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GPS BASED ATTITUDE DETERMINATION FOR SPACECRAFT: SYSTEM ENGINEERING DESIGN STUDY AND GROUND TESTBED RESULTS

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By differencing carrier phase measurements from multiple antennas, a global positioning systems (GPS) receiver can determine the attitude of a coordinate frame defined by the antenna baselines. This paper examines the potential role of such a capability within spacecraft avionics, The applications served by current GPS capabilities are identified, Architectural options are considered, and a baseline which satisfies the needs of most applications is defined. The majority of the paper then focuses of the prototyping of this baseline architecture within the Jet Propulsion Laboratory's (JPL's) Flight System Testbed (FST). The test setup is described, and test results are presented. The paper closes with an analysis of the limiting factors in the GPS based attitude determination error budget, a forecast of future capabilities, and a discussion of the advances that will be required to achieve those capabilities.

INTRODUCTION

Applications of Global Positioning System (GPS) receivers have proliferated well beyond the original vision of the system architects. Receivers are now being built which can measure translational velocity, attitude, attitude rate and time, as well as position. Moreover, technology is emerging which will make GPS receivers capable of routinely delivering highly accurate measurements of most of these states. Thus GPS receivers hold the promise of satisfying nearly all the guidance, navigation and control (GN&C) sensing requirements for Earth orbiting spacecraft in a single integrated, reliable, low mass, low volume, and low power package.

Between 1994 and 1996, a study was conducted at the Jet Propulsion laboratory (JPL) to define and prototype a GN&C avionics architecture around the particular strengths and weaknesses of GPS receivers, considering the requirements of a broad class of spacecraft with respect to performance, operations and fault protection. The study focused specifically on spacecraft attitude determination.

The paper presents the architecture that was decided upon plus the rationale behind it. It then details a test program designed to validate the concept within JPL's Flight System Testbed (FST), and describes the test results. It examines the classes of missions currently well served by available GPS technology, and discusses error budgets and the current limits of performance. Finally, the paper identifies future needs, describes the

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developments needed to bring about improvements in receiver performance, and provides a forecast of future capabilities.

BACKGROUND

The potential for spacecraft attitude determination using GPS measurements has been recognized for several years now. Reference 1, for example, describes simulation results for both single-antenna systems (received intensity based methods) and multiple-antenna systems (interferometric methods), the former requiring spacecraft acceleration or rotation measurements for full observability, An experiment on the RADCAL satellite, which is described in Reference 2, demonstrated that GPS attitude dc.termination could be done effectively in low earth orbit, although it must be noted that the RADCAL satellite was gravity-stabilized and not subject to rapid attitude changes. Other recent studies (see., for example, References 3 and 4) have focused on adapting commercial attitude-capable GPS receivers for space applications.

For high accuracy applications, a GPS receiver measures attitude by differencing carrier phase measurements from multiple physically separated antennas. A minimum of three antennas arc nccdcd for three degree of freedom attitude determination. The phase difference from any two antennas can be used to find the attitude of the baseline between those antennas with respect to the broadcasting satellite. To tic this attitude measurement to a useful frame of reference, like Earth fixed rotating (e.g., East-North-Up), the known location of the GPS satellite used in the observation and the solved for location of the receiver must be used, The former is tracked by the receiver using data broadcast by the satellites themselves. The latter is determined by the receiver, along with current GPS time (actually clock bias), by inverting time of flight measurements from a minimum of four satellites with suitable geometry. Indeed, this position determination is the primary function of a GPS receiv cr.

Notice that, in principle, observations from a minimum of two geometrically w c]] separated satellites are necessary for three degree of freedom attitude determination. A reasonable analogy is to consider a GPS receiver to be similar to a star tracker based attitude sensor. A minimum of two geometrically well separated stars (i.e., sources) arc needed for full attitude determination. In the case of a GPS based attitude sensor, the sources arc GPS satellites. in the case of a star tracker based attitude sensor, tbc position of the receiver dots not enter into the absolute attitude calculation, because the distances between the sources and receiver arc infinite. Therefore, the wavefronts that reach the receiver arc plane, with normals that point in the same celestial directions (the essentially fixed directions from the stars), regardless of where the receiver is. This is clearly not the case in GPS based attitude determination, where the received wavefronts arc spheres whose centers change with time.

Another complexity of GPS based attitude determination is that the carrier phase difference measurements do not uniquely determine the attitude of the connecting baselines. Integer wave ambiguities must be resolved for each particular satellite and each pair of antennas using an initialization procedure every time a new satellite is acquired. Once acquired, the integer portions of the relative phase solutions arc dynamically maintained. A number of methods have been developed to perform integer ambiguity resolution function. As this subject is beyond the scope of this paper, the reader is referred, for example, to References 5 and 6. Suffice it to say here that the requirement for such integer acquisition and tracking functions exist and must be accommodated in any architecture which plans to incorporate GPS based attitude determination.

GPS based attitude determination, like any other method of attitude determination, requires calibration of various system parameters. In the case of GPS, the most critical of these are the relative locations of the antennas and the radio frequency (RF) line biases. Fortunately, automatic procedures for calibrating both these sets of parameters have been developed. Automatic antenna location calibration has come to be known as self survey. This and the line bias calibration function need to be performed before the system can be made operational. Since accurate self survey requires relatively long records of observations (i. e., several hours), preferably under dynamically quiescent conditions, it is best performed, at least initially, on the ground.

GPS receivers track (i.e., process signals from) a number of (usually more than four) satellites at any given time, Selection of these satellites is aided by an almanac which allows the receiver to predict which satellites should be visible at any given time. The satellites used in any given position solution arc selected by the receiver to yield the best possible solution based on an assessment of their geometry. This is done by computing a so called position dilution of precision (PDOP) index for each satellite, A similar dilution of precision function can be used to select the. satellites to be used in the attitude solution.

Note that GPS satellites arc distinguished from each other by a pseudo random code. All satellites transmit at the same set of L-band RF frequencies (e.g., L1 at 1,575.42 MHz). Acquiring a particular satellite involves acquiring both its carrier and code.

Some GPS receivers compute velocity as well as position. This is done by using the Doppler shifts measured by the receiver in the course of carrier acquisition and tracking. In principle, attitude rates could be calculated from these measured Doppler shifts as well. '1'bus, a GPS receiver is in principle capable of directly measuring position, translational velocity, attitude, angular velocity and time, autonomously (i. e., 13 states), in real time, within a self contained package. For a more complete discussion of the GPS system, and the theory of operation of GPS receivers, the user is referred, for example, to References 4, 5 and 7.

The main error sources in a GPS attitude measurement arc receiver noise, finite resolution arithmetic, GPS satellite position errors, ionospheric propagation, tropospheric propagation, multipath, antenna location calibration errors, line bias calibration errors, and mechanical deformations (i.e., vibration, thermal deformation and material creep), The reader is again referred to the references (e. g., Reference 4 and 8 and the references contained therein) for a thorough discussion of GPS errors and their propagation into attitude measurements,

in current implementations, multipath limits performance to about

8x $10^{-3}/[L(T)^{1/2}]$ rad RMS, where L is the baseline length and T is the receiver integration time interval (Reference 8). Thus, for a 1 m baseline and a 1 sec integration time, performance is limited to just under 0.5 deg RMS. Multipath error estimation and compensation (see, e.g., References 8 and 9) can reduce this error substantially. However, this technology is just now emerging.

The next largest error sources are those associated with antenna phase center errors (i. e., calibration and drift). Collectively, these typically produce an attitude error of about lx $10^{-3}/1$. rad RMS (Reference 8). Note that errors due to mechanical deformations could easily become significant, indeed dominant, if not managed through careful mechanical and thermal design.

Next down the list of error sources is receiver noise at about $4x10^{-4}/[L(T)^{1/2}]$ rad RMS (Reference 8).

For relatively short baselines (i.e., a few meters), ionospheric error is negligible, For longer baselines, the ionospheric propagation delay at each antenna can be measured using dual or multiple frequency reception, and compensated to negligible levels.

Tropospheric propagation is negligible in space. GPS satellite position errors and numerical errors are negligible in genera].

ARCHITECTURE TRADE STUDY

Our particular interest was to define a GPS based attitude determination architecture that would serve the performance needs of most Earth Orbiting spacecraft and would be highly reliable and robust, Thus, we began by assessing the requirements of various classes of Earth orbiters.

Earth orbiters can be classified as either nadir pointed (spinning about their orbit normal at Earth's rate or dual spin), sun pointed (spinning about the sun line or 3-axis stabilized), or inertially pointed (spinners or 3-axis). Since GPS receivers must track GPS satellites, GPS receivers are not compatible with high rate (e. g., many RPM) spinners.

Missions that can tolerate attitude knowledge errors of a few degrees (e.g., broad coverage communication satellites) can be supported by either position knowledge correlation (in gravity gradient stabilized spacecraft) or magnetometers. In such missions, GPS receivers are most valuable for position measurement, and the additional cost of adding attitude capabilities may not be warranted,

Missions that require attitude knowledge on the order of a few tenths of a degree (e. g., most communication and remote sensing satellites) can be supported by either GPS receivers or Earth sensors and fine sun sensors. Since the state of the art in GPS technology is currently a few tens of a degree for 1 m baselines (see, e.g., References 3 and 5), these missions are ideal candidates for GPS based attitude determination. Discussion of the predicted future capabilities of GPS is deferred to a later section of this paper.

Missions that require attitude knowledge on the order of a few arc seconds or better (e.g., surveillance satellite and space based astrophysical telescopes) are normally supported by star sensors (imaging trackers and scanners). This level of performance is well beyond the current state of the art as well as the anticipated future capabilities of GPS. For these missions, the usc of GPS would be limited to position, velocity and time measurement. The usc of GPS based attitude determination for coarse knowledge is not likely to provide any benefits, except perhaps a level of functional redundancy, because emerging star trackers arc self initializing; that is they can determine their attitude autonomously without any a priori know] edge.

Onc other class of missions that is ideally served by GPS based attitude determination technology is orbiting interferometers. Since these missions involve very large spacecraft (10's of meters) or even discrete spacecraft separated by large distances (100's of meters to many kilometers), very large GPS baselines are possible. Neglecting structural deformation and antenna phase center knowledge errors, the attitude determination accuracy of interferometric GPS based attitude determination is given by the phase uncertainty divided by the baseline. Thus, given a 5 mm phase error (well within the current state of the art), the attitude of a 20 m baseline, for example, could be determined to about 250 μrad. This is in the neighborhood of tbc baseline attitude requirements for many (but certainly not all) of these missions.

As discussed above, GPS based attitude determination involves a number of interrelated elemental functions. Those that arc intrinsically RF functions arc as follows:

RF energy reception and amplification
RF down-conversion

'1'hose that arc either RF or high speed digital arc:

signal acquisition
signal tracking

'1'hose that arc computational functions arc:

satellite ephemerides maintenance
integer ambiguity solution
integer tracking
self survey
line bias calibration
position and time solution
translational velocity solution
position dilution of precision
attitude dilution of precision
attitude solution
angular velocity solution

in addition to these basic functions, a fully autonomous spacecraft GN&C system must also include the following elemental functions, which arc computational by nature:

attitude data fusion
attitude filtering
attitude rate estimation

- attitude maneuver planning and constraint avoidance
- attitude maneuver control
- attitude stabilization
- momentum management
- attitude control system fault protection
- position data fusion
- position filtering
- velocity estimation
- orbit determination
- trajectory planning and optimization
- delta-v maneuver planning and scheduling
- delta-v maneuver control
- thruster fault protection

Besides GPS receivers, other attitude sensors which may contribute measurements for data fusion arc gyros, Earth sensors, sun sensors, star sensors, and magnetometers. Possible non-GPS sources of position data include accelerometers, image based optical navigation sensors and ground tracking facilities.

Onc question that was considered was, given that a spacecraft already contains onc or more capable flight computers to host basic GN&C functions, would it be wise to host the GPS specific computational functions in onc of these computers as well. Certainly there is an economic advantage (ignoring non-recurring costs), to limiting the number of computers on board a spacecraft. Moreover, GPS based position and attitude acquisition and tracking may be facilitated by using feedback from position and velocity state estimators executing within the same computer; though this dots not absolutely require integration of GPS and other GN&C functions within onc computer, it is more convenient in that case.

Another question considered was, given the characteristics of a GPS receiver, what other sensors should be included in the avionics suite to optimize overall system performance, hasten acquisition and enhance reliability (i. e., fault protection). As mentioned above, GPS receivers cannot acquire and track satellites under high spacecraft rotation rates. information from inertial attitude sensors could be used to feed rate information into the receiver, allowing it to overcome this limitation. Also, inertial attitude and position sensors could be used to reliably (i. e., independent of excitation assumptions) propagate from GPS attitude and position updates, under both nominal conditions and periods of measurement gaps.

The baseline architecture that was decided upon for attitude determination consists of a GPS receiver and a low cost, tactical grade, inertial reference unit (IRU), both of which may be redundant in some applications, and a Kalman filter attitude estimator which fuses the data from the two types of sensors. The GPS receiver provides position, translational velocity, attitude and time measurements. The function of the IRU is to aid acquisition, to provide fault protection (e.g., to stop a tumble), and to allow propagation in case of gaps in the GPS based attitude measurements. The G1'S based attitude rate measure.mcnts arc not accurate enough to be Useful, and cannot be relied upon in high rate situations, such as those that may be encountered in certain fault scenarios. The architecture is shown in Figure 1.

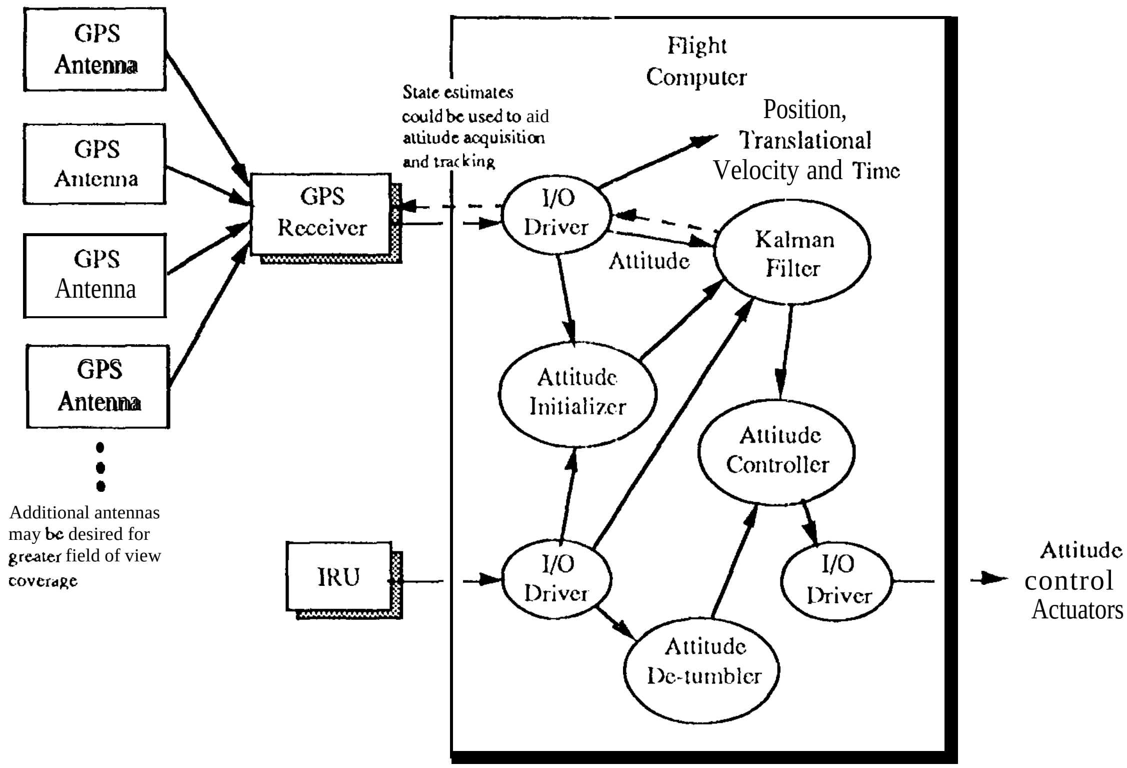

Figure 1. Basic Attitude Avionics Architecture

The figure above emphasizes the attitude sensing operations, However, one of the major benefits of having a GPS receiver within the architecture is that doing so allows a fully autonomous station keeping function to be implemented. Such a capability would operate off of the GPS position measurements, and would plan and execute velocity correction maneuvers as needed. Also not emphasized her-c. is the use of velocity and accurate time measurements. These would, among other things, greatly simplify payload operations (e. g., instrument pointing) and allow many of them to be automated on-board.

The selected baseline allows the direct use of currently available GPS equipment, and thus minimizes development time and costs. It also avoids loading down the GN&C flight computer or computers with GPS maintenance functions. Use of tactical gyros for rate measurement and a Kalman filter, which performs data fusion and filtering, and which allows extrapolation in case of GPS data drop outs, provides a low cost way of compensating for any weaknesses in the use of GPS alone. A readily available, low cost, 1 deg/hr tactical grade gyro, for example, would make a 10 second GPS data drop out unnoticeable. Feedback from the gyro and Kalman estimator can still be used to aid GPS attitude processing within this architecture with only minor modifications to the receiver software.

A more tightly integrated architecture may be warranted in certain special micro-spacecraft applications, where volume and mass are at a premium. However, it should be kept in mind that the computational requirements of an attitude capable GPS receiver keep an M68000 class processor busy producing 1 Hz updates. Thus, for such a tightly integrated architecture to be feasible, either the non-GPS processing requirements must be modest, or the flight computer must be many times more capable than an M68000 class machine. Another penalty of a tightly integrated architecture is a loss of functionally and physically separated (i. e., natural) fault containment regions.

TEST PROGRAM

The study included a ground test program to characterize the frequency, duration and nature of measurement gaps, especially attitude measurement gaps, and to assess the variability of the measurements in normal operation.

The specific objectives of the program were as follows:

1. Prototype a GPS based spacecraft attitude determination capability. Work out the architectural and interface aspects of incorporating such a function within future JPL spacecraft.

2. Contribute prototype GPS hardware and software to a spacecraft rapid prototyping testbed facility.

3. Develop a working familiarity with GPS operation, performance and reliability.

4. Validate the proposed GPS based spacecraft attitude determination architecture.

The test program was comprised of the following three phases.

1. Characterize the performance and reliability of GPS attitude measurements.

2. Validate the proposed GPS based spacecraft attitude determination architecture using real GPS measurements and simulated gyro measurements.

3. Replace the simulated gyros with real gyros,

Test Setup

The test bed consisted of an attitude capable commercial GPS receiver, four antennas, a rotatable roof top fixture with adjustable baseline dimensions, a low performance (i.e., 0.1 deg/sec drift instability) three axis IRU, power supplies and cabling. The test set-up exploited the resources of JPL's Flight System Testbed (FST), which include a network of computers and software that together are designed to emulate an end-to-end space system. In particular, a VME based real time computer that is part of the FST was used to host a spacecraft attitude estimator, and the FST's space-to-ground telemetry simulator was used to archive the receiver data for later, off line, analysis. The test system is depicted in Figure 2.

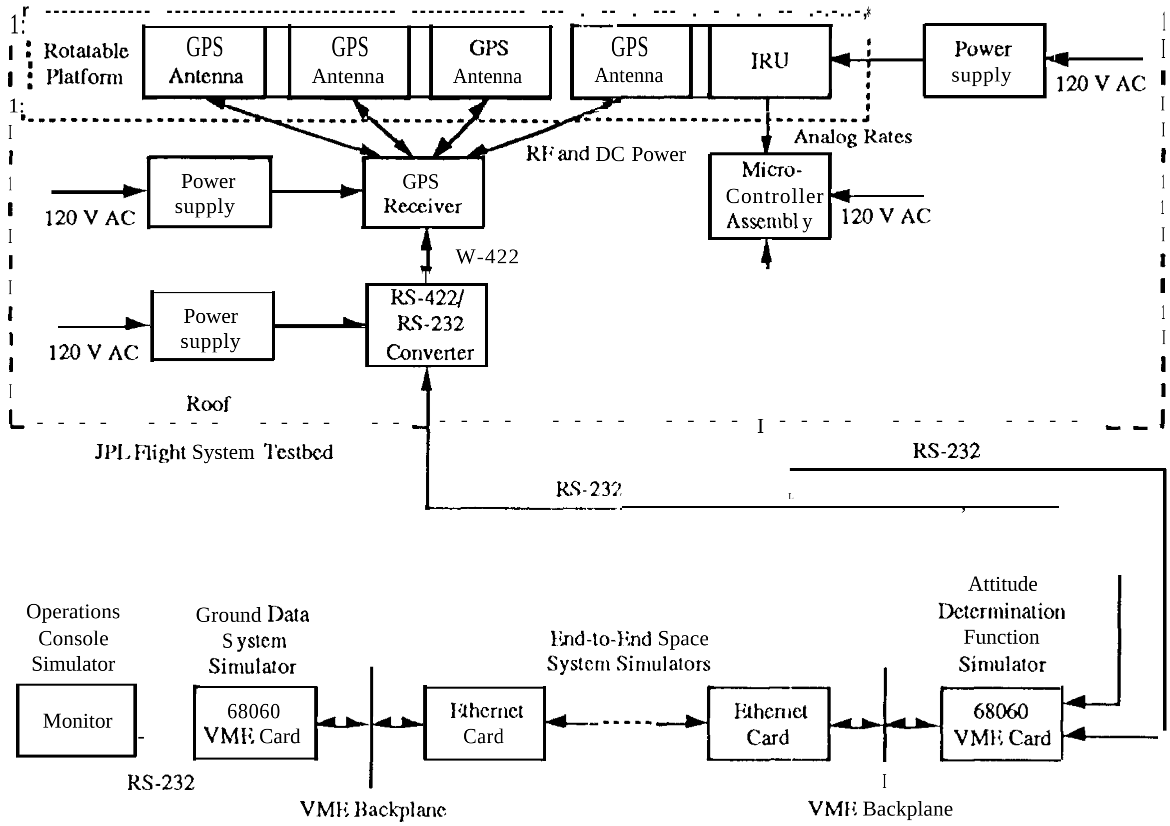

Figure 2. GPS Ground Test Set-up

The antenna support structure is shown in Figure 3. It is fabricated entirely from aluminum stock (i.e., I-beams, tubes, angles, and sheet), most of which was scrap. The four antennas and the IRU are mounted on an A-frame which has field adjustable legs. Baselines from roughly 0.5 to 3.5 m can be realized by changing the length of these legs. The A-frame is mounted to a base through a spherical bearing. The base is secured by cinder blocks.

Attitude motion is simulated by manually rotating the frame. A long term goal, funding permitting, is to mechanize the antenna platform to enable hardware-in-the-loop closed loop simulations.

When mounted on the stand, the antenna frame is secured by ropes tied to the base. During long data collection periods, the frame was set down on cinder blocks for better security and stability.

A photograph of the rooftop setup is shown in Figure 4. The picnic cooler lying on the floor contains all the rooftop electronics. The lid is left ajar for ventilation. Precipitation is prevented from entering the cooler by a sheet of plywood which serves as an over sized roof. Within the cooler, the electronics are on a platform raised above the. level of the drain in case any moisture does make it past the roof. No difficulties have been experienced from either heat, rain or wind after over six months of operation.

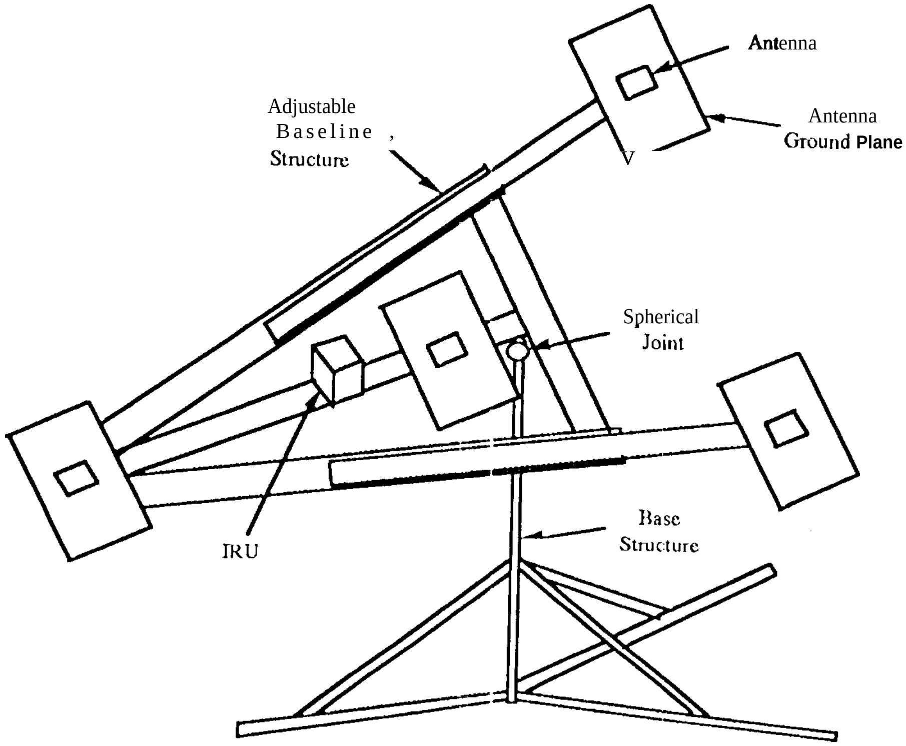

Figure 3. Antenna Assembly

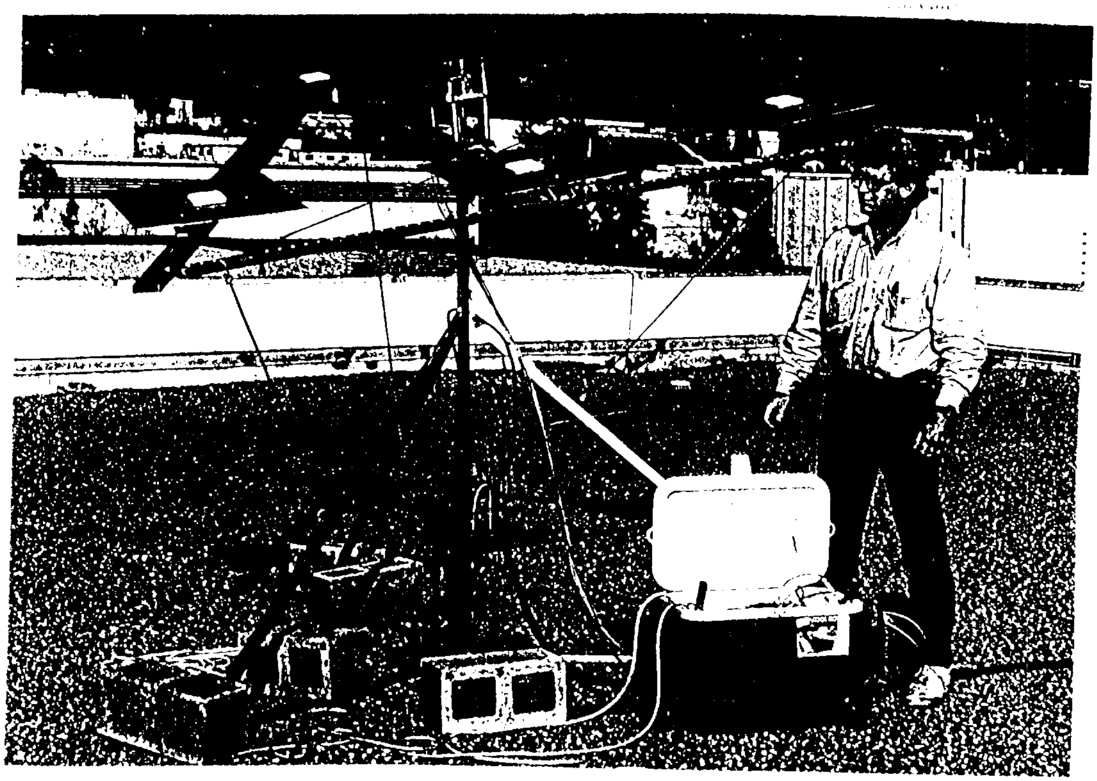

Figure 4. Rooftop setup

This equipment and the associated software are now permanent FST resources, available to JPL flight projects for rapid prototyping and hardware-in-the-loop integration and test simulations.

Kalman Filter

The attitude estimator has two components. One propagates the attitude estimate as data is received from the inertial reference unit (IRU). The second is the Kalman filter itself. When GPS data is received, the difference between the attitude estimate propagated from IRU data and the GPS estimate is calculated and the result fed to the filter.

The observables are attitude quaternion q_{GPS} (from the GPS receiver) and gyroscope rates ω (3-D from the IRU).

The gyro measurement based attitude propagation equations are

$$q_k = \frac{\Delta q_k * q_{k-1}}{\Delta q_k * q_{k-1}}$$

$$\Delta q_k = (0.5 * \Delta\overline{\theta_k}, g)$$

$$g = 1.0 - (\Delta\overline{\theta_k} + \Delta\overline{\theta_{k-1}})^2 / 32$$

$$\Delta\overline{\theta_k} = \rho_k(\tau_k - \tau_{k-1})$$

$$\rho_k = GM \frac{\omega_k + \omega_{k-1}}{2} - \overline{DR_0} - \overline{DR}$$

where

ω_k is the angle rate measurement received from the IRU at time τ_k,
GM is a coordinate transform matrix from IRU coordinates to body coordinates,
$\overline{DR_0}$ is the baseline gyro drift rate, in body coordinates,
DR is the estimated gyro drift rate (increments estimated by Kalman filter),
ρ_k is the corrected rotation rate for time τ_k,
$\Delta\theta_k$ is the calculated angular increment between time τ_{k-1} and τ_k, and
q_k is the propagated attitude quaternion for time τ_k.

The Kalman filter inputs are taken from the normalized quaternion $\kappa_n = q_{GPS_n} q^*_{IRU_n}$, where κ_n describes the difference between the GPS and IRU coordinate frames at time t_n. The actual observation vector r_n is composed of the first three components of κ_n.

The basic filter equations arc

Propagation:

$$x_n\ (-) = \Phi_n x_{n-1}(+)$$

$$X_n\ (-) = \Phi_n X_{n-1}(+)\Phi_n^T + \dot{Q}*(t_n - \text{tn.,})$$

$$\Phi_n = \frac{\partial x_n}{\partial x_{n-1}}$$

Update:

$$r'_n = r_n - Hx_n(-)$$

$$R'_n = R_n + HX_n(-)H^T$$

$$x_n(+) = \text{X}_\text{n}(-)\text{+XJ-)H\%;'17-'}$$

$$X_n(+) = \text{X}_\text{n}(-) -- \text{X}_\text{n}(-)\, H^T R_n'^{-1} H X_n(-)$$

$$H = \frac{\partial r_n}{\partial x_n}$$

where

x_n is thc filter state for the nt[h] timc interval,

X_n is the corresponding statc covariancc matrix,

$\dot{Q}$ is the process noise covariancc matrix (expressed as a rate),

Φ_n is the state transition matrix for the nth time interval,

r'_n is the Kalman filter innovation,

R_n is the mcasurcmcnt noise covariancc matrix, and

H is the mcasurcmcnt sensitivity matrix.

The state vcctor was defined as the real 6-vector consisting of the first three components of the normalized attitude error quatcrnion followed by the three gyroscope bias drift rates. The corrcsponding Transition matrix, Φ_n, has the form

$$\Phi_n = \begin{bmatrix} \Phi_{AA} & \Phi_{AG} \\ 0 & I \end{bmatrix},$$

where $\Phi_{AA} \approx I - \Theta^\times$ and $\Phi_{AG} \approx I*(t_n - t_{n-1})$, provided that the change in attitudc bctwccn t_{n-1} and t_n is small.

The overall signal processing flow is as follows.

IRU Collector:
1. Request IRU data
2. Read IRU data from previous request and accumulate.
3. If on 100 ms boundary, call IRU propagator.
4. Zero IRU accumulator.
5. Sleep for 10 ms.
6. Go to 1.

IRU propagator:
1. Apply IRU propagation equations to obtain attitude estimate.
2. Log attitude estimate and time of estimate in circular buffer.

GPS Data collection and Kalman filter:
1. Read GPS packet.
2. Use packet time tag to find appropriate attitude estimate in log buffer.
3. Calculate attitude correction measurement from GPS data and corresponding IRU-propagated estimate.
4. Feed attitude correction measurement to Kalman filter to calculate current state (attitude correction) estimate.
5. Apply attitude correction estimate into all attitude estimates with times greater than or equal to the GPS time tag; zero Kalman filter state estimate.
6. Go to 1.

The attitude update process of the GPS receiver used in this study is free running with a nominal rate of approximately 1 Hz. IRU rate data is collected asynchronously at a rate of 100 Hz using a simple query-response communication protocol under the control of th flight computer simulator. Attitude is propagated forward in time, from the last best estimate, at a rate of 10 Hz using IRU measurements averaged over 10 samples and corrected with the best available estimates of gyro biases. The 10 most recent samples of this attitude are stored at all times. When a GPS sample arrives, the associated time tag is used to locate the most appropriate propagated attitude sample. This, together with the GPS measurement, is then used to develop the Kalman filter innovation. After a Kalman filter update is processed, the best estimates of attitude and gyro biases, used in subsequent attitude propagation operations, are redefined. The Kalman filter state estimates are thus zeroed.

GPS Receiver Test Results

GPS data was collected and archived for several weeks. The archived information included the receiver's estimates of attitude and reports of receiver and satellite status. The commercial receiver wc used computes attitude estimates at a rate of 1 Hz; outages were defined as. intervals of 3 seconds or more between reported estimates.

During the three week period between October 19, 1995 and November 6, for example, there were a total of 49 data gaps of varying durations; these are.

listed in Table 2. Only two were longer than 100 scc (2673 and 1638). All but twelve were shorter than 10 scc. Twenty three were four seconds or shorter.

Table 2. Data Gap Results

Week Number	Time of week (seconds)	Duration (seconds)
827	498500.5 “	4
827	559796.5	95
827	559893.5	7
827	572329.5	5
828	7.5	75
828	38631.0	4
828	67448.0	4
828	99299.0	6
828	99309.0	7
828	99318.0	56
828	101673.5	3
828	172615.0	4
828	203590.5	2673
828	206268.5	78
828	206385.5	60
828	217243.0	4
828	254545.5	3
828	384441.5	4
828	493609.5	11
828	493621.5	5
828	493627.5	5
828	493632.5	8
828	493641.5	7
828	493649.5	4
828	493661.5	10
828	493674.5	4
828	533532.0	4
829	5840.0	4
829	15238.0	4
829	68274.5	4
829	156640.5	4
829	390258.5	54
829	406939.5	3
829	424590.0	4
829	466516.0	5
829	518272.5	5
829	534682.5	55
830	35679.5	6
830	35743.0	8
830	54037.0	5
830	126979.0	4
830	185717.5	1628
830	206165.5	4
830	292573.0	4
830	537956.0	6
830	537963.0	56
830	595260.0	4

830	595731.0	4
831	4320.5	4

The receiver status reports allowed determination of proximate causes for outages in a straightforward manner: during outages, status values were tabulated. None of the outages proved to be due to-failures in GPS satellite coverage. Our working hypothesis is that most of the gaps are simply due to the time it takes to resolve integer cycle ambiguities at satellite acquisition times. This time is usually less than one second, and hence is usually unnoticed. However, occasionally the integer solution takes longer. A few of the gaps are clustered; in these clusters, successive gaps are separated by a few seconds to less than two minutes. These might indicate ambiguities induced by multipath reflections under conditions of poor viewing geometry.

The GPS receiver's attitude estimates were well within expectations. The means and standard deviations, respectively, for attitude components over the three week period discussed above were

pitch:	(-0.007°, 0.136°)
roll:	(-0.088°, 0.110”)
azimuth:	(215.795°, 0.078°)

for the configuration shown in Figure 4 with approximately 2 meter baseline separations. ‘The covariance matrix of these same measurements was

	Pitch	Roll	Azimuth
Pitch	0.01858	-0.003236	-0.001082
Roll	-0.003236	0.01207	0.0009515
Azimuth	-0.001082	0.0009515	0.006145

Temporal autocorrelations of deviations from the mean were also calculated from the archived data. While the cross-correlations between pitch, roll, and azimuth were negligible for all time, it was found that successive attitude measurements were highly correlated over time scales exceeding an hour. These data are summarized in ~’able 3.

Table 3. Autocorrelations for attitude measurements.

Elapsed Time (Sees)	Pitch	Roll	Azimuth
00000	1.0000	1.0000	1,0000
00001	0.9472	0.92’73	0.6695
00002	0.9452	0.9212	0.6655
00003	0.9445	0.9140	0.6618
00004	0.9409	0.9049	0.6587
00005	0.9403	0.8989	0.6542
00006	0.9393	0.8910	0.6514
00007	0.9396	0.8850	0.6481
00008	0.9397	0.8791	0.6459
00009	0.9393	0.8722	0.6430
00010	0.9403	0.8671	0.6409
00020	0.9369	0.8484	0.6288

00030	0.9334	0.8752	0.6115
00040	0.9293	0.8675	0.5929
00050	0.9250	0.8507	0.5748
00060	0.9212	0.8485	0.5557
00070	0.9177	0.8467	0.5384
00080	0.9147	0.8415	0.5219
00090	0.9119	0.8375	0.5058
00100	0.9093	0.8346	0.4923
00200	0.8862	0.8092	0.3851
00300	0.8672	0.7884	0,3001
00400	0.8528	0.77s9	0.2460
00500	0.8411	0.7687	0.2112
00600	0.8315	0.7602	0.1837
00700	0.8229	0.7513	0.1715
00800	0.8117	0.7400	0.1729
00900	0.8022	0.7296	0.1791
01000	0.7935	0.7224	0.1777
02000	0.7203	0.6943	0.00977
03000	0.6480	0.6663	0.00483
04000	0.6079	0.6471	0.02588
05000	0,5587	0.6234	-0.02390
06000	0.5033	0.5883	-0.06812
07000	0.4565	0.5501	-0.07200
08000	0.4224	0,4917	-0.01822
09000	0.3917	0.4614	0.01399
10000	0.3617	0.4473	-0,05202
11000	0.3236	7).3969	0.02306
12000	0.3048	0.3395	0.06407
13000	0.2797	0.3011	0.06387
14000	0.2542	0.2575	0.01099
15000	0.2300	0.2271	-0.04710
16000	0.2164	0.1965	-0.06888
17000	0.2042	0.1552	-0.04117

The observed autocorrelations are surprisingly high, Clearly, there is an unmodelled source (or sources) of error. A few reasonable hypotheses can be identified (e. g., multipath reflections, temporal variations in line biases), but further investigation is clearly warranted. If the cause(s) were to be identified and included in the estimation model, simple calculations suggest that attitude error could be reduced by as much as a factor of 10.

CONCLUSION

The test program confirmed that GPS measurement gaps, though relatively infrequent, arc to be expected, and that the architecture described in this paper is effective in maintaining system performance through such outages. Further study is needed to determine what type of gyros will be necessary to guarantee a specified level of performance.. However, the 0.1 deg/hr gyros wc used in our prototype arc capable of preserving attitude knowledge to about 1.0 degree during all but 10 of the 49 data gaps wc observed during one three week period, those 10 gaps amounting to less than 1.9 % of the total observation duration. Measured GPS attitude measurement performance was consistent with the receiver manufacturer's specification.

FUTURE PLANS, POSSIBLE FUTURE DIRECTIONS AND PROJECTIONS

The autocorrelation results shown in Table 3 above will be extended to cover a 24 hour period. This will expose temperature dependent diurnal effects as well as **12** hour periodicities due to the GPS satellite orbits.

Results with simulated and real gyros were. not available in time to appear in this paper. Completion of system integration and test, and demonstration of full system functionality is scheduled to occur within the next month.

A follow-on to the test program described in this paper is currently being negotiated. 'I'his fourth phase will be aimed at

1. developing a thorough understanding and explanation of the results that were observed in phases 1 through 3

2. developing bounds on data gap statistics

3. developing contingency measures for events such as excessive data gaps and receiver lock-up.

The results of this follow-on activity will make it possible to fly a GPS based attitude determination capability in space with con fidence, and will allow rigorous specification of the types of gyros needed to ensure a given level of system performance.

As mentioned in the text, a long term facility oriented goal is to mechanize the antenna platform to enable hardware-in-the-loop closed loop simulations. This would involve the addition of motors and associated electronics and drive couplings, and truth sensors (e. g., precision encoders) to the rooftop equipment. Additional cabling and testbed software will also be required.

The uses of GPS have already far exceeded the vision of the original system architects. The full potential will undoubtedly involve the continued development of many GPS technologies. Indeed GPS offers fertile ground for future technology development efforts in several areas. Multipath error, being the current dominant error source, will certainly benefit from ongoing research into suppression techniques. Reference 9, for example, describes an algorithm which can reduce the effects of multipath error to about the level of the receiver noise. Phase ambiguity initialization is another potentially fruitful avenue of research. Superior ambiguity solvers could reduce or eliminate the types of data gaps we observed in our test program, further relaxing the demands on adjunct gyros. The feed forward or feedback of information from outside the receiver to aid CiPS acquisition and tracking, which is allowed by the architecture defined in this paper, requires detailed development in order to be realized. This will involve making software changes inside the receiver and developing the necessary information exchange protocol. On line self survey and line bias calibration, possibly with the aid of data from other sensors, could be used to compensate for antenna phase center drifts. Finally, new hardware implementations offer large potential payoffs. For example, JPL's proposed "GPS on a chip" (Reference 10) promises to package the benefits of a dual frequency, precision, full function (i.e., time, position, velocity, attitude and attitude rate) receiver into a module roughly the size of a cigarette pack. This has obvious implications for micro spacecraft.

The absolute limits of performance of GPS attitude sensing are difficult to assess. However, performance projections for the year 2000 were developed in Reference **8.** These predict an RMS attitude error of about 600 μrad (i.e., about 2 arc min or **0.035** deg) for 1 m baselines and 1 sec integration through a combination of advances in multipath suppression, low noise amplifiers, and line bias (i.e., antenna phase center) calibration and tracking, Should these projections become fact, GPS based attitude sensing will be far superior to all but star tracker based sensing,

A possibility worth exploring for missions that can tolerate a few degrees of attitude knowledge error is GPS intensity based attitude determination. This non-interferometric method calculates attitude essentially by operating on received intensities with the inverse of the antenna beam patterns. Accurate phase measurements and ambiguity resolution are not required. It has been estimated that attitude can be determined to about 5 degrees by this method using only two non-aligned antennas and averaging over many satellites (Reference 11).

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