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AGILE QUATERNION BASED ATTITUDE DETERMINATION AND CONTROL TO SUPPORT NEAR-SPACE VEHICLE OPERATIONS

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Near-space is becoming more and more important both for space vehicle experimentation and for Earth surveillance applications. Flight conditions in near-space region emulate closely those of an extra-atmospheric space vehicle especially for its re-entry flight phase. European Space Agency is using near-space operations to perform critical test flights intended for validation of attitude control of space vehicles such as Space Rider. Here the issue of attitude control of a vehicle for operation in near-space conditions is addressed. A 6 DOF model has been developed to provide the framework for the quaternion based attitude control. The model was programmed on a dedicated autopilot installed onboard a dedicated and unique flying wing designed to be operated in near-space conditions. Autopilot uses a microcontroller, MEMS gyroscopes and accelerometers to provide a stable real-time attitude and trajectory solution. The attitude determination and control is performed using quaternions and their time derivatives instead of the usual Euler angles, allowing a stable attitude solution without the gimbal lock problems of a typical Euler formulation. A high altitude test flight has been performed from Cape Midia firing range which proved that the attitude determination and control system successfully allowed accurate control both of the roll and pitch angles.

Keywords: attitude determination, space vehicle, near-space operations.

1. Introduction

Near-space region is defined to be the region contained from 20 km to 100 km [1]. Near-space represents the region which is typically too low for satellites to achieve a stable orbit, due to atmosphere drag, and too high for a typical airplane

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to operate. Near-space access is usually enabled by high altitude balloons and sounding rockets [2-3].

Due to the unique characteristics, the near-space region provides an ideal environment for test flights of scale down space vehicles representative of larger space vehicles during the re-entry phase. A GNC (guidance, navigation and control) unit for such space vehicles is a difficult to be developed especially due to lack of ground infrastructure that could replicate conditions of flight at high altitude [4]. Hence, the near-space region with its low atmospheric density enables the development of such a GNC for a space vehicle. In near space region the space vehicle is expected to have a similar attitude dynamic as it would have in orbit operations.

European Space Agency (ESA) performs several test flight operations in order to validate the GNC of space vehicles (e.g.: Space Rider [5]) through test flights performed on scale down vehicles in near-space conditions (25-40 km altitude). The vehicles are typically brought to altitude using either weather balloons or sounding rockets [2].

In support of such tests, our research team has developed a flying wing vehicle with a high wing loading in order to achieve attitude dynamics in near-space conditions typical to the ones achieved by a lifting body space vehicle [3, 6-7].

The flying wing is built using composite materials and has electronic systems on-board that can provide tracking and remote control capabilities for ranges of up to 700 km [8]. A typical tracking of the trajectory using these systems is shown in Fig. 1.

The developed flying wing was used as a test flight laboratory for near space operations and an in-house developed autopilot is installed on-board. The autopilot has a microcontroller that allows fast run of the on-board algorithms and also a tri-axial accelerometer and tri-axial gyroscopes which provides the signals associated with translation accelerations and angular velocities.

The flying wing is attached to a weather balloon filled with helium and allowed to take-off freely as shown in Fig. 2. It typically ascends with 4-6 m/s and reaches altitudes of 25-35 km. Upon reaching the desired altitude (point P in Fig. 1) the flying wing is separated from the balloon using a customized pyrotechnic knife [8].

In Fig. 1 the ascent trajectory is AMP while the descent trajectory is ANP with P being the point where the flying wing separates from the balloon. The altitude of point P was 27 km.

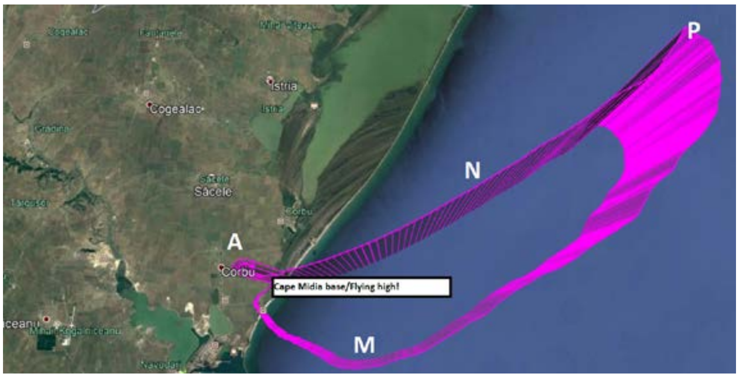

Fig. 1 Test flight trajectory.

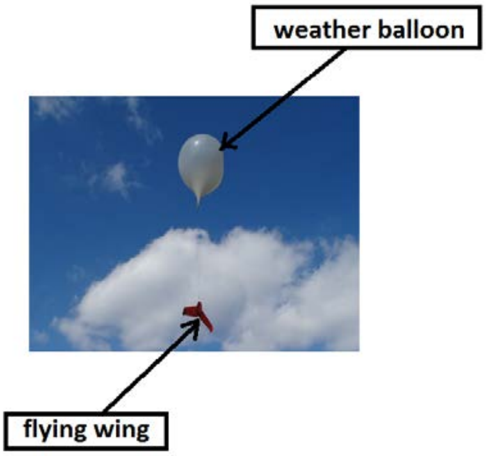

Fig. 2 Flying wing attached to weather balloon.

2. The 6 DOF model. Quaternion based attitude determination and control

In order to provide an agile quaternion based attitude control for near-space operations a 6 DOF model has been developed. The model is applicable for any space vehicle. The model contains a set of differential equations that describe both the translational movements of the vehicle's center of mass and the rotation of the vehicle around its center of mass. For writing the equations of motion two reference systems are used – the Earth's system of reference and the spacecraft's system of reference. They are shown in Fig. 3.

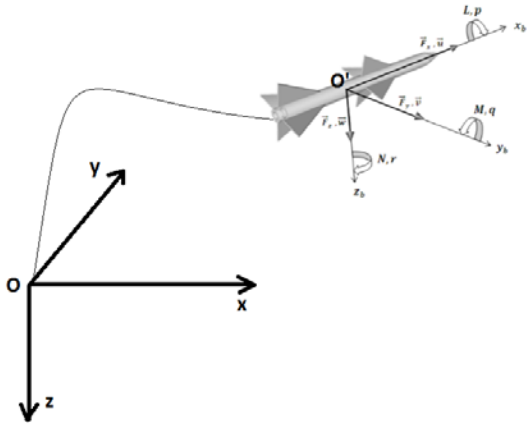

Fig. 3 The system of reference.

The basis of the 6 DOF model is represented by the following vector equations associated with the translation and rotations dynamics [9-11]:

$$m\frac{d\vec{V}}{dt} \equiv m\left(\frac{\partial \vec{V}}{\partial t} + \vec{\Omega}\times\vec{V}\right) = \sum_i \vec{F}_i + \vec{T}$$
$$\frac{d\vec{K}}{dt} \equiv \frac{\partial \vec{K}}{\partial t} + \vec{\Omega}\times\vec{K} = \sum_i \vec{H}_i + \vec{H}_T \qquad (1)$$

In equations (1), Ω represents the angular velocity of the flying wing. In order to program the equations on a computer they were rewritten in a matrix. Hence, the translation equations were obtained [9, 12-18]:

$$\begin{bmatrix} u \\ v \\ w \end{bmatrix} = \frac{1}{m}\left\{ F_0 \begin{bmatrix} C_x^A \\ C_y^A \\ C_z^A \end{bmatrix} + T_0 \begin{bmatrix} C_x^T \\ C_y^T \\ C_z^T \end{bmatrix} \right\} + A_i \begin{bmatrix} 0 \\ 0 \\ -g \end{bmatrix} + \begin{bmatrix} rv - qw \\ pw - ru \\ qu - pv \end{bmatrix} \qquad (2)$$

respectively the attitude equations:

$$\begin{bmatrix} p \\ q \\ r \end{bmatrix} = I^{-1}\left\{ H_0 \begin{bmatrix} C_l^A \\ C_m^A \\ C_n^A \end{bmatrix} + H_{T0} \begin{bmatrix} C_l^T \\ C_m^T \\ C_n^T \end{bmatrix} \right\} - I^{-1} \begin{bmatrix} (B-C)qr \\ (C-A)rp \\ (A-B)pq \end{bmatrix} \qquad (3)$$

The rotation between the body reference frame and the Earth reference frame may be expressed by quaternion formulation. The advantage of quaternion formulation is that it provides lock free formulation when compared to Euler formulation.

The direct matrix of rotation A_i is [10,11]:

$$A_i = \begin{bmatrix} q_4^2 + q_1^2 - q_2^2 - q_3^2 & -2\left(q_1 q_2 + q_3 q_4\right) & -2\left(q_3 q_1 - q_2 q_4\right) \\ 2\left(q_1 q_2 - q_3 q_4\right) & -q_4^2 - q_2^2 + q_3^2 + q_1^2 & -2\left(q_2 q_3 + q_4 q_1\right) \\ 2\left(q_1 q_3 + q_2 q_4\right) & -2\left(q_2 q_3 - q_4 q_1\right) & -q_4^2 - q_3^2 + q_1^2 + q_2^2 \end{bmatrix} \quad (4)$$

The inverse matrix of rotation is B_i, which is obtained by transposing the direct matrix of rotation A_i, and is given by:

$$B_i = \begin{bmatrix} q_4^2 + q_1^2 - q_2^2 - q_3^2 & 2\left(q_1 q_2 - q_3 q_4\right) & 2\left(q_3 q_1 + q_2 q_4\right) \\ -2\left(q_1 q_2 + q_3 q_4\right) & -q_4^2 - q_2^2 + q_3^2 + q_1^2 & -2\left(q_2 q_3 - q_4 q_1\right) \\ -2\left(q_1 q_3 - q_2 q_4\right) & -2\left(q_2 q_3 + q_4 q_1\right) & -q_4^2 - q_3^2 + q_1^2 + q_2^2 \end{bmatrix} \quad (5)$$

The connection between the components of the velocity vector in the inertial trihedral and in the mobile trihedral can be expressed using the above relations:

$$\left[\dot{x}_0 \quad \dot{y}_0 \quad \dot{z}_0\right]^T = B_i \left[u \quad v \quad w\right]^T \quad (6)$$

Then the relations between the components of the quaternions and the attitude angles can be found:

$$\begin{aligned} q_1 &= -\cos\frac{\varphi}{2}\cdot\sin\frac{\theta}{2}\cdot\sin\frac{\psi}{2} + \sin\frac{\varphi}{2}\cdot\cos\frac{\theta}{2}\cdot\cos\frac{\psi}{2} \\ q_2 &= \cos\frac{\varphi}{2}\cdot\sin\frac{\theta}{2}\cdot\cos\frac{\psi}{2} + \sin\frac{\varphi}{2}\cdot\cos\frac{\theta}{2}\cdot\sin\frac{\psi}{2} \\ q_3 &= \cos\frac{\varphi}{2}\cdot\cos\frac{\theta}{2}\cdot\sin\frac{\psi}{2} - \sin\frac{\varphi}{2}\cdot\sin\frac{\theta}{2}\cdot\cos\frac{\psi}{2} \\ q_4 &= \cos\frac{\varphi}{2}\cdot\cos\frac{\theta}{2}\cdot\cos\frac{\psi}{2} + \sin\frac{\varphi}{2}\cdot\sin\frac{\theta}{2}\cdot\sin\frac{\psi}{2} \end{aligned} \quad (7)$$

Next equation connects the gyroscope signals (p,q,r) to the time derivatives of the quaternions:

$$\left[p \quad q \quad r\right]^T = U_q \left[\dot{q}_1 \quad \dot{q}_2 \quad \dot{q}_3\right]^T \quad (8)$$

where

$$U_q = 2\begin{bmatrix} q_4 + \frac{q_1^2}{q_4} & q_3 + \frac{q_2 q_1}{q_4} & -q_2 + \frac{q_3 q_1}{q_4} \\ -q_3 + \frac{q_1 q_2}{q_4} & q_4 + \frac{q_2^2}{q_4} & q_1 + \frac{q_3 q_2}{q_4} \\ q_2 + \frac{q_1 q_3}{q_4} & -q_1 + \frac{q_2 q_3}{q_4} & q_4 + \frac{q_3^2}{q_4} \end{bmatrix} \tag{9}$$

The attitude dynamics formulated in quaternion space is represented by the following matrix [10, 11, 19-21]:

$$\begin{bmatrix} \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \\ \dot{q}_4 \end{bmatrix} = \frac{1}{2}\begin{bmatrix} 0 & r & -q & p \\ -r & 0 & p & q \\ q & -p & 0 & r \\ -p & -q & -r & 0 \end{bmatrix}\begin{bmatrix} q_1 \\ q_2 \\ q_3 \\ q_4 \end{bmatrix} = \frac{1}{2}\begin{bmatrix} q_4 & -q_3 & q_2 \\ q_3 & q_4 & -q_1 \\ -q_2 & q_1 & q_4 \\ -q_1 & -q_2 & -q_3 \end{bmatrix}\begin{bmatrix} p \\ q \\ r \end{bmatrix} \tag{10}$$

In order to achieve control a target value is set for a given axis. For example, in our test flight the pitch and roll axis target values were set to 0 degrees which represents stable and level flight. Hence, the objective of the controller is to achieve stable and level flight after the flying wing is separated from the weather balloon.

The control signals are generated using a PID loop with the control coefficients being set to constant values for the first 20 seconds and then to variable values for the remainder of the flight. For the constant control coefficients the P, I and D coefficients are set according to previous flight data. For the variable control coefficients a proprietary and classified algorithm is used. The basis of the algorithm is constituted by a Jacobian function of the form [22]:

$$J = \sum \left[t_k - k(\beta) \right] \tag{11}$$

The optimal value of the control coefficients is reached when the Jacobian reaches a very small value, in practice considered as 10^{-3} to 10^{-4}.

The variable values take into account the current controller performance, determined by the error between the current pitch/roll value and the target value and through a proprietary algorithm determines the coefficients that minimize the Jacobian shown in Eq. (11).

Eqs. (2), (3), (6)-(11) have been implemented in the autopilot microcontroller using C and assembly language. The integration technique is Runge-Kutta 4th order because it provides very good stability with a reasonable integration step

[23-26]. By running the entire loop containing the above mentioned equations and using inputs from the accelerometers and gyroscopes the autopilot is able to accurately determine the attitude and then correspondingly send commands on control channels in order to control the attitude.

Upon separation from the balloon the target of the attitude control is to obtain 0 degrees pitch and roll angles. In other words, the target of the attitude control is to provide a stabilized flight in near space conditions. During the ascent of the flying wing under the balloon the attitude is not controlled: the wing is free to roll and pitch according to the encountered winds. Hence, at the separation point the flying wing has an arbitrary attitude while the air density is very low hence the controls are not very effective. The attitude determination and control have a difficult task of agile determining the attitude and immediately control it in order to achieve level flight.

3. Results

The test flight took place at Cape Midia firing range which is a military firing range used by Romanian anti-aircraft forces for training purposes. Cape Midia is located on the sea-shore of Black Sea.

The flying wing was attached to a weather balloon filled with helium and allowed to ascent to 27 km where it was released by a radio command from the ground.

In Fig. 4 we have represented the altitude profile of the flight with the flying wing release point from the balloon being denoted with W.

Upon release from the balloon the attitude determination and control system correctly determined the attitude and issued the control commands in order to stabilize the flight in pitch and roll axis.

The pitch and roll data was recorded throughout the flight and the entire flight data is shown in Figs. 5 and 6.

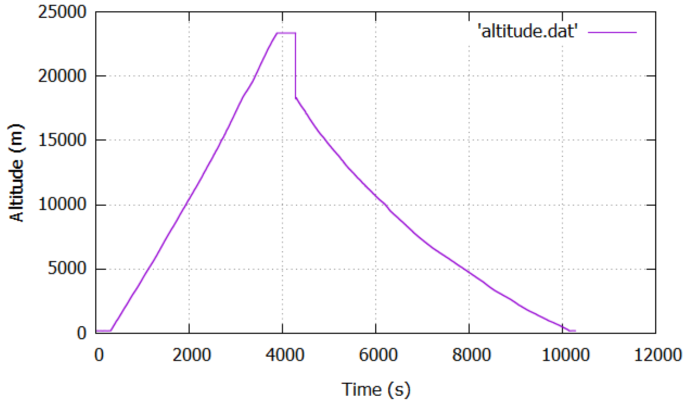

Fig. 4 Altitude variation with time.

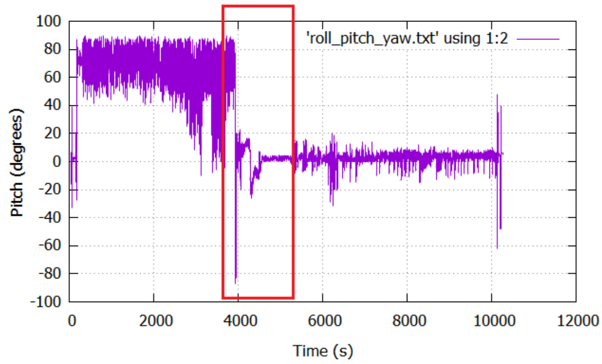

Fig. 5   Pitch as a function of time.

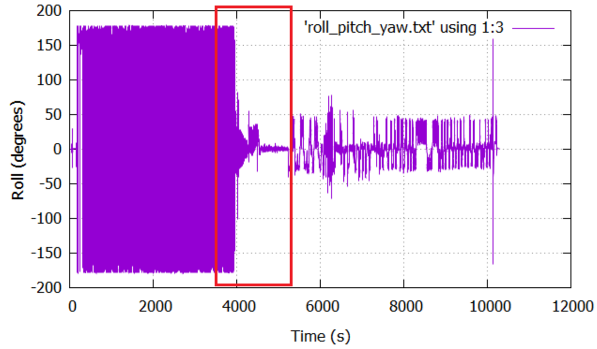

Fig. 6   Roll as a function of time.

The dynamics of the attitude in near space conditions is outlined in Figs. 5 and 6 by the red regions. That is the region where the attitude determination and control system behaves similarly to what would behave on a space vehicle.

Figs. 7 and 8 show a zoom in of the red regions outlined in Figs. 5 and 6. As it can be noticed the pitch was nearly 90 degrees (nose up) at the separation point while the roll had an arbitrary value due to torqueing of the flying wing under wind's influence.

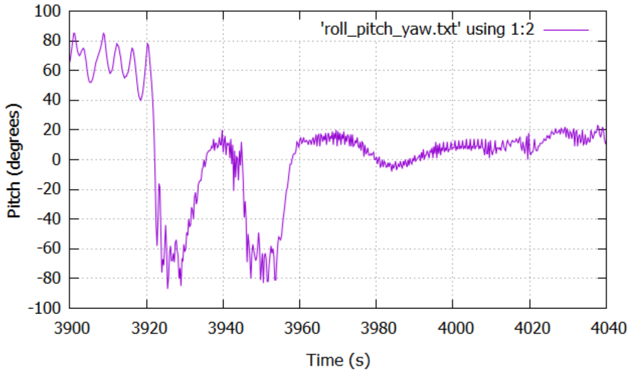

Fig. 7   Zoom-in on pitch angle flight data.

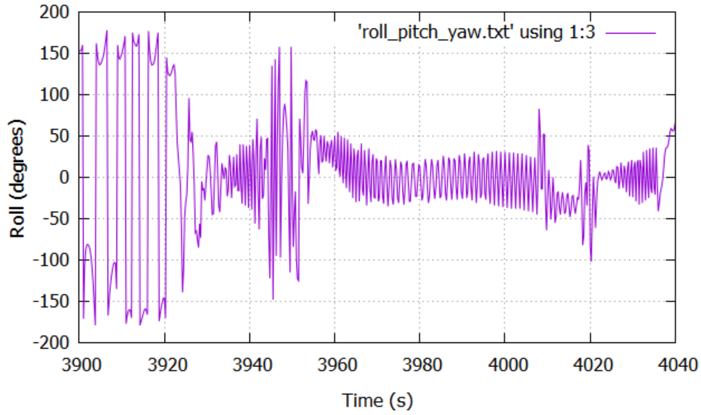

Fig. 8 Zoom-in on roll angle flight data.

As it can be observed the pitch and roll where quickly stabilized reaching values around 0 degrees in less than 20 seconds. Following the initial stabilization the control system switched to variable control coefficients ensuring a smother and more accurate stabilization of the flying wing as shown in Fig. 9 and 10.

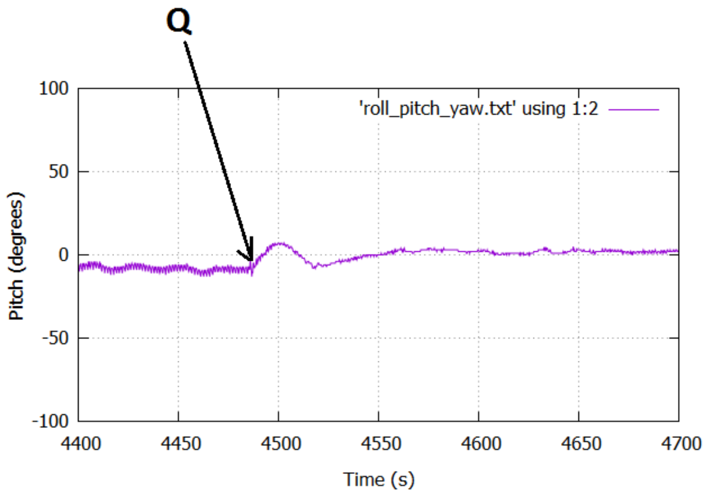

Fig. 9 Pitch performance with constant and variable command coefficients.

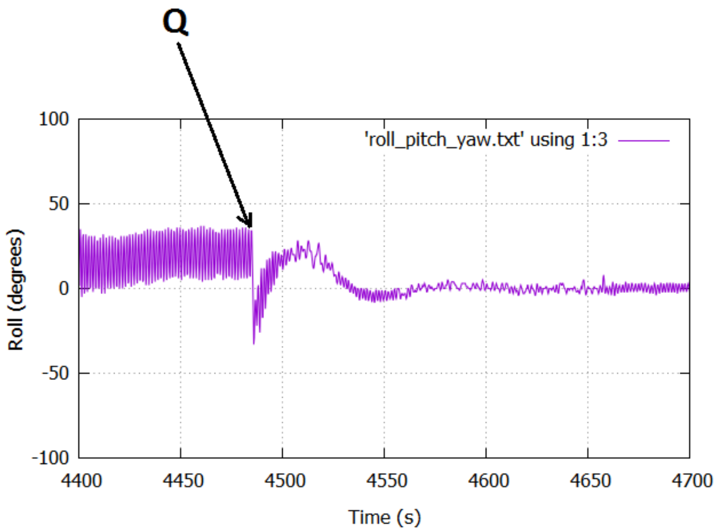

Fig. 10 Roll performance with constant and variable command coefficients.

The moment of transition from constant to variable command coefficients is denoted in Figs. 9 and 10 by point Q. The switching from constant to variable coefficients is simultaneous in both roll and pitch axis.

The variable control coefficient systems act in two phases:

1) Initial calibration: about 20 seconds
2) Actual control: about 4 seconds

The initial calibration is necessary because it is impossible to accurately predict the varying near-space atmospheric conditions before the flight. It actually uses the previous constant command coefficients to perform the calibration phase and searches for optimal command coefficients for the given conditions. Whenever the flight conditions change the system would re-adapt looking for the new command coefficients.

Fig. 11 shows roll and pitch angles with constant and variable command coefficients as well as the control efforts (control surfaces torques). Hence, it can be noticed that after switching to variable control coefficients the roll and pitch are stabilized closer to the desired target (0 degrees) while the control efforts (control surfaces torques) decrease significantly when compared to the values that they had during the constant command coefficient phase.

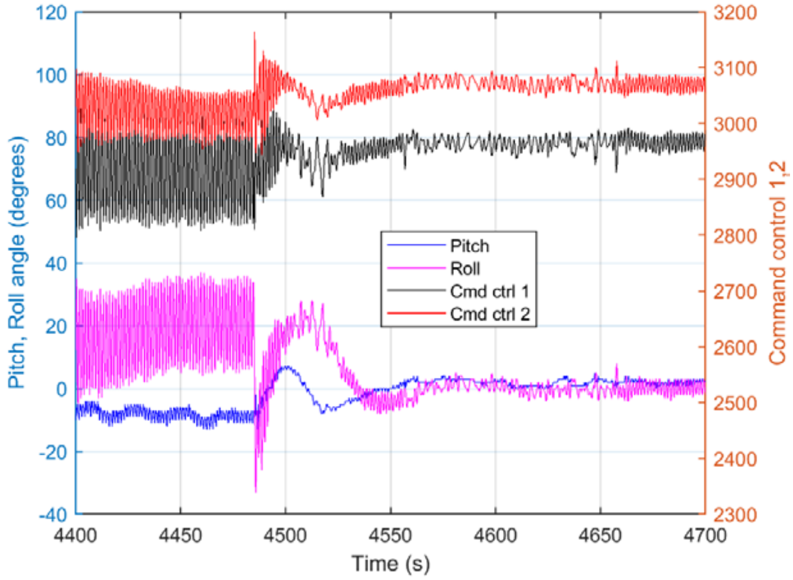

Fig. 11 Roll angle, pitch angle and control commands on channel 1 and 2.

5. Conclusions

A flying wing test vehicle was flown in near space region in order to provide a test flight for an agile quaternion based attitude determination and control system. The flight was performed from 27 km altitude using a weather balloon filled with helium as the carrier.

Upon separation, the autopilot stabilized the flying wing both in pitch and roll angles while the flight was taking place in near space conditions. In order to provide the stabilization a 6 DOF quaternion based model was programmed onto the autopilot and used tri-axial accelerometer and tri-axial gyroscope signals as inputs.

The pitch and roll were stabilized in less than 20 seconds despite the very low air density and very small control torques available. Initial stabilization was achieved through a constant command coefficient system. The constant command coefficient system switched to variable command coefficient system which was able to fine tune the pitch and roll while ensuring that this was achieved with minimum control effort (actuation of command surfaces).

The overall attitude determination and control system is being proposed both for near space vehicles as well as for space vehicles that aim to obtain an agile attitude determination and control with minimum on-board computational resources.

The novelty of the current work is the agile quaternion based attitude control applicable to near space vehicles. The attitude determination for a near space vehicle needs to be precise and fast in order to ensure an agile attitude control capability. The method proposed covers both the agile attitude determination and control. The agile attitude control exhibited excellent results providing fast and precise stabilization on pitch roll and yaw axis. The roll and pitch axis were stabilized from an arbitrary value to a commanded value despite the difficult flight conditions characteristic to near space flight vehicles.

The proposed agile determination and control system offers a unique and strategic near space vehicle test flight capability both in EU and outside of EU.

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