GPS Based Attitude Determination for the Flying Laptop Satellite

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Abstract

This paper introduces the GPS based attitude determination system (GENIUS) onboard the university small satellite Flying Laptop. The attitude determination algorithm which is based on a Kalman Filter and processes single differences of the C/A-code and carrier phase measurements is shortly described. The algorithm uses the LAMBDA-method to resolve the integer ambiguities of the double differences of the carrier phase measurements. These resolved ambiguities are then used to fix the single difference ambiguities in the filter. The results of ground based tests and numerical simulations are introduced and the accuracy of the attitude determination algorithm is assessed.

1 GENIUS Experiment on the Flying Laptop

The micro-satellite Flying Laptop is currently under development at the Institute of Space Systems, Universität Stuttgart [1]. The satellite has a mass of about 100 kg, is three-axis stabilized and will be launched in a polar, sun-synchronous, low earth orbit.

GENIUS (Gps Enhanced Navigation Instrument for the Universität Stuttgart micro-satellite) is an experiment for GPS based determination of the spacecraft attitude which

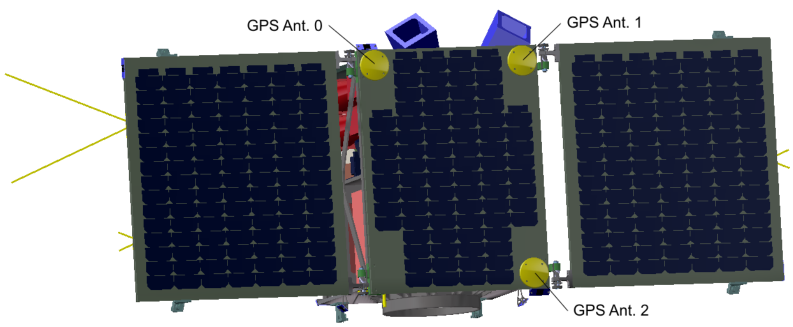

Figure 1: GPS satellite antennas in L-shaped arrangement

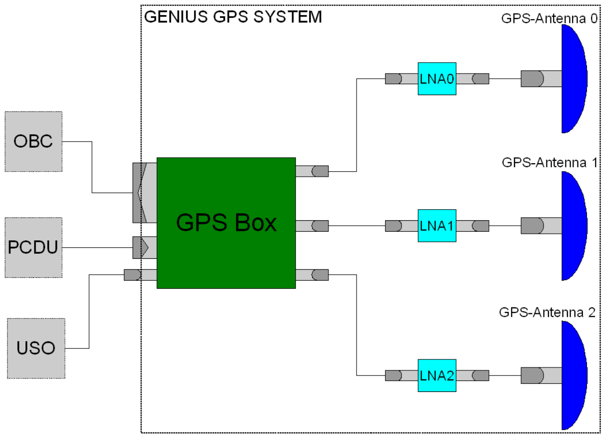

Figure 2: GENIUS system overview

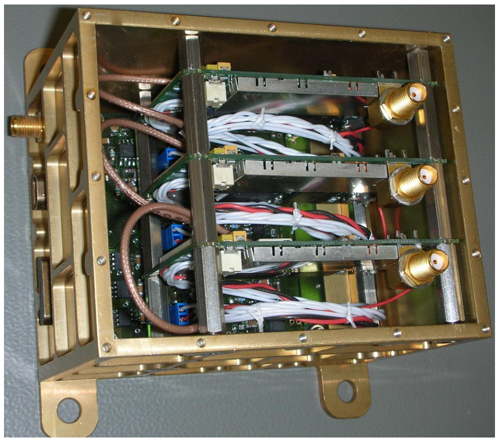

Figure 3: GPS Box hardware assembly (opened)

is developed in cooperation with the German Space Operation Center (DLR/GSOC). It also supplies the standard real-time position and velocity needed by the attitude control system for the nadir and target pointing mode. The GENIUS GPS system consists of three independent GPS receiver boards, each connected to a separate antenna and low noise amplifier (LNA) as shown in Fig. 2. The GPS Box is connected to the on-board computer (OBC), the power control and distribution unit (PCDU) and the ultra stable oscillator (USO). The used Phoenix boards are commercial 12-channel GPS L1 receivers with a DLR/GSOC developed firmware for space and high dynamics applications [4]. Three GPS antennas are mounted on the middle solar panel in an L-shaped arrangement, creating two baselines with a length of 440 mm and 610 mm respectively (Fig. 1). The antennas are pointing in the opposite direction of the payload cameras and therefore have optimum visibility of the GPS constellation during Earth observation. The three GPS receivers are integrated in a single $100 \times 80 \times 67$ mm box together with an interface board for RS-422 conversion, the USO signal distributor and a latch-up protection for each receiver (Fig. 3). To achieve a high level of redundancy, each receiver can be switched on/off independently varying the system input power from 0.9 W for 1 receiver to 2.6 W for all 3 receivers according to measurements at the testing model. Among other hardware modifications to prepare the receiver boards for space usage, the oscillators were removed to synchronize all receivers to the central 10 MHz USO. This external oven controlled crystal oscillator with a high accuracy of 10^{-13} over a period of 1000 s drives the clocks of the receivers and also eliminates variations between individual receiver clocks.

GPS position and velocity are sent to the OBC every second, while the raw carrier phase measurements are recorded every 10 s and will be dumped during ground station contacts. With the algorithm described below, the satellites's attitude is calculated post facto and can be compared to the star tracker reference data for accuracy assessment.

2 Attitude Determination Algorithm

GPS attitude determination with carrier phase measurements is based on the principal of interferometry. Fig. 4 depicts a carrier phase signal from a GPS satellite arriving at two antennas separated by the distance b. The differential range from the antennas to the GPS satellite consists of the fractional wavelength $\Delta\Phi$ which can be measured by the receiver and the number N of complete cycles, which is unknown and therefore called integer ambiguity. If N is known, the angle θ between the baseline and the line-of-sight vector to the GPS satellite can be determined via

$$\cos\theta = \frac{\Delta\Phi + N}{b} \tag{1}$$

Having four or more measurements from different GPS satellites, the differential antenna vector (baseline) and the differential clock offset of the receivers can be determined [3]. Thus, GPS based attitude determination can be separated into two different problems: First, the ambiguities of the carrier phase measurements must be resolved prior to be able to use these measurements as precise range measurements. Secondly, a filter algorithm must be implemented to improve the accuracy of the attitude solution. In this section the filter algorithm shall be explained first and afterwards the ambiguity resolution procedure shall be introduced.

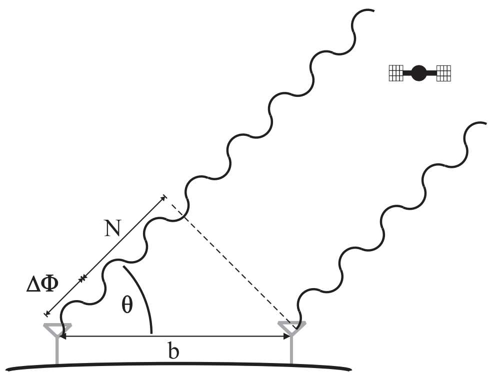

Figure 4: Single baseline GPS interferometry for attitude determination

The attitude determination algorithm used for GPS based attitude determination of the small-satellite Flying Laptop is based on a Kalman Filter. The filter state consists of two baseline vectors, which are the relative position vectors between the GPS antennas. Since the GENIUS experiment employs three independent Phoenix GPS receivers, the differential receiver clock offset between two receivers involved in each baseline must be estimated. Finally the angular velocity vector of the satellite is estimated along with the ambiguities of the carrier phase measurements for each baseline. The propagation of the relative antenna motion is based on a simple kinematic model, which assumes a rotation of the satellite body with a constant angular velocity vector. The differential clock offset and the ambiguities are assumed to be constant over time. Note that no differential equations must be solved in this case, since the change in the baseline can be propagated using a simple rotation matrix computed from the angular velocity vector. Since this simplified system model is only an approximation of the truth, process noise is included in the filter to avoid a filter divergence due to unmodelled effects. In this case, variations of the satellite's angular velocity vector are compensated by angular accelerations, which are modelled as Gaussian noise with zero mean. Variations in the differential receiver clock offset are modelled as a clock drift which consists of Gaussian noise with zero mean. No process noise is added to the ambiguity estimates, since their values do not change as long as the satellite is tracked continuously.

The filter processes inter-receiver single differences of the C/A-code and carrier phase measurements. The advantage of using differencial measurements is that errors common to both measurements cancel out. In this case the GPS satellites' clock offset and all error which originate at the signal transmission path are eliminated. The remaining errors are due to receiver noise, multiple signal reception at the GPS antenna and antenna phase center variations. The angular velocity is not directly observable through the measurements and must therefore be estimated by the Kalman Filter.

As already mentioned previously, the ambiguities of the carrier phase measurements must be resolved before these observables can be used for attitude determination. There exists a great variety of ambiguity resolution methods in the literature. One of the most powerful methods is the LAMBDA-method [5], which can resolve the integer ambiguities of carrier phase double differences based on a float estimate of these ambiguities and their corresponding covariance matrix. The attitude determination algorithm performs a check for unresolved ambiguities after each propagation step. If unresolved ambiguities exist, the estimates of the single differences from the filter state are transformed to double differences using a common reference satellite. These float estimates of the double difference ambiguities are passed to the LAMBDA-method, which tries to resolve the integer values. If the solution of the integer ambiguities passes a validation scheme, they are used to fix the single difference ambiguities in the filter state with a second measurement update. During this update, the integer double differences are used as "pseudo"-measurements of the single difference ambiguities. Since the double difference ambiguities are known exactly from the LAMBDA-method, their standard deviation is set to zero during the measurement update. As a result, the entries in the filter covariance matrix for the ambiguities are reduced to zero, which prevents the filter from updating the ambiguities during the following epochs. Thus the "pseudo"-measurement update allows to constrain the single difference ambiguities in the filter efficiently.

Finally, the attitude information must be computed from the baseline vectors. The Kalman filter estimates the baseline vectors in the earth-fixed earth-centered frame. Additionally, the vector components of both baseline vectors in the satellite's body-fixed frame are known from pre-flight evaluations. The Triad-Method or the QUEST-Method are used to process the information to attitude coordinates parametrized as a rotation matrix or quaternions.

Fig. 5 depicts the complete attitude determination procedure. The algorithm starts at each epoch with an editing of the available GPS satellites. At this point satellites below a certain elevation threshold or below a minimum signal-to-noise ratio are excluded. In the next two steps newly acquired and lost GPS satellites are included or excluded from the filter state. Then, a residual check of the carrier phase measurements is preformed for the satellite with already resolved ambiguities to detect if the ambiguities of individual satellites have change by cycle slips. If the residual check fails, the ambiguity of the affected satellite is resetted and marked as unresolved to ensure that the ambiguity resolution procedure is repeated for this satellite. After the measurement update of the Kalman Filter, the ambiguity resolution branch described previously is entered if necessary. Finally, after the baseline vectors are been transformed to attitude parameters, the state vector and the state covariance matrix are propagated to the next epoch and the complete procedure is iterated.

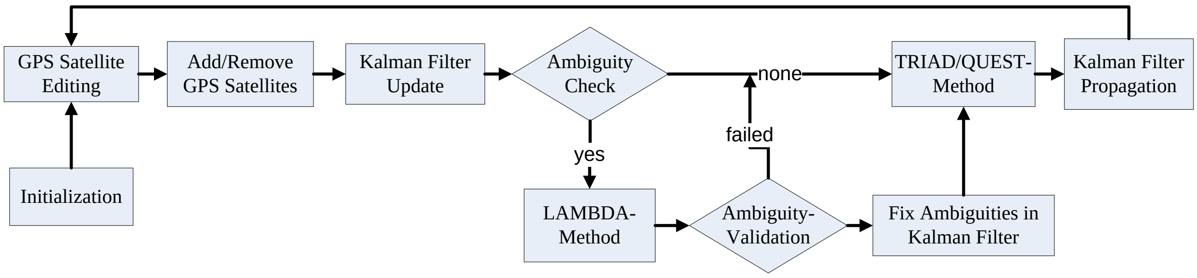

Figure 5: Flowchart of the GPS Attitude Determination Algorithm

3 Performance Evaluation

The verification and performance evaluation of the attitude determination algorithm has been accomplished with test data, collected during a field experiment. For this experiment, three GPS antennas have been mounted to an aluminum rack in a comparable arrangement to the antenna system on the Flying Laptop satellite. The two baseline vectors had a lengths of 0.4 m and 0.6 m. The aluminum rack has been placed on a turntable in order to simulate the rotation of the satellite. The turntable allows a rotation of the antenna array around the vertical axis with a constant rate. The antennas were connected to three Phoenix GPS receivers. The plots in Fig. 6 show the Euler angles yaw, pitch and roll over time for measurements taken over an interval of one hour with a rotation rate of 1°/s. The angles describe the rotation of the antenna array with respect to the local horizontal coordinate system. The upper plot for the yaw angle clearly shows the rotation rate of the platform. The plots for the roll and pitch angle in the lower plot represent the tilt of the platform with respect to the local horizon. Ideally, both angles should be zero at all times. Instead, both pitch and roll angle show variations with a standard deviation of about 1°. The maximum variations are in the order of 2°. The deviations of the roll and pitch angle are caused by errors in the horizontal components of the differential antenna positions. These positioning errors can be of different origin. The dominating error source are multipath errors on the carrier phase measurements, which are caused by multiple signal receptions at the antenna due to reflections of the electromagnetic wave at objects in the vicinity of the antenna array. During the field experiments, the carrier phase multipath errors of low elevation satellites reached magnitudes of up to 2.5 cm [2], which corresponds to the observed deviations of the Euler angles. In contrast to the measurement errors causes by the receiver noise, the assumption of a Gaussian distribution does not hold for multipath errors. Therefore these errors are more difficult to be filtered out using a Kalman Filter. Additionally, systematic errors like antenna phase center variations and mounting offsets introduce errors in the solution for the Euler angles. A rigor assessment of the magnitude of the different error sources cannot be provided for the experiment, since no reference attitude from a more accurate sensor is available.

It can be stated from the results of the experiment that the algorithm provides reasonable results for the attitude solution. The noise on the measurements can be filtered out effectively by the Kalman Filter, but systematic errors cause deviations of the attitude solution from the true solution. The single difference carrier phase ambiguities for new satellites could be resolved within several seconds using the LAMBDA-method.

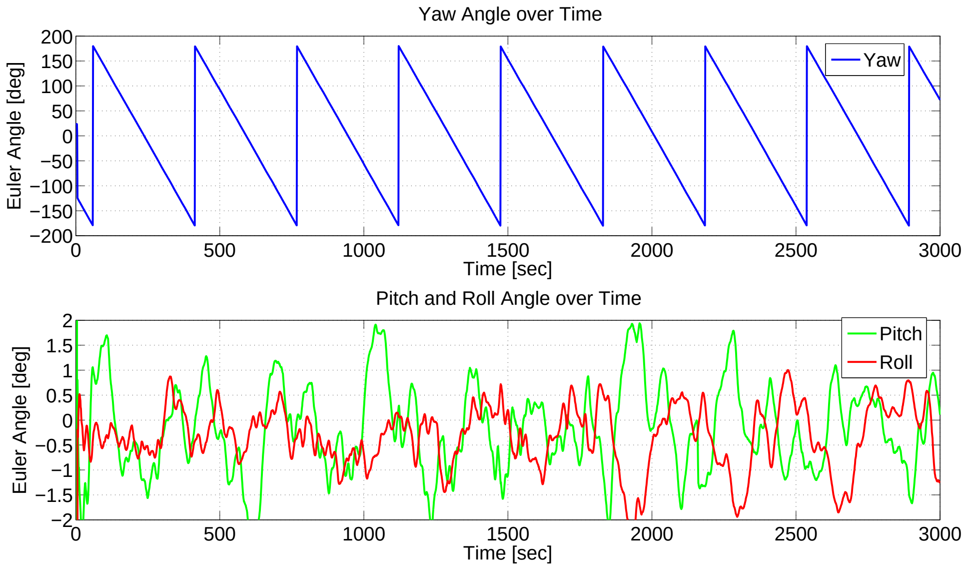

Figure 6: Yaw, pitch and roll angle with respect to the local coordinate frame for the dual baseline experiment with a rotation rate of 1°/s

In order to prove that the algorithm can also handle the frequent changes of the GPS satellite constellation typical for space-borne applications, numerical simulations have been executed. During these simulations, the C/A-code and carrier-phase measurements of the GPS system on the Flying Laptop satellite have been created using realistic attitude scenarios. In the first attitude scenario, the nadir-pointing-mode, the satellite orbits the earth with a constant angular velocity vector in nadir orientation. In this attitude mode, the assumed system model for the Kalman Filter comes close to the real world dynamics, thus the filter does not suffer performance problems. The second attitude mode is the target pointing mode. During this mode, the satellite is pointed to a target fixed on the earth's surface and thus continuously controlled by the attitude control system, which applies control torques to keep the satellite aligned with the reference frame during its pass over the ground target. Due to the applied torques, the angular velocity vector of the satellite changes and the assumption made for the system model in the Kalman filter is no longer true.

The plots in Fig. 7 show the errors in the Euler angles between the reference attitude solution and the estimated attitude from the Kalman filter for the target pointing mode. For the upper plot in the figure a process noise of $\sigma_{\dot{\omega}} = 0.5°/\text{sec}^2$ has been selected. It is obvious that the errors in the yaw, pitch and roll angles are close to zero over the complete simulation interval. For the middle plot the Kalman filter has been executed with the same settings as before, only the process noise of the system model has been reduced by one order of magnitude to $\sigma_{\dot{\omega}} = 0.05°/\text{sec}^2$. In this plot, the error for the pitch angle shows deviations of approximately 2° with respect to the nominal attitude. The yaw angle also shows deviations from the nominal attitude, which are much smaller and hardly recognizable in the plot. These deviations appear during the transitions between nadir pointing and target pointing mode. Reducing the process noise by another order of magnitude leads to the Euler angle errors as shown in the lower plot of Fig. 7. Obviously, the pitch error shows larger deviations compared to the nominal attitude with a maximum error of 10° during the transitions between the two attitude modes. Additionally, the yaw and roll angle deviate from the reference attitude in the same interval.

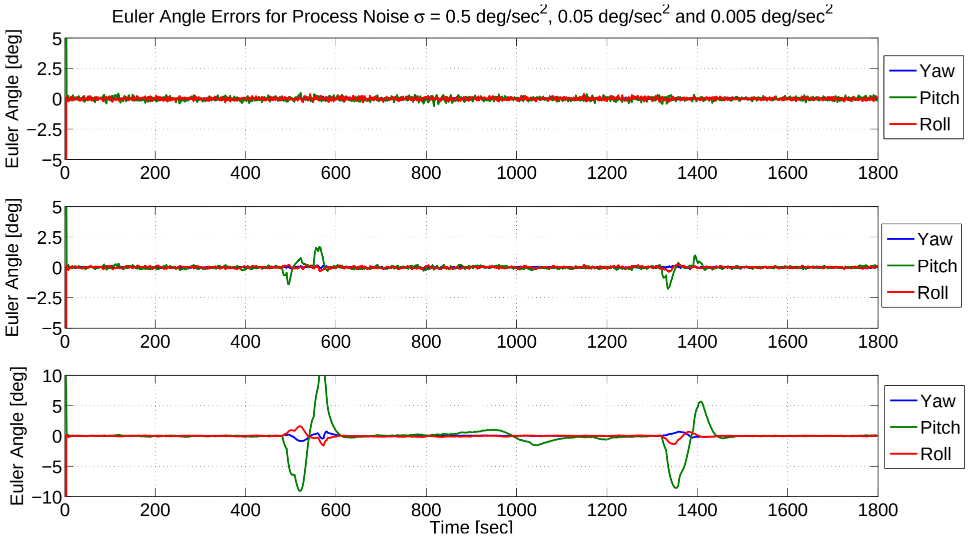

Figure 7: Euler Angle Errors for the target pointing mode from PEM simulations, the figure shows the errors with different setting for the process noise $\sigma_{\dot{\omega}}$ of $0.5°/\mathrm{s}^2$, $0.05°/\mathrm{s}^2$ and $0.005°/\mathrm{s}^2$ (from top to bottom)

It becomes obvious from the plot that choosing a comparably large amount of process noise avoids the deviation of the Euler angles from the true solution. This is a reasonable results, since the Kalman Filter assigns a higher weight to the measurements than to the propagated system model during the state update. Decreasing the process noise leads to deviations in the attitude solution since the filter relies more and more on the (incorrectly) propagated system model, which does not take the applied control torques into account. Thus parts of the information from the measurements are ignored by the filter and the corrections applied to the baseline vectors and the angular velocity vector are too small. According to the previous discussion the necessary process noise for a satisfying filter performance is determined by the target pointing mode, which is the worst case scenario for the filter. The standard deviations of the Euler angles errors for the process noise of $\sigma_{\dot{\omega}} = 0.5°/\mathrm{s}^2$ are $\sigma_\psi = 0.0445°$, $\sigma_\theta = 0.1385°$ and $\sigma_\phi = 0.0972°$. Thus an attitude determination with approximately $\sigma = 0.1°$ appears to be possible, which is one order of magnitude smaller than the accuracy reached for the ground based experiments. It must be noted though, that the numerical simulation does only contain the C/A-code and carrier phase measurement errors due to receiver noise. Multipath errors are difficult to model and are thus not included in the simulation. Fortunately, signal reflections in space can only result from the satellite body itself. Since the three GPS antenna are mounted the panel with the solar generators, no objects which could cause strong signal reflections are in the vicinity of the antennas. Though the multipath errors had a comparably large impact on the filter performace during the ground based experiment, their influence on the accuracy for the space-borne scenario can be expected to be much smaller.

4 Summary, Conclusions and Future Work

In this paper the GPS based attitude determination system of university small satellite Flying Laptop has been introduced. An algorithm based on a Kalman Filter is used to process the measurement data and produce an offline attitude solution which will be compared to the attitude information available from the satellite's star camera. The algorithm uses the LAMBDA-method to resolve the integer ambiguities of the double differences of the carrier phase measurements. These resolved double difference ambiguities are then used to fix the single difference ambiguities in the filter. Thus the algorithm provides a seamless transition from the ambiguity resolution to the attitude determination. The algorithm can be initialized without any a priori knowledge about the ambiguities or the initial attitude of the satellite. Ground based tests and numerical simulation are used to validate the algorithm and assess its accuracy. It has been found that the attitude can be determined with a standard deviation of about $1°$ for ground based tests under the influence of multipath and about $0.1°$ for the space-borne application without multipath errors. The accuracy of the algorithm in current form can be expected to be between $0.1°$ and $1°$ depending on the magnitude of the systematic measurement errors in space.

During target pointing mode, the changes in the angular velocity vector lead to a divergence between the dynamics of the real system and the modelled system in the Kalman filter. Though large deviations in the attitude solution can be avoided by increasing the process noise, the suppression of measurement noise is less effective in this case. To overcome this drawback, a more sophisticated system model must be used for the Kalman Filter which reflects the real system more closely. Additionally, measurements from a rate gyro system can be used in addition to the GPS measurements

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