \bar{\mathrm{g}} + (2\bar{\omega}_{ie} + \bar{\omega}_{en}) \times \bar{\mathrm{v}} \ , \qquad (1)$$

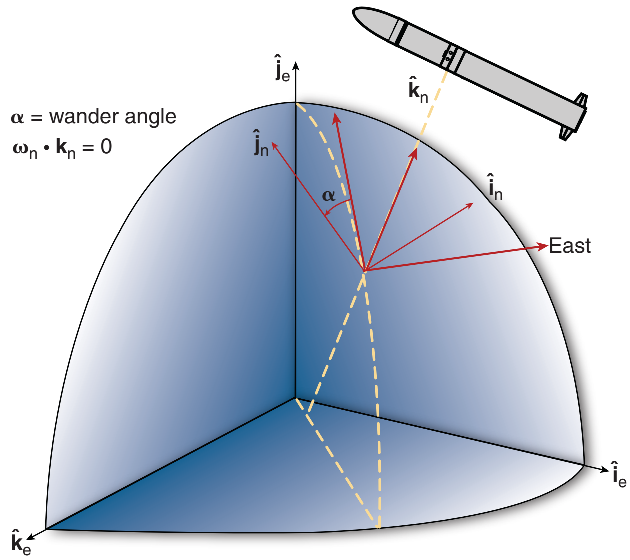

Figure 2. Navigation coordinates.

where $\bar{\mathrm{f}}$ is the specific force (non-gravitational acceleration) measured by the accelerometers, $\bar{\mathrm{g}}$ is the gravity vector observed in an Earth-rotating frame (includes centripetal effects), $\bar{\omega}_{ie}$ is the angular velocity of the Earth relative to an inertial frame (taken to be the non-rotating Earth frame), and $\bar{\omega}_{en}$, is the angular velocity of the navigation frame relative to the Earth-fixed frame. The gravity term, $\bar{\mathrm{g}}$, must be computed as a function of altitude and latitude, and it includes the centripetal acceleration caused by the Earth's rotation. Note that Eq. 1 is a vector equation that may be expressed in any coordinate frame. Most commonly, either the local-level navigation frame described above or the ECEF frame is chosen.

Figure 3 illustrates an example set of navigation computations. The gyro and accelerometer measurements are accumulated over a measurement interval and compensated by factory-measured errors, typically bias and scale factor versus temperature. Coning and sculling compensation are approximations to account for the vehicle's rotational motions during the measurement interval, and size compensation accounts for the fact that the accelerometers cannot be physically collocated, so a lever-arm term caused by case rotation must be removed. The coning and sculling compensations may be performed within the IMU internal software or in the navigation computer. The compensated body-angle increments, $\Delta\theta$, then are used to compute a body-to-navigation-frame transformation. This process typically is implemented via computation of a body-attitude quaternion and an associated orientation vector. The orientation vector is a direct function of the gyro incremental-angle measurements.[2–5] Although shown in Fig. 3 as a direction cosine matrix transformation, an equivalent quaternion transformation often is used to transform the compensated incremental velocity measurement vector, $\bar{\mathrm{v}}_c$, from the body frame to the navigation frame. The resulting incremental velocity terms are summed and compensated per Eq. 1 to produce the computed velocity. The linear velocities are converted to angular velocity, $\bar{\omega}_{en}$, and then used to update the direction cosine matrix that describes the orientation of the navigation frame relative to the Earth frame. Latitude and longitude may be extracted from the direction cosine matrix. Altitude is computed separately by using velocity in the vertical direction ($\mathbf{k}_n$).

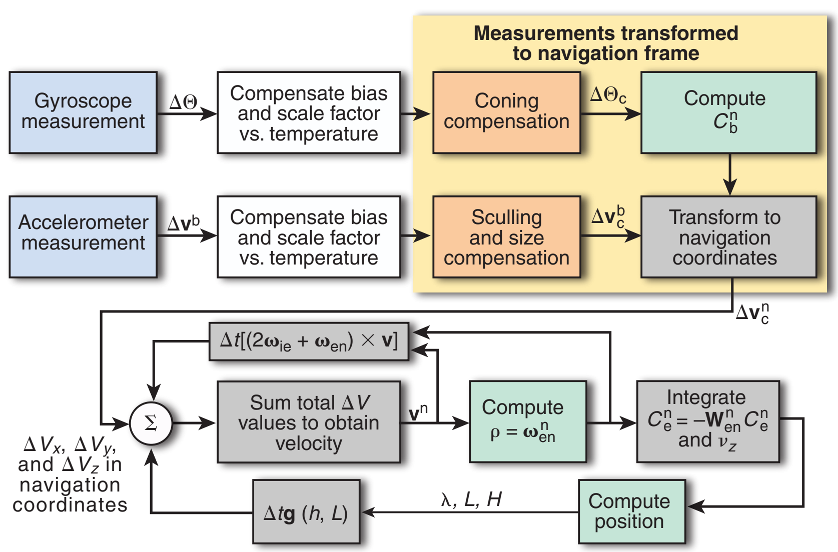

Figure 3. Strapdown mechanization.

Table 1. Navigation-system classes.

	Commercial	Tactical	Navigation
Gyro bias (°/h)	10–100	1–10	0.002–0.01
Accelerometer bias (mg)	1	1	0.05
Cost range ($1000)	<1	4–15	50–90

The gyroscope and accelerometer technologies used in navigation systems vary considerably in construction and accuracy. Gyroscope technologies fall into the categories of mechanical gyros that depend on the angular momentum of a spinning mass, vibratory gyros that depend on Coriolis acceleration effects, or optical gyros. The most common gyros used today for navigation in tactical missiles are optical and employ either ring-laser or optical-fiber technologies. Accelerometers are constructed by using either pendulous or resonant-beam technologies. Table 1 illustrates typical inertial instrument accuracies for different classes of navigation systems. The commercial or tactical grade typically is found in guided missiles. The key parameter is gyro accuracy, which also drives the cost of the system. Navigation-grade systems typically are found in aircraft and cruise missiles.

Table 1 illustrates the different classes of navigation systems in terms of instrument bias, but it is important to note that the actual accuracy of a navigation system is a function of many factors. In addition to bias, other modeled error sources typically will include scale-factor error, acceleration sensitivity, angular-rate sensitivity, random walk, random noise, and instrument misalignment. The propagation of error to computed velocity, position, and attitude will be a function of the inertial-instrument error, initialization error, and host-vehicle dynamics. The next section discusses how these errors may be estimated by application of a Kalman filter with independent measurements of position and/or velocity.

Aided Inertial Navigation

An aided INS employs external measurements of missile position and/or velocity to estimate and correct the navigation-system errors. This estimation and correction is accomplished by use of a Kalman filter[6,7] and careful modeling of the dynamics of navigation-error propagation and the errors in the external aiding measurements. A discrete-time Kalman filter model is described briefly as follows. The error dynamics of an INS can be modeled by the following linear discrete-time system. A discrete-time dynamic system is represented by

$$\bar{\mathbf{x}}_{k+1} = \varphi_k^{k+1}\bar{\mathbf{x}}_k + \bar{\mathbf{w}}_k \qquad \text{(dynamics equation)}$$

and

$$\bar{\mathrm{z}}_k = \mathbf{H}_k\bar{\mathbf{x}}_k + \bar{\mathbf{v}}_k \qquad \text{(measurement equation),}$$

where $\bar{\mathbf{x}}_k$ is the error-state vector of the system at time t_k, $\boldsymbol{\varphi}_k^{k+1}$ is the state-transition matrix that maps the error states from time t_k to t_{k+1}, $\bar{\mathbf{w}}_k$ is the random white-noise vector that represents input process noise, $\mathbf{H}_k$ is the measurement matrix that relates the measurements to the states of the system, and $\bar{\mathbf{v}}_k$ is the random white-noise vector that represents the output measurement noise.

A standard assumption is that the initial error state of the system, the process noise, and the measurement noise are mutually uncorrelated. The covariance of the initial state error, the process noise, and the measurement noise are normally expressed as $\mathrm{cov}(\bar{\mathbf{x}}_0) = \mathbf{P}_0$, $\mathrm{cov}(\bar{\mathbf{w}}_k) = \mathbf{Q}_k$, and $\mathrm{cov}(\bar{\mathbf{v}}_k) = \mathbf{R}_k$, respectively. The corresponding optimal Kalman filter that can be used to estimate the states of the system is given by the following recursive algorithm:

Update Step

$$\mathbf{K}_k = \mathbf{P}_{k/k-1}\mathbf{H}_k^T(\mathbf{H}_k\mathbf{P}_{k/k-1}\mathbf{H}_k^T + \mathbf{R}_k)^{-1}$$
(Kalman gain computation)

$$\hat{\bar{\mathbf{x}}}_{k/k} = \hat{\bar{\mathbf{x}}}_{k/k-1} + \mathbf{K}_k(\bar{\mathrm{z}}_K - \mathbf{H}_k\hat{\bar{\mathbf{x}}}_{k/k-1})$$
(state update)

$$\mathbf{P}_{k/k} = (\mathbf{I} - \mathbf{K}_k\mathbf{H}_k)\mathbf{P}_{k/k-1}$$
(covariance update)

Prediction Step

$$\hat{\bar{\mathbf{x}}}_{k+1/k} = \boldsymbol{\varphi}_k^{k+1}\hat{\bar{\mathbf{x}}}_{k/k}$$
(state extrapolation)

$$\mathbf{P}_{k+1/k} = \boldsymbol{\varphi}_k^{k+1}\mathbf{P}_{k/k}(\boldsymbol{\varphi}_k^{k+1})^T + \mathbf{Q}_k$$
(covariance extrapolation)

Notice that in the previous recursive algorithm, the computation of the state-error covariance, $\mathbf{P}_{k/k}$, is not explicitly a function of the estimated state of the system. Hence, the estimated state-error covariance of the system can be computed by a single recursive run for any given reference trajectory. This capability is a very powerful tool, forming the basis of covariance analysis, which is discussed in *Covariance Simulation* below.

The modeling of navigation-error propagation can be accomplished by several methods, and each method will produce a different form of the error equations. For example, the error equations might be based on navigation quantities computed via Eq. 1 in either a local-level navigation frame or an ECEF frame. Furthermore, an important consideration is the different coordinate frames that must be defined to account for differences between computed and true quantities. For example, as illustrated in Fig. 4, a "computer frame" may be defined as the local-level frame located at the computed position. To describe the navigation position and velocity errors, we must recognize that computed velocity

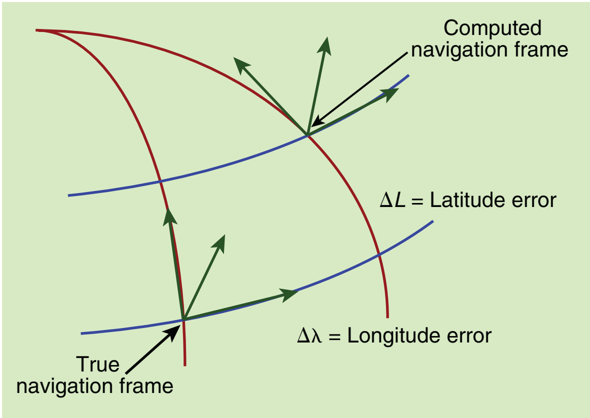

Figure 4. True and computed local-level frames.

fundamentally exists in the computer frame, whereas true velocity fundamentally exists in the true frame. Thus, one way to properly model velocity error is to convert true velocity to the computer frame and express error quantities in that frame. This conversion leads to the so-called computer-frame error-state equations shown below in Eq. 2. The states are position errors, velocity errors, and tilt errors. The tilt-error vector, $\bar{\boldsymbol{\psi}}$, describes the tilt in the computer frame caused by gyro errors.

$$\delta\dot{\bar{\mathbf{r}}}^n = -\bar{\omega}^n_{en} \times \delta\bar{\mathbf{r}}^n + \delta\bar{\mathbf{v}}^c \tag{2a}$$

$$\delta\dot{\bar{\mathbf{v}}}^c = \hat{\mathbf{C}}^n_b \delta\bar{\mathbf{f}}^b + \delta\bar{\mathbf{g}}^c - \bar{\boldsymbol{\psi}} \times \bar{\mathbf{f}}^c - (2\bar{\omega}^c_e + \bar{\omega}^c_{ec}) \times \delta\bar{\mathbf{v}}^c \tag{2b}$$

$$\dot{\bar{\boldsymbol{\psi}}} = -\bar{\omega}^c_{ic} \times \bar{\boldsymbol{\psi}} - \mathbf{C}^c_b \delta\bar{\omega}^b_{ib}, \tag{2c}$$

where $\delta\bar{\mathbf{r}}$ is position error, $\delta\bar{\mathbf{v}}$ is velocity error, $\delta\bar{\mathbf{f}}$ is accelerometer errors, and $\delta\bar{\omega}$ is gyro errors. Superscripts indicate a vector being coordinatized in the body frame or IMU case (b), the computer frame (c), or the navigation frame (n). The gravity error, $\delta\bar{\mathbf{g}}$, is caused by position error. It is given by

$$\delta\bar{\mathbf{g}}(R) \approx \nabla\bar{\mathbf{g}}(R)\cdot\delta\bar{\mathbf{r}} \approx \begin{bmatrix} -g/R & 0 & 0 \\ 0 & -g/R & 0 \\ 0 & 0 & 2g/(R+h) \end{bmatrix} \cdot \delta\bar{\mathbf{r}}.$$

Depending on the application, several of the terms in Eq. 2 may be neglected to provide a simpler form.

Eq. 2 forms the core of the Kalman filter for estimation of navigation error where the state vector consists of $\delta\bar{\mathbf{r}}$, $\delta\bar{\mathbf{v}}$, $\delta\bar{\mathbf{f}}$, and $\delta\bar{\omega}$, and the state transition matrix, φ, is based on the dynamics given by Eq. 2. In addition to the basic kinematic relationships described by Eq. 2, another important factor in the Kalman filter formulation is the modeling of IMU instrument error states as well as states to account for fixed or slowly varying errors in the external aiding measurements. The IMU error states will enter through the gyro and accelerometer error terms in Eq. 2. Typically, biases are modeled as first-order Markov processes, whereas scale factor and misalignment are modeled as biases. The external aiding error states will enter through the Kalman filter measurement equation. The measurement equation relates the measurement error to the Kalman filter error states. External aiding errors such as misalignments, time-tag bias, and radar refraction can be modeled as biases. The external measurement errors also may be a function of the current state of the system.

For guided missile systems, inertial aiding often is provided by an external radar track of the missile and/or Global Positioning System (GPS) receiver measurements. A sample system configuration is shown in Fig. 5. The IMU gyro and accelerometer measurements, typically on the order of 100 Hz, are sent to the navigation computer. These measurements then are corrected by Kalman estimates of instrument errors and used to update the attitude, velocity, and position computations. These navigated quantities then are compared to the aiding signals to form the Kalman filter residual and the subsequent filter update.

GPS has become a common form of inertial aiding because it provides highly accurate navigation data at a low cost. There are two basic forms of GPS aiding. One approach, called loose coupling, uses the position and velocity solutions of the GPS receiver. A second approach, called tight coupling, uses the more basic GPS pseudorange and delta pseudorange measurements. The advantages of the latter approach is that navigation updates are still possible as the number of tracked satellites falls below the threshold of four that is required to develop a GPS position solution for the loosely coupled approach. Moreover, the tightly coupled approach avoids noise-correlation issues and potential stability problems caused by having two Kalman filters in cascade. The tightly coupled configuration is illustrated in Fig. 6. To compute a filter residual, the estimated vehicle position and velocity are used with the downlinked satellite ephemeris data to compute pseudorange and delta pseudorange errors. The Kalman filter measurement matrix then establishes the relationships between the measurement residuals and the filter states.

The use of external radar tracking data to aid the guided missile navigation system enables the alignment of the missile seeker to the radar reference frame. The alignment of these frames will tend to reduce the missile-to-target terminal handover error baskets.

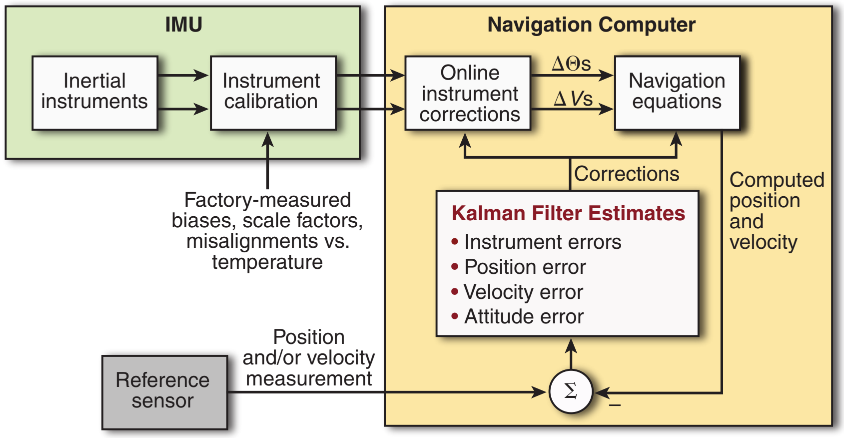

Figure 5. Sample aided INS configuration.

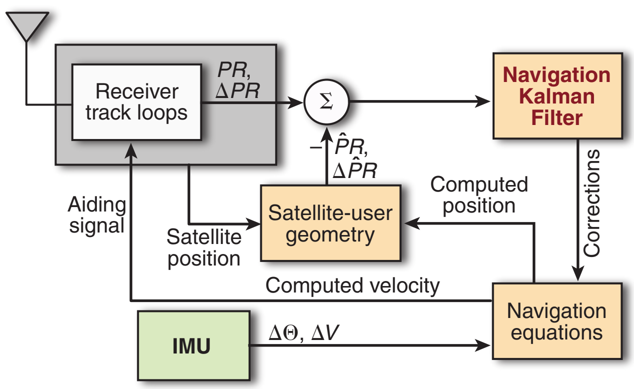

Figure 6. Tightly coupled INS/GPS configuration.

Transfer Alignment

As described previously, inertial-system alignment accuracy often is critical in meeting system-performance goals. Alignment accuracy may be achieved by using accurate initialization or postinitialization improvement. Transfer alignment is a term applied to a broad class of techniques that can be used in both approaches. Transfer alignment represents a form of aided INS. In its simplest form, transfer alignment is based on the idea that measurements made by the IMU and externally obtained knowledge of the corresponding motions can be used to determine the alignment of the inertial sensors and computed seeker frame with respect to the external reference frame. From this idea, a wide variety of alignment techniques can be constructed. Many are based on the inertial aiding approach described in *Aided Inertial Navigation*.

The external measurement may be made by using either another inertial system or a non-inertial system. As long as the same kinematic quantity can be derived from the non-inertial data, transfer alignment can be accomplished. In-flight alignment (IFA) of a missile inertial system using GPS is an excellent example of a non-inertially derived measurement. GPS-measured position or velocity can be compared to position and velocity estimates produced from IMU measurements. Comparing these data allows a determination of the inertial-system attitude with respect to the GPS coordinate system.

Because the Kalman filter dynamics matrix is a function of missile acceleration and rotation rate, the missile trajectory will impact the ability of the filter to estimate inertial errors. Typically, it is best if the motion of the missile contains acceleration of sufficient duration in two spatially diverse directions. The design of the appropriate shaping strategy often is a significant task in and of itself.[8]

The simplest form of initialization is use of external knowledge of the missile attitude to directly initialize the attitude computation. For guided missiles, initialization typically is done by using the attitude output from an INS on the launch platform and knowledge of the relative mechanical orientation between the launch platform and the missile IMU. When prelaunch time is available, such as in the case of cruise missiles, an inertial transfer alignment between the INS on the launch platform and the missile INS can be performed to estimate the relative missile INS errors. However, when prelaunch transfer alignment is not possible because of a short prelaunch timeline, IFA techniques can be employed.

NAVIGATION-SYSTEM DESIGN AND TESTING

In this section, we describe the process used to design a guided missile navigation system. Typically, there are four major steps involved in the development of a navigation system: the initial requirements specification and formulation of error budgets; the initial design phase, which often involves the aid of a covariance simulation; the implementation of the actual navigation equations and Kalman filter algorithms; and the final stage of design verification and validation through Monte Carlo simulation and actual hardware testing.

Navigation Requirements and Error Budgets

The initial phase of the navigation-system design involves the development of the system's fundamental requirements. During this initial stage of the design, the navigation-system design engineer must become knowledgeable about the overall weapon system and its intended modes of operation. The designer should be aware of what types of external information will be available and what types of information will be required from the navigator to support the other guided missile subsystem functions. In short, to guarantee overall mission success, the design engineer should have a good understanding of how the navigation system fits within the overall weapon system to ensure development of a design that will satisfy overall weapon-system performance.

Usually during the initial stage of missile design, the system engineers will have a good understanding of the top-level objectives of the overall guided missile system. Furthermore, the system designer will need to identify the specific hardware components that will be included in the missile along with their basic, key performance parameters. For missile navigation, the field of regard of the seeker is a very important driver of navigation-system performance requirements. IR missile seekers, in particular, typically have smaller fields of regard than do RF seekers. In addition, over the past decade there has been a continued desire to extend the range of missile operation to engage longer-range, higher-speed, and lower radar cross-section threats. These objectives come with an added price because they each tend to increased overall weapon-system errors. The increased system errors and smaller seeker fields of regard have required significant improvements in overall missile navigation performance. The missile navigation system has become a critical component within the overall weapon system, and the quality of its design is very important to enable the weapon system to minimize overall system error and satisfy top-level system-error budgets.

Covariance Simulation

Once the top-level navigation-system requirements have been determined and a general architecture of a missile navigation system has been chosen, the actual navigation-system design begins. A particularly useful tool during the initial stages of the design is a covariance simulation. In covariance simulation, it is the uncertainty of the navigation state that is important and not the state vector itself. The basis of a covariance simulation is the recursive Kalman filter error-covariance equations that were discussed previously in *Aided Inertial Navigation*. Another advantage of covariance analysis is that a single run of a covariance simulation provides a statistical assessment of system performance. Hence, covariance analysis avoids the need to perform a very large number of individual simulation runs to carry out a statistical assessment of system performance.

In the design of a navigation system, the designer must strike a balance between a conflicting set of hardware and software objectives. Better performance may be achieved by using better sensors or by including more states within the Kalman filter model. Covariance analysis provides a systematic approach to quickly evaluate alternate system implementations. Another benefit of covariance analysis is that it allows the designer to easily determine the dominant error sources within the system and to determine whether it is feasible to reduce the error through either better hardware or more detailed modeling. In practice, covariance analysis can be a very effective tool in predicting performance, refining error budgets for components and external sensors, and tuning the Kalman filter in aided-navigation systems. More detailed information regarding covariance simulation and suboptimal analysis can be found in Refs. 6 and 7.

Monte Carlo Simulation

Once the basic navigation-system configuration has been determined, the detailed design is initiated. The detailed design involves the selection of computational methods, data rates, algorithms to compensate for data latencies, and tuning of Kalman filter parameters. The detailed navigation design is developed and then integrated within either a high-fidelity navigation simulation or a high-fidelity missile six degree-of-freedom simulation (6-DOF). The purpose of these simulations is to assess navigation-system performance within a high-fidelity computer environment that contains detailed models of all significant missile system components that are relevant to the navigation system. These simulations will include a high-fidelity representation of the complex linear and rotational motions of the missile dynamics that occur in the flight environment. They also will include high-fidelity models of the navigation sensors, which may include IMU model and GPS receiver as well as detailed implementations of the navigation and Kalman filter equations. In addition, they also will include high-fidelity models of the relevant weapon system components, such as the radar system, the launch platform navigation system, as well as the missile initialization process and weapon-system-to-missile uplink communications. These simulations also incorporate Monte Carlo test capabilities that allow comparison of the missile navigation data against the corresponding simulation truth data. Unlike covariance analysis, Monte Carlo testing typically requires a very large number of individual simulation runs be performed to generate a statistical assessment of system performance.

These simulations are essential because they enable navigation-system testing in a high-fidelity computer environment that includes all relevant weapon systems components needed to assess missile navigation-system performance. These simulations are used to assess the detailed navigation algorithm development, evaluation of linear and non-linear effects in IMUs and any external sensors, and verification of performance under the complex linear and rotational motions produced by guided missiles. APL engineers have developed highly detailed 6-DOF simulations and navigation simulations for many different guided missile systems that have been developed over the past 35 years. These simulations currently are maintained in the Advanced Missile Simulation and Evaluation Laboratory (AMSEL) and have played a critical role in the design, testing, and evaluation of many advanced guided missile navigation systems currently in use today.

It should be noted that the high-fidelity computer models are based on the engineer's understanding of the system and the components within it. There is no guarantee that these high-fidelity computer models accurately represent all of the specific features associated with the actual hardware, which might not be well understood in advance. These simulations are only as good as the accuracy of the mathematical models that are contained in them. Hence, it is important to validate these simulation models against actual flight data.

Navigation-System Testing

The final step of the navigation design is to verify that the actual flight hardware and software components operate as expected and the top-level design objectives are satisfied. This final stage of testing typically will involve both open-loop and closed-loop hardware-in-the-loop (HIL) tests that are can be performed in the laboratory and in the field. The goals of this testing are to verify that the real hardware and software components perform as expected in the real-world setting and that the top-level performance goal of the system actually is achieved. There is no guarantee that the actual hardware and software components operate exactly the same as the high-fidelity models that have been developed for the Monte Carlo simulation and testing. More often than not, this final stage of hardware/software testing will uncover an important feature of the hardware or an error in the flight software that had previously been unknown or undetected. This process may result in better characterization of the system errors, or the error may be fixed and incorporated into the high-fidelity models. A new version of the hardware and software then is developed and re-tested. Needless to say, the final hardware/software testing is a process that may involve much iteration before a final flight-ready design is developed. APL engineers perform a significant amount of laboratory testing of flight hardware and software in the Guidance Section Evaluation Laboratory (GSEL) and Navigation and Guidance Integration and Test Facility (NAVSIL). These APL test facilities provide a critical testing capability for many of the guided missile systems in use today.

GUIDED MISSILE NAVIGATION FUNCTIONS

The guided missile navigation system is responsible for maintaining the current state of the system relative to a designated frame of reference. This missile state data are required by virtually all other missile subsystems. Figure 7 illustrates this data flow. High-rate inertial-acceleration and angular-velocity data provided by the IMU are needed by the various missile-control systems, such as the missile autopilot and the terminal sensor pointing-control system. The lower-rate missile navigation data (including position, velocity, and attitude data) and derivatives of these data (such as Mach, angle-of-attack, and altitude) are needed by the autopilot, guidance, and seeker search functions. Missile-position, velocity, and attitude error-covariance information is required by the terminal sensor search algorithms to construct overall missile-to-target position and velocity error baskets that are needed to characterize an uncertainty region of the target. In addition, the missile navigation system often is responsible for performing the missile timing and rocket motor staging functions. Hence, missile navigation is a critical component of the overall guided missile system.

The following sections describe the operation of the missile navigation system during the various phases of the guided missile flight, which are illustrated in Fig. 8.

Initialization

The guided missile typically is a fast-reaction-time weapon that is powered and launched within seconds after the weapon system determines a threat must be engaged. The missile navigation system, which can

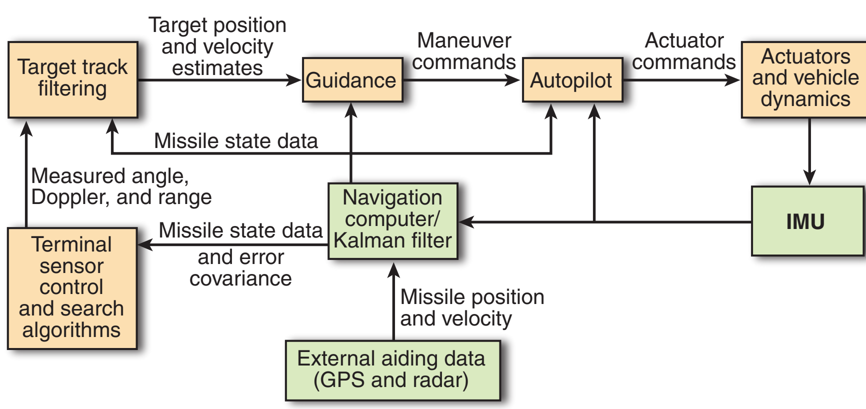

Figure 7. Guidance, navigation, and control loop.

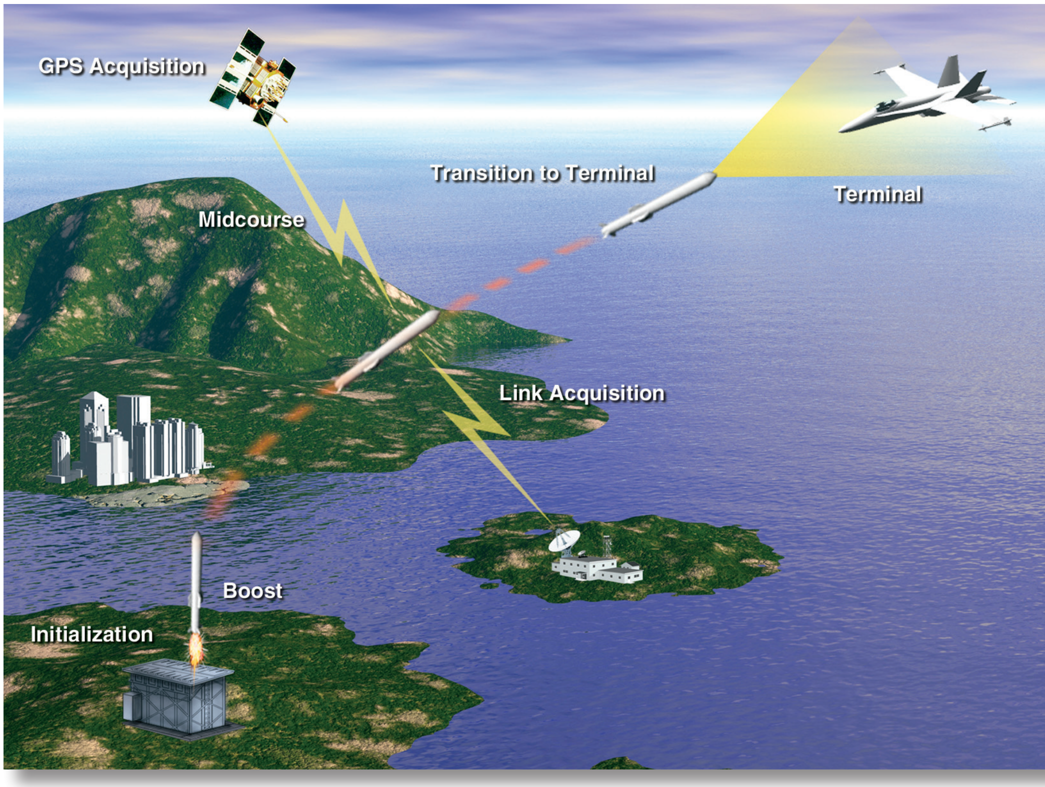

Figure 8. Phases of missile flight.

include a GPS receiver and an IFA Kalman filter, typically will receive initialization data from an electronic interface between the launch platform and the missile. The initialization data may include targeting data to characterize the threat and initial missile-state data needed to initialize the missile navigation system. Initialization data often will include missile position, velocity, and attitude and may include initial error-covariance data for the IFA Kalman filter.

Boost Phase

Boost phase begins when the missile rocket motor is ignited and the missile emerges from its canister. During this phase, the missile quickly increases speed. At some prescribed time or distance away from the launch platform, the missile will perform an initial pitch-over maneuver that begins to steer the missile away from the launch platform toward the direction of the predicted intercept point. The missile INS computes missile position, velocity, and attitude by integrating the IMU high-rate missile acceleration and rotational rate measurements. This period is characterized by free inertial operation of the navigation equations. During boost, the high-rate IMU acceleration and rotational rate data also are required by the missile autopilot.

During this initial phase of flight, the weapon system will attempt to establish a communications link between the missile and the launch platform. Also, if the navigation system is equipped with a GPS receiver, it will complete its initialization and begin to acquire the GPS signals. This phase is important because once the missile communications link is established or GPS acquisition occurs, external missile measurements will be available and can be used to aid the missile navigation system. The IFA Kalman filter processes these external measurements and computes missile navigation-state corrections that then are fed back and used to adjust the navigation states. The main purpose of using external missile measurements is to minimize the errors in the missile's low-rate navigation data. Reducing error in the navigation-state estimates can be very beneficial because the overall missile-to-target error baskets will be lowered, and the probability of target acquisition will be increased. Figure 9 illustrates the guided missile navigation performance that can be obtained for three basic navigation configurations: free inertial operation (no aiding), free inertial aided by radar, and free inertial aided by radar and GPS.

For short times of flight, the most significant error in free inertial navigation is the initial attitude error, that is, the error in the initial estimate of missile attitude provided in the initialization message from the launch platform. Acceleration error in the navigation frame is produced by using a coordinate transformation that is corrupted by the initial attitude error. During free inertial operation, the acceleration error integrates into a velocity error that is proportional to the product of velocity and the INS attitude error. The position error is equal to the integral of the INS velocity error. During free inertial operation, the INS position and velocity error growth due to the initial missile attitude error quickly becomes significant. Figure 9 illustrates the dramatic improvement in the missile navigation-state estimates that can be obtained by incorporating external navigation-aiding data. Both radar only and radar plus GPS aiding can dramatically improve the missile navigation-state estimates.

In general, the error in the external radar missile-position measurements increase as the range between the radar and the missile increases. Thus, when radar measurements are the only aiding source available, the missile navigation position errors tend to increase as the range between the radar and the missile increases. However, the error in the radar position measurements is substantially less than the INS position error due to free inertial operation. Thus, a substantial improvement in the missile navigation-state estimates can be obtained by radar aiding. The error in the missile navigation-state estimates can be further reduced by incorporating GPS pseudorange and delta range measurements. The added benefit of GPS aiding is derived from the fact that the GPS "position" and "velocity" measurements are very good and remain essentially constant over time. GPS aiding therefore enables very accurate missile navigation position and velocity estimates with errors that also remain essentially constant over time.

Midcourse Phase

The midcourse phase of guided missile flight is fairly benign. During this phase, the missile is flying away from the launch platform toward the direction of the target, but it is not yet close enough to acquire the target with its terminal sensors. The missile navigation-system functions to maintain its navigation solution. The high-rate IMU acceleration and rotational rate data are sent to the missile autopilot and the lower-rate navigation-state data are provided to the autopilot, guidance, and various other missile functions.

Transition to Terminal (Target Search/Acquisition)

At some point during the missile flight, which is characterized by a predicted time-to-go or a range to the

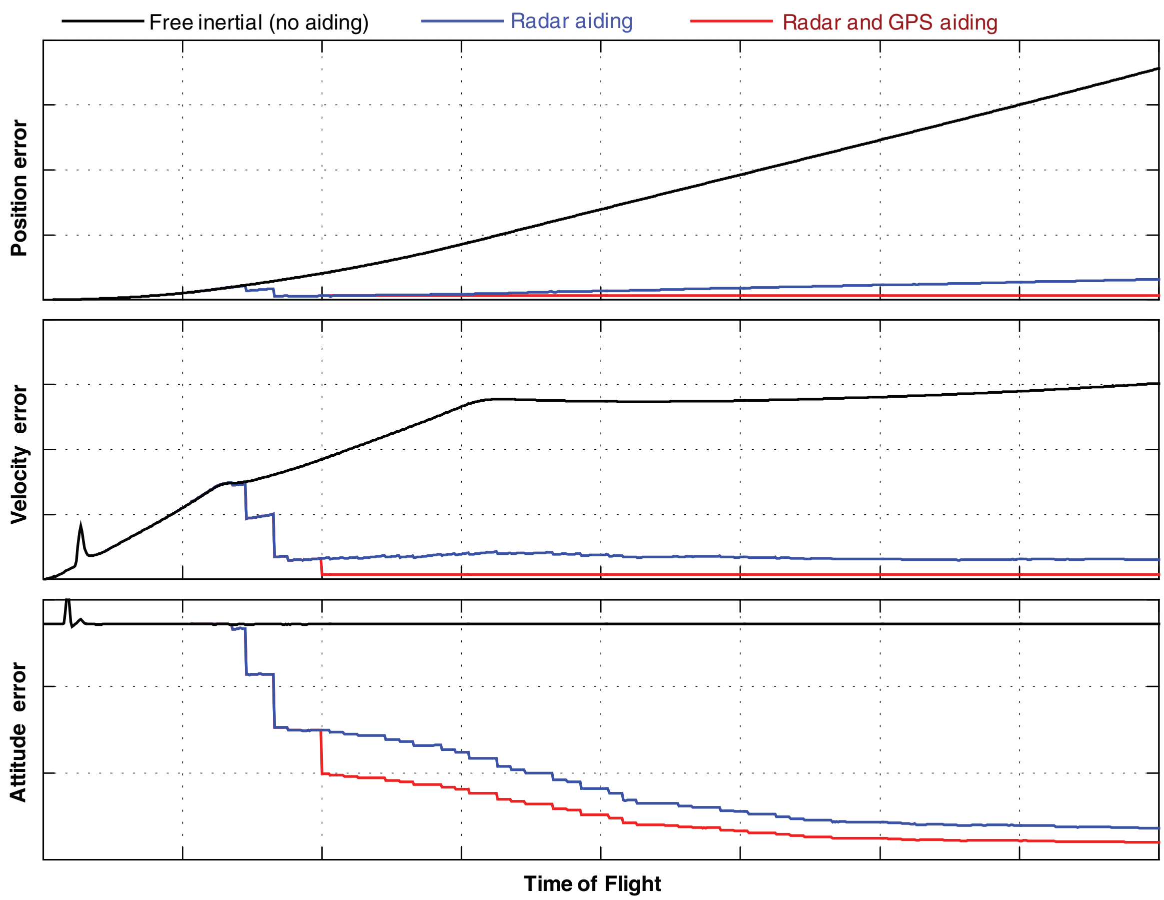

Figure 9. Navigation performance comparisons.

target, the guided missile system will power up its terminal sensor and begin to look for the target. In acquisition mode, the terminal sensor requires pointing commands to cause the sensor to look in the expected direction of the target. If the target is not found, the terminal sensor then will begin to search for the target in a region about the expected direction. The seeker-pointing commands and uncertainty regions can either be constructed in the weapon system and provided to the missile via the communication link or be generated internally within the missile. In either case, the relative missile-to-target state and uncertainty regions must be computed by appropriately combining the individual missile and target state and uncertainty data.

During the transition-to-terminal phase, the missile navigator still is responsible for maintaining its navigation solution. The high-rate IMU acceleration and rotational rate data are sent to the missile autopilot and terminal sensor control system, the lower-rate navigation-state and/or error-covariance data are provided to the autopilot, guidance, seeker search, and other functions.

Terminal Phase

The missile will begin terminal phase after the terminal sensor acquires and then confirms that it has successfully locked onto the target. If the terminal sensor does lose lock on the target, the sensor will require state and covariance information to attempt to re-acquire the target. Hence, missile-to-target state and covariance data should be provided to the terminal sensor for a possible target reacquisition.

During terminal phase, the missile navigator is still responsible for maintaining its navigation solution. The high-rate IMU acceleration and rotational rate data are sent to the missile autopilot and terminal sensor control system, the lower rate navigation-state and/or error-covariance data are provided to the autopilot, terminal guidance, and possible other missile functions such as the target detection device and the warhead.

FUTURE DIRECTIONS

Future improvements in the application and performance of INSs for guided missiles will be based on advances in technology and algorithms. Developments in navigation technology have focused on the reduction in size and cost of inertial instruments, improvements in GPS receiver design, and improvements in the GPS infrastructure. For guided missile systems, some of the important technology considerations are tolerance to shock, vibration, and temperature variations, with a continuing interest in reducing size and cost while maintaining accuracy. Currently, the use of tactical-grade systems having a gyro bias on the order of 1°/h has become generally accepted. Either ring-laser gyro or fiber-optic gyro technology can provide this level of performance with size, weight, and environmental specifications suitable for tactical guided missiles. The newest inertial devices employ microelectromechanical systems (MEMS) technology, although gyro accuracy is typically more on the order of 30°/h and may exhibit greater vibration sensitivity. Table 2 illustrates the differences between a typical ring-laser gyro IMU and a commercially available MEMS IMU. The volume is much less, but weight and power are comparable. The suitability of current MEMS capability should be evaluated for a specific application.

For tactical guided missiles employing GPS aiding, major considerations include the speed of initial GPS signal acquisition and vulnerability to jamming. Improvements in GPS receiver technology have focused on the number of correlators per channel and various methods to improve jam resistance. All-in-view receivers having 12 independent tracking channels and 1024 correlators for acquisition are standard for military application. The major trend is to increase the number of correlators to 4096 or more and to introduce additional anti-jam processing within the receiver. Another area of research has been the use of very tight coupling with the INS. This approach takes the tightly coupled configuration shown in Fig. 6 a step further by removing the traditional GPS code and carrier track loops in the receiver and embedding those functions within the single Kalman navigation filter. This technique is intended to provide additional anti-jam capability.

Aside from inertial instrument and GPS improvements, algorithmic improvements may lead to additional performance gains. For example, more careful modeling of the radar used for IFA can reduce the errors caused by radar refraction and inaccurate measurement time tags. Because inertial error depends on missile trajectory, a tighter integration between guidance and navigation could improve observability of system errors, leading to overall accuracy improvements. Finally, system architecture improvements could directly account for relative error between the targeting frame and missile seeker frame by employing a remote sensor track of the missile and/or improving relative alignment between targeting and missile-tracking sensors

Table 2. Comparison between ring-laser gyro and MEMS IMUs.

	Ring-Laser Gyro	MEMS
Gyro bias (°/h)	1	30
Volume (in.^{3})	32	4
Weight (lb)	1.9	1.2
Power (W)	6	6

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