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INTEGRATED ATTITUDE AND NAVIGATION SYSTEM FOR SMALL AIRPLANE

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Abstract

Navigation system is one of the most important aircraft systems. Accuracy and precision of position and attitude is extremely important for safe aircraft operations. The integrated INS/GNSS systems are commonly used as autonomous on-board devices for fulfilling this task. The INS sensors like accelerometers and gyroscopes are mainly affected by drift. The GNSS encounter stochastic disturbances with no tendency to grow in time but as each radio navigation system may be jammed or its signal can just be not available. These base properties of errors make these two systems well suited for integration. These were the main motivations for development of integrated navigation and attitude determination system, presented in this article. In the developed system, data is integrated from all available sensors, particularly INS, GPS, and air data computer. Navigation information from these sensors is combined using Kalman filtering algorithms to obtain robust solution, effective also in a case of failure/inaccessibility of GPS. Position calculated using the accelerations from INS is corrected by position from GPS and optionally by position calculated using the true airspeed (TAS) from ADC. Navigation system is modelled and programmed in MATLAB environment. The system was tested using the data from real experiments, proving efficiency of the method.

Keywords: *navigation, integration, data fusion, INS, GPS, ADC*

1. Introduction

The research presented in this work was performed as a part of NCBiR (National Centre of Research and Development) funded "Methods of Synthesis of Aircraft Control System Taking into Account High-Risk Situations" (MYSTERY) project, which main goal was to elaborate navigation algorithms taking into account incomplete measurement information.

The main objective of this part of the project was to design, model and implement robust integrated navigation and attitude determination system, which is able to operate in case of failure / inaccessibility of Global Positioning System (GPS) signal. The main requirements for the system were to:

- provide reliable navigation information during all flight operations,
- be robust for lack of the GPS signal,
- integrate available navigation data,
- be adaptable to the actual, specific hardware / software design,
- be computationally efficient.

Existing aircraft navigation systems usually integrate only Inertial Navigation System (INS) and GPS data with triple redundancy. The INS sensors like accelerometers and gyroscopes are mainly affected by drift. The GPS encounter stochastic disturbances with no tendency to grow in time but as each radio navigation system may be jammed or its signal can just be not available. These base properties of errors make these two systems well suited for integration. Such solution is robust – while no malfunction of the navigation system, it compensates errors obtained from INS and allows continuing precise navigation for some time if GPS signal is not available for some reasons. But because of the triple redundancy, it is not cost and mass effective, especially for the small airplane where payload is strongly limited and desirable.

The aim of the study is to provide a novel solution with prospective application in a long-term perspective. In the designed system, navigation data is obtained from three devices - INS, GPS and Air Data Computer (ADC). Using the tuned Kalman filtering methods, gathered navigation data from sensors is combined. Position determined by INS is corrected by position obtained from GPS, which is treated as a main, most reliable source of navigation data. In case of lack of GPS signal, INS is corrected by processed signal from ADC. Such data fusion seems to be robust in case of failure / inaccessibility of GPS signal and can ensure higher reliability with lower redundancy requirements.

2. System architecture

For the system architecture, it is assumed that navigation information from sensors like GPS, INS and ADC is provided. Firstly, raw information from INS (accelerations and angular velocities) is subjected to filtration to eliminate noises and bias from the data. Next, information from all available sensors is converted and processed to the desired output values in WGS-84 coordinate system. It is made using developed navigation algorithms established and designed for each of the sensors.

At final step, available data is fused and integrated, using error models, providing high precision navigation. First and final filtration / data fusion is made using Kalman filtering methods. System architecture is presented in Fig. 1. Such architecture is adaptable to the actual, specific aircraft hardware / software design.

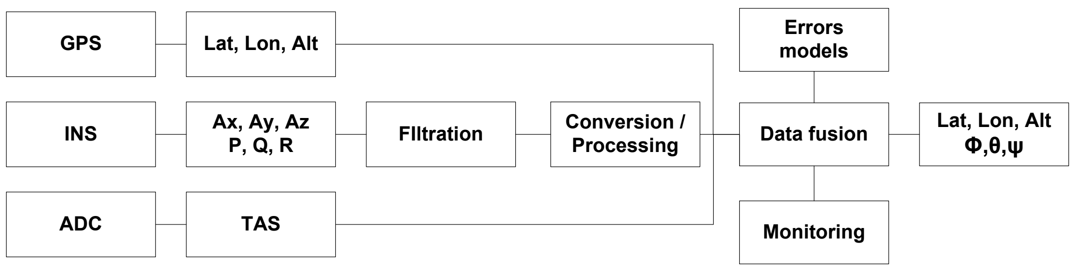

Fig. 1. System architecture

3. System model

In the designed navigation system, several sensors are used:

- 3-axis accelerometer,
- 3-axis gyroscope,
- piezoresistive dynamic pressure sensor,
- GPS module.

Accelerometers and gyroscopes are part of the Inertial Navigation System. Piezoresistive dynamic pressure sensor is a part of Air Data Computer. Information from static pressure sensors is not considered in this work.

For the sensors signal conversion, basic coordinate systems are used which are described below:

- inertial coordinate system $O_i x_i y_i z_i$, the origin O_i is located in the Earth's gravity centre, axis $O_i z_i$ coincides with Earth rotation axis and faces toward North Pole, axis $O_i x_i$ lies in the equatorial plane and has a direction and turn to the vernal equinox, axis $O_i y_i$ completes the system to dextrorotatory,
- Earth coordinate system $O_e x_e y_e z_e$, the origin O_e is located in the Earth's gravity centre, axis $O_e z_e$ coincides with Earth rotation axis and faces toward North Pole, axis $O_e x_e$ is formed by the plane of the equator and zero meridian and is faced toward zero meridian, axis $O_e y_e$ completes the system to dextrorotatory. System $O_e x_e y_e z_e$ rotates with respect to the system $O_i x_i y_i z_i$, around axis $O_i z_i$, with the constant Earth daily rotation angular velocity Ω,
- navigation coordinate system $O_n x_n y_n z_n$, associated with Earth, the origin O_n is located at any point on the Earth's surface, axis $O_n z_n$ has a direction and turn of the gravity acceleration vector in point O_n, plane $O_n x_n y_n$ is perpendicular to the Earth gravity acceleration vector in point O_n, axis $O_n x_n$ is directed and turned toward true north, axis $O_n y_n$ completes the system to dextrorotatory and is directed to the east,
- gravity coordinate system $O_g x_g y_g z_g$, associated with the moving object, origin O_g is placed in selected point (object gravity centre for example), system $O_g x_g y_g z_g$ is moved parallel with respect to system $O_n x_n y_n z_n$, turns of both coordinate systems are compatible,
- airplane coordinate system $O_b x_b y_b z_b$, connected with moving object, origin O_b is placed at the selected point of the object, axis $O_b x_b$ lies in the plane of symmetry of the object $O_b x_b z_b$ and is turned toward the front of the fuselage, axis $O_b z_b$ is placed in the plane of object symmetry, is perpendicular to axis $O_b x_b$ and turned "downward", axis $O_b y_b$ completes the system to dextrorotatory and is directed to the right,
- aerodynamic coordinate system $O_a x_a y_a z_a$, connected with the moving object, origin O_a coincides with the point O_g, axis $O_a x_a$ has a direction of vector of the object velocity with respect to the air stream and opposite turn, axis $O_a z_a$ lies in the plane of symmetry of the object, is perpendicular to axis $O_a x_a$ and turned "downward", axis $O_a y_a$ completes the system to dextrorotatory,
- geodetic coordinate system WGS-84 - origin and axes of the system $O_w x_w y_w z_w$ coincide with the system $O_e x_e y_e z_e$, however it is assumed that Earth is rotational ellipsoid. Position in this coordinate system is described in polar coordinates $\boldsymbol{r} = [\varphi \;\; \lambda \;\; h]$,
  where:
  φ – geodetic latitude defined as an angle between equator plane and the straight perpendicular in this point to the surface of ellipsoid,
  λ – geodetic longitude defined as an angle between meridian plane of point and zero meridian plane,
  h – altitude above ellipsoid, measured along the direction of local straight, which is perpendicular to the ellipsoid.

Navigation, gravity and airplane coordinate systems are shown in Fig. 2.

For the sensors signal processing, navigation algorithms for INS, ADC and GPS are used which are described below.

INS Accelerometers and gyroscopes with processing algorithm form Inertial Navigation System (INS). Final navigation information has to be presented in WGS-84 coordinate system. For this purpose, firstly, navigation values in $O_g x_g y_g z_g$ coordinate system are calculated.

Position and velocity. Acceleration of the point O_g relative to point O_e is calculated as:

$$\dot{\boldsymbol{v}}_e^g = \boldsymbol{C}_b^g \boldsymbol{f}^b - [2\boldsymbol{\omega}_{ie}^g + \boldsymbol{\omega}_{eg}^g] \times \boldsymbol{v}_e^g + \boldsymbol{g}_l^g \,, \tag{1}$$

where:

$\boldsymbol{v}_e^g = [v_e^{g,x} \quad v_e^{g,y} \quad v_e^{g,z}]^T$ – velocity vector of point O_g with respect to point O_e,

$\boldsymbol{C}_b^g$ – transformation matrix of linear accelerations vector $\boldsymbol{f}^b$ measured in system $O_b x_b y_b z_b$ to system $O_g x_g y_g z_g$,

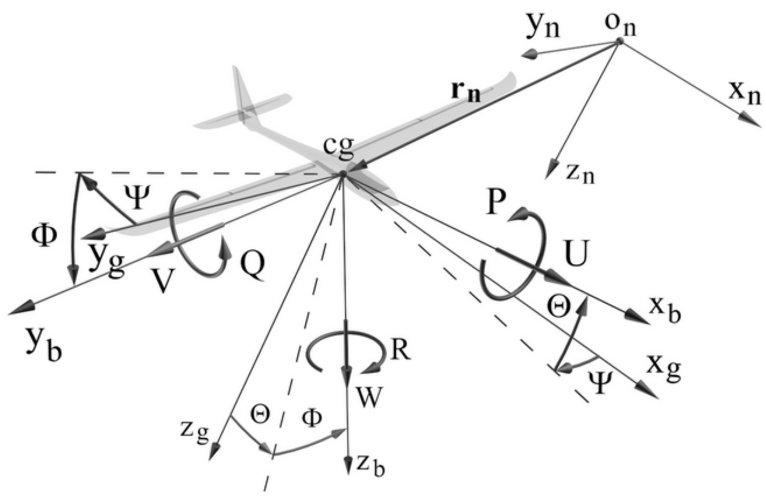

Fig. 2. Navigation, gravity and airplane coordinate system

$$\boldsymbol{C}_b^g = \begin{bmatrix} \cos\Theta\cos\Psi & \sin\Theta\sin\Phi\cos\Psi - \cos\Phi\sin\Psi & \cos\Phi\sin\Theta\cos\Psi + \sin\Phi\sin\Psi \\ \cos\Theta\sin\Psi & \sin\Theta\sin\Phi\sin\Psi + \cos\Phi\cos\Psi & \cos\Phi\sin\Theta\sin\Psi - \sin\Phi\cos\Psi \\ -\sin\Theta & \sin\Phi\cos\Theta & \cos\Phi\cos\Theta \end{bmatrix},$$

Ψ, Θ, Φ – attitude angles, respectively yaw, pitch and roll,
$\boldsymbol{f}^b = [\,f^{b,x} \quad f^{b,y} \quad f^{b,z}\,]^T$ – vector of measured linear accelerations,
$2\boldsymbol{\omega}_{ie}^g \times \boldsymbol{v}_e^g$ – Coriolis acceleration, resulting from the progressive movement of the point O_g with velocity over the surface of a rotating Earth $\boldsymbol{v}_e^g$,
$\boldsymbol{\omega}_{ie}^g = [\Omega\cos\varphi \quad 0 \quad -\Omega\sin\varphi]^T$ – angular velocity of the Earth in the system $O_g x_g y_g z_g$,
where:
Ω – value of the Earth angular velocity,
φ – latitude,
$\boldsymbol{\omega}_{eg}^g \times \boldsymbol{v}_e^g$ – object centripetal acceleration, resulting from movement of the object with velocity $\boldsymbol{v}_e^g$ over the Earth surface which is rotational ellipsoid. As a result, system $O_g x_g y_g z_g$ moves with the angular velocity $\boldsymbol{\omega}_{eg}^g$ with respect to system $O_e x_e y_e z_e$,
$\boldsymbol{\omega}_{eg}^g = [\dot{\lambda}\cos\varphi \quad -\dot{\varphi} \quad -\dot{\lambda}\sin\varphi]^T$ – vector of angular velocity of the system $O_g x_g y_g z_g$ with respect to system $O_e x_e y_e z_e$, expressed in system $O_g x_g y_g z_g$,
$\boldsymbol{g}_l^g$ – local gravity acceleration vector:

$$\boldsymbol{g}_l^g = \boldsymbol{g}^g - \boldsymbol{\omega}_{ie} \times [\boldsymbol{\omega}_{ie} \times \boldsymbol{R}], \tag{2}$$

$\boldsymbol{\omega}_{ie} \times [\boldsymbol{\omega}_{ie} \times \boldsymbol{R}]$ – centripetal acceleration resulting from the Earth's rotation:

$$\boldsymbol{\omega}_{ie} \times [\boldsymbol{\omega}_{ie} \times \boldsymbol{R}] = \frac{\Omega^2 (R_0 + h)}{2} \begin{bmatrix} \sin 2\varphi \\ 0 \\ 1 + \cos 2\varphi \end{bmatrix}, \tag{3}$$

$\boldsymbol{g}^g = [0 \quad 0 \quad g^g]^T$ – gravity acceleration, where g^g is the value of the gravity acceleration in point O_g, value of g^g is determined by International Gravity Formula:

$$g^g = g_e (1 + A\sin^2\varphi - B\sin^2 2\varphi)\left(1 - \frac{2h}{a}\right), \tag{4}$$

where:
$g_e = 9.780327$ [m/s^2] – value of the gravity acceleration on the equator,
$a = 6378137$ [m] – length of semi-major axis of the Earth ellipsoid,
$A = 0.0053024$ – constant,
$B = 0.0000058$ – constant.

Using the mentioned above relationship, velocity $\boldsymbol{v}_e^g$ of point O_g with respect to point O_e is determined. Navigation information has to be determined in system WGS-84, so transformation of calculated values from system $O_g x_g y_g z_g$ to system WGS-84 has to be done:

$$\dot{\varphi} = \frac{v_e^{g,x}}{R_N + h}, \tag{5}$$

$$\dot{\lambda} = \frac{v_e^{g,y}}{(R_E + h)\cos\varphi}, \tag{6}$$

$$\dot{h} = -v_e^{g,z}, \tag{7}$$

where:
meridian beam cross:

$$R_N = \frac{a(1-e^2)}{(1-e^2\sin^2\varphi)^{3/2}}, \tag{8}$$

first vertical beam cross:

$$R_E = \frac{a}{(1-e^2\sin^2\varphi)^{1/2}}, \tag{9}$$

first eccentricity:

$$e = \sqrt{1-\frac{b^2}{a^2}}, \tag{10}$$

where:
$a = 6378137$ [m] – semi-major axis length,
$b = a(1-f) = 6356752.3142$ [m] – semi-minor axis length,
$f = \dfrac{1}{298.257223563}$ – flattening.

Position in system WGS-84 $\boldsymbol{r}_{INS}$, calculated using INS, can be determined by solving the equation:

$$\dot{\boldsymbol{r}}_{INS} = \begin{bmatrix} \dot{\varphi} \\ \dot{\lambda} \\ \dot{h} \end{bmatrix} = D_g^w \boldsymbol{v}_e^g = \begin{bmatrix} \frac{1}{R_N + h} & 0 & 0 \\ 0 & \frac{1}{(R_E + h)\cos\varphi} & 0 \\ 0 & 0 & -1 \end{bmatrix} \cdot \begin{bmatrix} v_e^{g,x} \\ v_e^{g,y} \\ v_e^{g,z} \end{bmatrix}. \tag{11}$$

To determine the position, it is necessary to calculate matrix $\boldsymbol{C}_b^g$, which can be described by equation:

$$\dot{\boldsymbol{C}}_b^g = \boldsymbol{C}_b^g [\boldsymbol{\Omega}_{gb}^b]_\times, \tag{12}$$

where:

$[\mathbf{\Omega}_{gb}^{b}]_{\times}=\begin{bmatrix}0 & -\omega_{gb}^{b,z} & \omega_{gb}^{b,y}\\ \omega_{gb}^{b,z} & 0 & -\omega_{gb}^{b,x}\\ -\omega_{gb}^{b,y} & \omega_{gb}^{b,x} & 0\end{bmatrix}$ – skew-symmetric matrix formed from the vector $\boldsymbol{\omega}_{gb}^{b}=[\omega_{gb}^{b,x}\quad \omega_{gb}^{b,y}\quad \omega_{gb}^{b,z}]^{T}$ which describes angular velocity of the system $O_b\,x_b\,y_b\,z_b$ with respect to the system $O_g\,x_g\,y_g\,z_g$.

Vector $\boldsymbol{\omega}_{gb}^{b}$ can be calculated using the equation:

$$\boldsymbol{\omega}_{gb}^{b}=\boldsymbol{\omega}_{ib}^{b}-\boldsymbol{C}_{g}^{b}\,(\boldsymbol{\omega}_{ie}^{g}+\boldsymbol{\omega}_{eg}^{g}), \tag{13}$$

where $\boldsymbol{C}_{g}^{b}=(\boldsymbol{C}_{b}^{g})^{-1}=(\boldsymbol{C}_{b}^{g})^{T}$.

Attitude angles. Vector of angular velocity $\boldsymbol{\omega}_{gb}^{b}$ is also necessary to determine object attitude:

$$\begin{bmatrix}\dot{\Phi}\\ \dot{\Theta}\\ \dot{\Psi}\end{bmatrix}=\begin{bmatrix}1 & \sin\Phi\,\mathrm{tg}\Theta & \cos\Phi\,\mathrm{tg}\Theta\\ 0 & \cos\Phi & -\sin\Phi\\ 0 & \frac{\sin\Phi}{\cos\Theta} & \frac{\cos\Phi}{\cos\Theta}\end{bmatrix}\boldsymbol{\omega}_{gb}^{b}, \tag{14}$$

where:

Φ, Θ, Ψ – attitude angles of the aircraft.

ADC. True airspeed vector $\boldsymbol{v}_{TAS}^{a}=[v_{TAS}\quad 0\quad 0]^{T}$ is read and is based in aerodynamic coordinate system $O_a\,x_a\,y_a\,z_a$. Transformation from aerodynamic coordinate system to airplane coordinate system has a form:

$$\boldsymbol{v}_{TAS}^{b}=\boldsymbol{C}_{a}^{b}\,\boldsymbol{v}_{TAS}^{a}, \tag{15}$$

where:

$\boldsymbol{C}_{a}^{b}=\begin{bmatrix}\cos\alpha\cos\beta & -\cos\alpha\sin\beta & -\sin\alpha\\ \sin\beta & \cos\beta & 0\\ \sin\alpha\cos\beta & -\sin\alpha\sin\beta & \cos\alpha\end{bmatrix}$ – transformation matrix from aerodynamic to airplane coordinate system,

$\alpha=\mathrm{tg}^{-1}\left(\dfrac{v_{IAS}^{b,z}}{v_{IAS}^{b,x}}\right)$ – angle of attack (AoA),

$\beta=\sin^{-1}\left(\dfrac{v_{IAS}^{b,y}}{|v_{IAS}^{b}|}\right)$ – angle of sideslip (AoS).

Transformation from airplane coordinate system to gravity coordinate system has a form:

$$\boldsymbol{v}_{TAS}^{g}=\boldsymbol{C}_{b}^{g}\,\boldsymbol{v}_{TAS}^{b}, \tag{16}$$

where:

$\boldsymbol{C}_{b}^{g}$ – transformation matrix of linear accelerations vector $\boldsymbol{f}^{b}$ measured in system $O_b\,x_b\,y_b\,z_b$ to system $O_g\,x_g\,y_g\,z_g$,

$$\boldsymbol{C}_{b}^{g}=\begin{bmatrix}\cos\Theta\cos\Psi & \sin\Theta\sin\Phi\cos\Psi-\cos\Phi\sin\Psi & \cos\Phi\sin\Theta\cos\Psi+\sin\Phi\sin\Psi\\ \cos\Theta\sin\Psi & \sin\Theta\sin\Phi\sin\Psi+\cos\Phi\cos\Psi & \cos\Phi\sin\Theta\sin\Psi-\sin\Phi\cos\Psi\\ -\sin\Theta & \sin\Phi\cos\Theta & \cos\Phi\cos\Theta\end{bmatrix}.$$

Position in system WGS-84 $\boldsymbol{r}_{TAS}$, calculated using ADC, can be determined by solving the equation:

$$\dot{\boldsymbol{r}}_{TAS_k} = \begin{bmatrix} \dot{\varphi}_{TAS} \\ \dot{\lambda}_{TAS} \\ \dot{h}_{TAS} \end{bmatrix} = D_g^w \boldsymbol{v}_{TAS}^g = \begin{bmatrix} \frac{1}{R_N + h} & 0 & 0 \\ 0 & \frac{1}{(R_E + h)\cos\varphi} & 0 \\ 0 & 0 & -1 \end{bmatrix} \cdot \begin{bmatrix} v_{TAS}^{g,x} \\ v_{TAS}^{g,y} \\ v_{TAS}^{g,z} \end{bmatrix}. \tag{17}$$

Information from static pressure sensors is not considered in this work.

GPS. From GPS module position vector $\boldsymbol{r}_{GPS} = [\varphi_{GPS} \quad \lambda_{GPS} \quad h_{GPS}]^T$ in system WGS-84 is obtained.

4. Filtration method

To improve the accuracy of position and attitude estimation, accelerations and angular velocities are at first filtered to remove noises and bias. Next, using the filtered and processed by navigation algorithms data, information fusion is done. It is assumed that position determined by GPS is most reliable and it is used to correct the position determined by INS. Optionally, in case of GPS malfunction or signal inaccessibility, information from ADC is used to correct INS. Filtration algorithm is shown in Fig. 3.

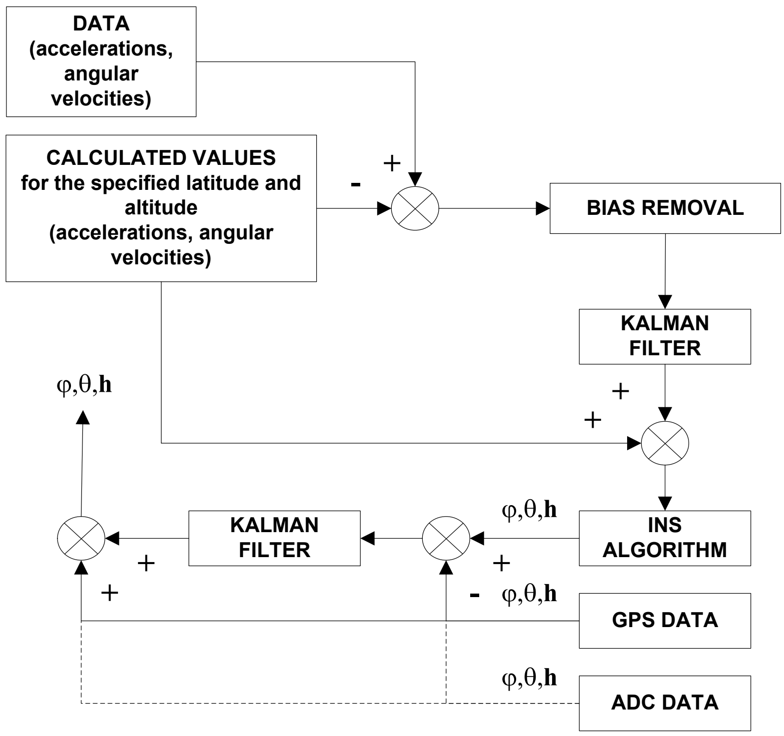

Fig. 3. Filtration algorithm

For the filtration and data fusion purposes, Kalman filter is used. It is widely used algorithm, first described in 1950s. Since then it was developed and extended. It is widely used in aviation, especially for guidance, navigation and control. Algorithm includes influence of statistical noise. If well designed it gives precise estimates of unknown variables. Because of its simplicity, it is convenient for real time computations.

For the discrete process shown in the form:

$$\boldsymbol{x}_{k+1} = \Phi\, \boldsymbol{x}_k + \boldsymbol{w}_k\,, \tag{18}$$

$$\boldsymbol{z}_k = \boldsymbol{H}\boldsymbol{x}_k + \boldsymbol{v}_k\,, \tag{19}$$

discrete Kalman filter algorithm is as follows:

- initial estimate $\hat{\boldsymbol{x}}_0^-$, $\boldsymbol{P}_0^-$,
- Kalman gain:

$$\boldsymbol{K}_k = \boldsymbol{P}_k^- \boldsymbol{H}^T (\boldsymbol{H}\boldsymbol{P}_k^- \boldsymbol{H}^T + \boldsymbol{R})^{-1}\,, \tag{20}$$

- updated state estimate:

$$\hat{\boldsymbol{x}}_k = \hat{\boldsymbol{x}}_k^- + \boldsymbol{K}_k (\boldsymbol{z}_k - \boldsymbol{H}\hat{\boldsymbol{x}}_k^-)\,, \tag{21}$$

- updated error covariance:

$$\boldsymbol{P}_k = (\boldsymbol{I} - \boldsymbol{K}_k \boldsymbol{H})\, \boldsymbol{P}_k^-\,, \tag{22}$$

- predicted state estimate:

$$\hat{\boldsymbol{x}}_{k+1}^- = \Phi\, \hat{\boldsymbol{x}}_k\,, \tag{23}$$

- predicted error covariance:

$$\boldsymbol{P}_{k+1}^- = \Phi\, \boldsymbol{P}_k \Phi^T + \boldsymbol{Q}\,, \tag{24}$$

where:

$\boldsymbol{x}$ – state vector,
$\boldsymbol{v}$ – disturbance vector of observation vector,
$\boldsymbol{w}$ – disturbance vector of state vector,
$\boldsymbol{z}$ – observation vector,
Φ – state matrix,
$\boldsymbol{H}$ – observation matrix,
k – index of point in time t_k,
$\boldsymbol{K}$ – Kalman gain matrix,
$\boldsymbol{P}$ – error covariance matrix,
$\boldsymbol{Q}$ – process error covariance matrix,
$\boldsymbol{R}$ – observator error covariance matrix.

5. Simulation model

The navigation algorithms and filtration / data fusion procedures described in the previous chapter form a base structure for precise estimations of aircraft position and attitude. The modular simulation software was developed which allows adapting navigation system dedicated to a particular aircraft. The program is prepared in MATLAB/Simulink R2014b environment. It was used for presenting applicability of fused navigation system; and to illustrate how the proposed methodology may be implemented in practice. The subsequent steps of calculations are:

- loading input data, i.e. basic navigation information (accelerations, angular velocities, true airpseed, GPS position),
- first filtration of accelerations and angular velocities using Kalman filtering,
- processing the data to the desired output values using navigation algorithms,
- fusion of navigation data from GPS, INS and ADC using Kalman filtering,
- presenting calculated data.

The software is composed into a modular computation environment to create the navigation system for a particular structure of aircraft.

6. Sample results

For the calculations of position, real measurement data was used from three devices: INS and GPS. Data was logged statically at the airport with a time period of 700 [s] and with frequency of

50 [Hz]. ADC data is not included at this step of work (lack of airspeed because of static measurement - position calculated by ADC is constant at this case).

At first, accelerations were filtered to remove bias and noises (Fig. 4). The value of process error covariance was established approximately, based on knowledge of the process. Value of the observator error covariance was calculated as the measurement variance. For the accelerations bias and noises removal, in each step difference between measured and calculated for actual latitude and altitude accelerations was calculated and given to Kalman filter algorithm. Position calculated using the INS algorithm, based on filtered accelerations is shown in Fig. 5.

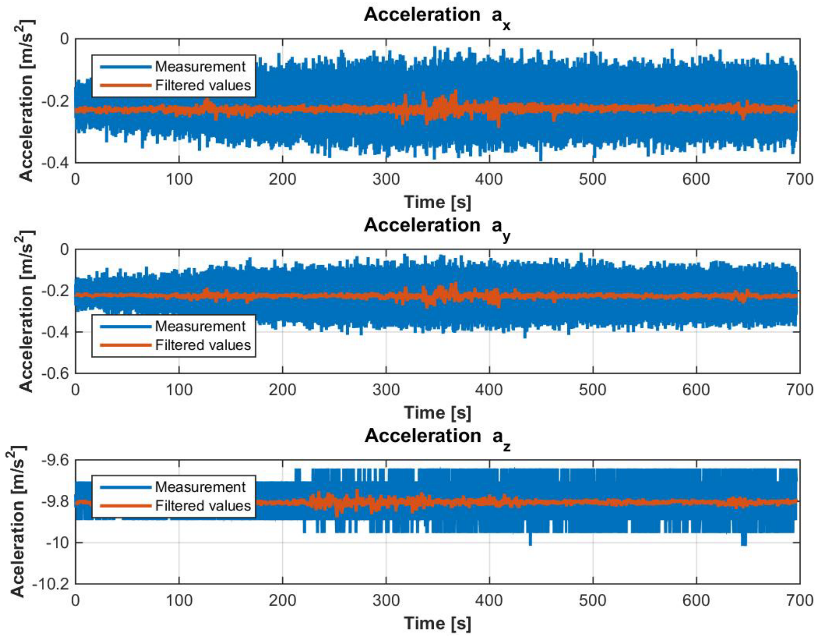

Fig. 4. Measured and filtered accelerations

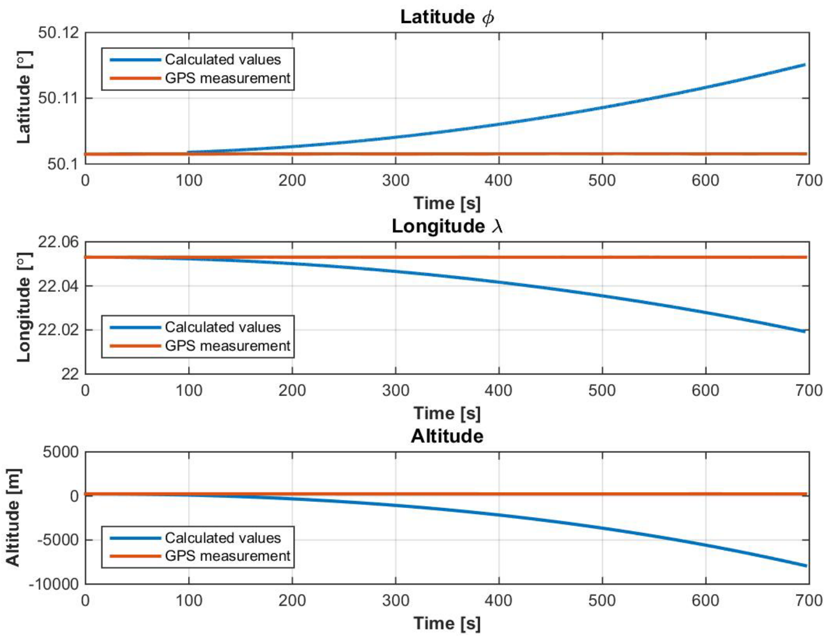

Fig. 5. Calculated position based on filtered accelerations

Next, position from INS and GPS was fused using the Kalman filter. It was assumed that INS navigation algorithm work independently – it does not use the position obtained from GPS as the input in each step of calculations. Values of filtered accelerations were used. The value of process error covariance was established approximately, based on knowledge of the process. Value of the observator error covariance was calculated as the measurement variance. In each step of calculations, difference between INS and GPS position was given to Kalman filter algorithm. Fused position data is shown in Fig. 6.

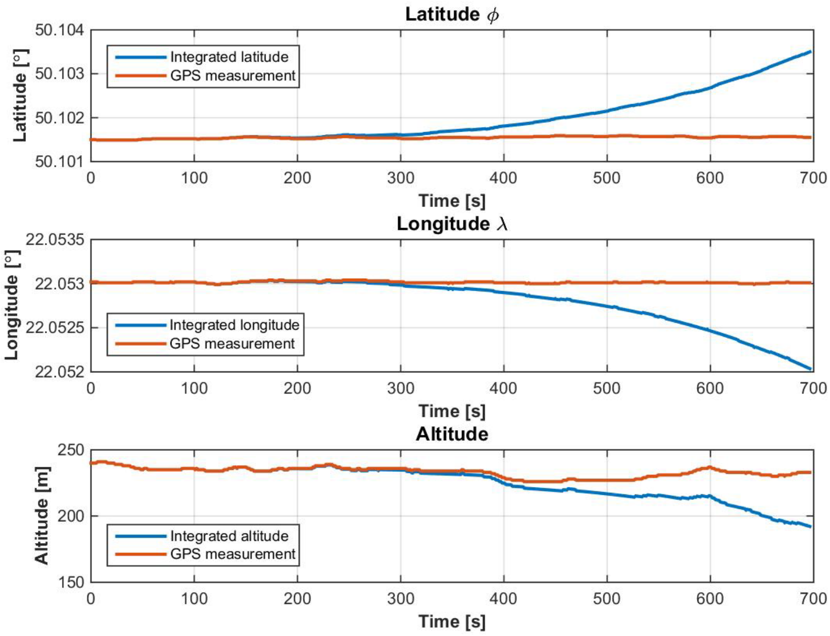

Fig. 6. Fused INS / GPS position

7. Sample results

The proposed navigation system is based on data fusion of position calculated by INS, GPS and ADC algorithms. Necessary navigation algorithms are presented and deeply described.

The proposed navigation system is adaptable and scalable to various configurations of aircraft. Modular, computational efficient simulation software was developed.

Sample results based on static measurements were presented. In this case, filtration of accelerations bias and noises decreases position estimation errors. Data fusion of position obtained from INS, GPS and optionally ADC efficiently increases position accuracy.

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