



Validation of flight dynamic stability optimization constraints with flight tests

J. Mieloszyk, A. Tarnowski, A. Tomaszewski, T. Goetzendorf-Grabowski *

Institute of Aeronautics and Applied Mechanics, Warsaw University of Technology, Warsaw, Poland

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ABSTRACT

Numerical optimization in aerospace sciences and industry is now used on a daily basis. Since, aircrafts are very complicated objects, analyzing multiple disciplines and using Multidisciplinary Design Optimization became a regular practice. Coupling both aerodynamic and structural analyses in the optimization framework is classical example. On the other hand, flight dynamic stability analyses are very rarely incorporated into the optimization process, or if so, in very limited way. Yet, having appropriate flight dynamic stability characteristics is crucial for the aircraft handling qualities, which are regulated by the airworthiness regulations. Methodology for the design and optimization coupled with dynamic stability constraints has been previously established by the authors of this article. This paper presents validation of this design methodology by comparing data acquired from the design and optimization process with the test flight data of UAV aircraft, which was designed basing solely on the developed methodology of the multidisciplinary optimization framework.

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1. Introduction

Aircraft preliminary design and optimization process often lacks dynamic stability constraints. Flight dynamic stability characteristics, on the precisely defined level, are required by flight regulations and aircraft handling quality norms [1][2]. During the preliminary aircraft design mainly static stability constraints are tested. The dynamic characteristics are usually analyzed as post-processing of design activities when the project is already advanced. If the aircraft is designed in a conventional configuration, in most cases a few adjustments are enough, but in the case of unconventional configurations, such as canard, flying wing, box-wing, or their variations, meeting the required flight dynamic stability characteristics can become an issue, as shown in [3]. As a consequence the project will have delays, more engineering hours and computational time will have to be spent and all this will rise costs.

Although, there have been developed a few design environments that consider dynamic stability of an aircraft within the early stage of design (e.g. CEASIOM [4][5] which was developed within SimSAC project [6]), their fundamental drawback is the lack of automatic feedback that could improve handling qualities without deterioration of other, mainly performance characteristics. There are many examples of successful application of such an approach, also for analysis of the unconventional configurations, with the use of full CEASIOM environment [7], or it's component dedicated for dynamic stability analysis [8][9] or using other computational tools [10][11][12][13].

The real improvement however can be only performed by including the dynamic stability constraints into the optimization loop. Only in few described design cases flight dynamic characteristics were considered during the early design process, or within the optimization procedure. Moreover, in the most of the previous cases they were limited only to the longitudinal characteristics [14][15], or to the lateral dynamic stability modes [16][17], or related to reduced configurations of isolated wing [18].

The idea of combining the basic disciplines used in aircraft design resulted in the first design framework, which linked aerodynamics with stability constraints [19], however the possibility of joining other disciplines was presumed. After few design cases exercised, the current numerical optimization design framework has been developed [20][21], which couples aerodynamic analyses, structural analyses, mass estimation and flight dynamic char-

* Corresponding author.

E-mail addresses: jmieloszyk@meil.pw.edu.pl (J. Mieloszyk), atarnowski@meil.pw.edu.pl (A. Tarnowski), atomaszewski@meil.pw.edu.pl (A. Tomaszewski), tgrab@meil.pw.edu.pl (T. Goetzendorf-Grabowski).

Symbols and Abbreviations

a_1	oscillations amplitude	S_x, S_y, S_z	static moments in body axis system
A	inertia matrix	t	time
b	wing span	T	period
CL	lift coefficient	u, v, w	velocity components in body axis system
d	traveled distance	V	flight velocity
F	forces acting on the aircraft	η	frequency coefficient
g	gravity acceleration	ζ	damping ratio
h	height of flight	λ	eigenvalues
I_{aircraft}	moment of inertia of the airplane about chosen axis	ξ	damping coefficient
I_{ring}	moment of inertia of the ring from the test rig	φ	phase shift
I_{xx}, I_{yy}, I_{zz}	moments of inertia about roll, pitch and yaw axis accordingly	ω	angular velocity
I_{xy}, I_{xz}, I_{yz}	inertia products	ω_n	undamped neutral frequency
K	angular momentum	Π	momentum
l	length of ropes of the test rig	Ω	angular velocity of an aircraft
m	mass of aircraft	AR	Aspect Ratio
m_{ring}	mass of ring from the test rig	CG	Center of Gravity
M_o	moments of forces	CFD	Computational Fluid Dynamics
n	number of measurements	FEM	Finite Element Method
p, q, r	angular velocity components in body axis system	L/D	Lift to drag ratio
R	radius of the rig plate	MDO	Multidisciplinary Design and Optimization
s	state vector	NORD	Numerical Design Results Demonstrator
S	reference area	TAS	True Airspeed
		TOW	Takeoff Weight

acteristics (Fig. 1). Dynamic stability component is analyzed repeatedly during the entire optimization run. In every optimization loop, full aerodynamic data needed for dynamic stability analysis is obtained. Aircraft's structure is resized and mass properties updated. Results from all these analyses are used for complete dynamic stability evaluation and constraints verification. This is the novel part of the optimization workflow and wasn't used previously in other aircraft optimization frameworks. The optimization process differs also from the traditional approach, where many assumptions are made by experienced designers. In the presented optimization approach physical relationships and constraints are set up, without any final result predictions. Only the technological and exploitation assumptions are made, which are reflected in constant values of chosen parameters, or constraints. The algorithm is free to search for the optimum solution within the defined optimization space limited only by physical relationships and constraints. At the beginning of the optimization, initial designs are created with random parameters and may give unfeasible solutions. In subsequent iterations, with changing optimization variables, more designs achieve feasible solutions. While the optimization process continues, all designs from the population give feasible solutions and have more uniform parameters close to the global optimum solution. This is a big advantage and strength of the optimization algorithm. Level of simulations complexity can be scaled by using simple analytical, or empirical relationships, which can be described by single equations, or by coupling optimization tools with advanced simulation tools for CFD, FEM and other types of simulations. Parallel computation is used for the simulations, which makes the optimization process more efficient. When the optimization framework is setup with simulation tools of moderate fidelity, like panel method and FEM solver [22], it takes single day for complete optimization run. The result is mature preliminary design of an aircraft.

2. Demonstrator configuration

Three aircrafts have been already designed with the developed optimization framework, in three different aircraft aerodynamic

configurations, respectively: box-wing [23], flying wing [24][25] and quad-plane based on the classical aerodynamic configuration [26]. To confirm true capabilities of the developed framework, the UAV in flying wing configuration has been designed within this procedure, manufactured and tested in flight. Flying wing configuration has long history [27][28] and is known for complex design with many issues [29][30][31] especially related to flight stability [32][33][34][35][36][37]. The designed aircraft was small size reconnaissance UAV with electric propulsion. The initial requirements for the demonstrator were payload of 1 kg and mission endurance of at least 1h. A good quality of recorded photographs and video for such small UAV requires operations at low altitudes below 500 m. The aircraft is launched with a bungee cord almost immediately and lands on relatively small and short grass fields. Therefore, the most significant mission segments are cruising and longtime loitering around observed object, which both requires good aerodynamic performance and low weight of the airframe structure. Average airspeed during the mission was considered to be between 15 m/s and 20 m/s, which ensures low energy consumption, effective object tracking capabilities and good wind resistance. Moreover, one of the most important design requirements for the aircraft was very good stability and appropriate maneuvering characteristics, to ensure good quality of the collected data and demonstrate usefulness of the design and optimization framework. Layout of the designed aircraft is presented in Fig. 2 and basic specification in Table 1.

During the flight campaign data was gathered and compared with the design predictions. Advantages of the presented approach are demonstrated, and accuracy of the design calculations are revealed by comparing the design predictions with the actual performance of the aircraft. The comparison focuses especially on the terms of the flight dynamic characteristics.

The designed aircraft, using introduced optimization framework, is called NORD – Numerical Design Results Demonstrator and is presented in Fig. 3. The flying wing configuration, is one of the aircraft's possible configurations, that may be especially vulnerable to poor flight characteristics. Main specification of the aircraft is given in Table 2. The aircraft was equipped with the

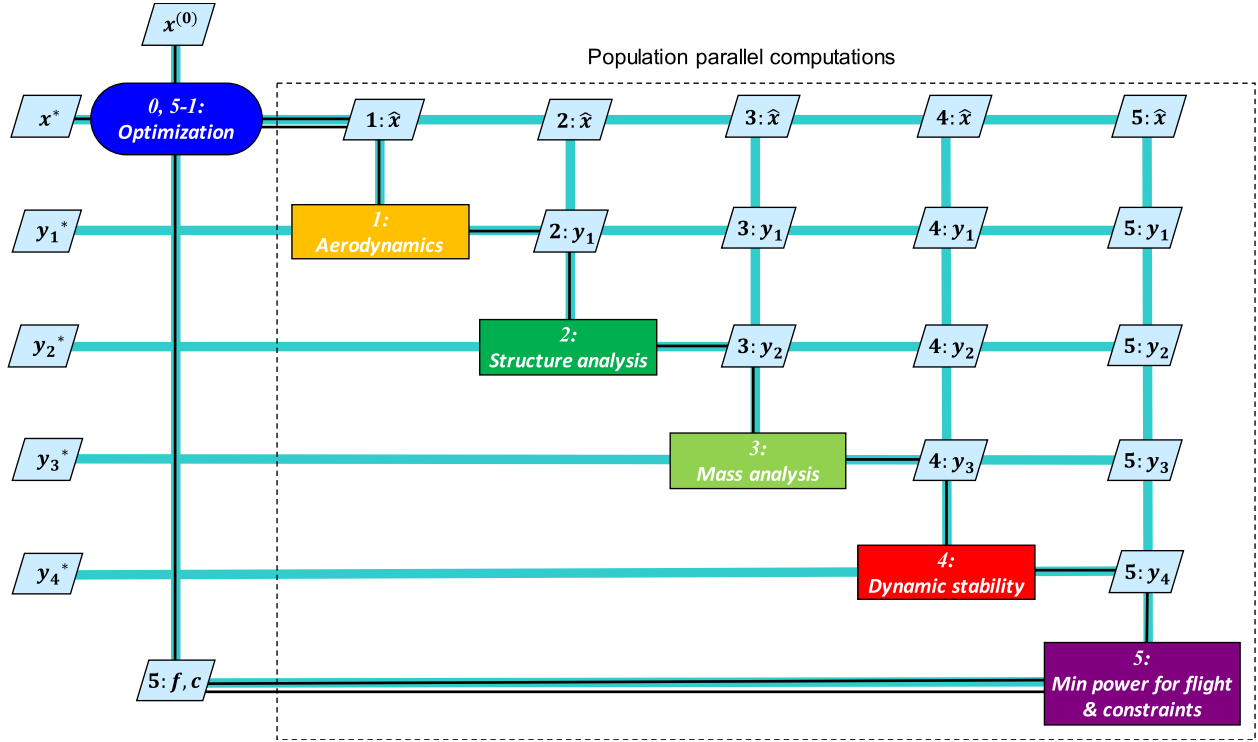


Fig. 1. Developed design and optimization framework.

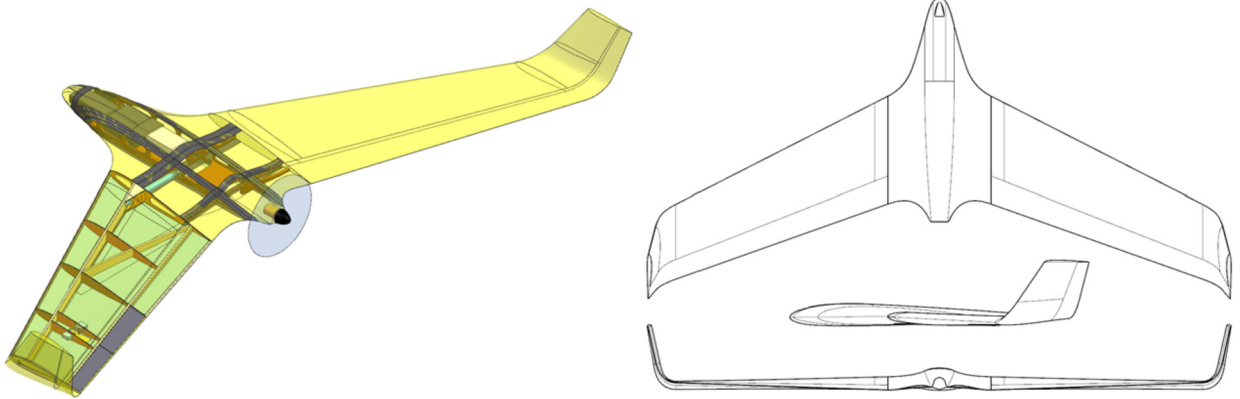


Fig. 2. The designed aircraft layout.

Table 1
NORD aircraft main specification.

Aircraft specification		
b	2.4	m
S	1	m ²
AR	5.8	–
m_{TOW}	5	kg
$m_{payload}$	1	kg
V_{min}	12	m/s
V_{cruise}	18	m/s
V_{max}	30	m/s
propulsion	electric	
configuration	flying wing	

autopilot Pixhawk 2.1 Cube (Fig. 4). While performing test flights autopilot's stabilization and navigation capabilities were disabled and only data acquisition was active. Hence, during the flight tests the aircraft had only its own uninfluenced natural stability characteristics, which were investigated.

3. Dynamic stability of an aircraft

The dynamic stability analysis was made by use of SDSA package, developed and validated within SimSAC project [38] and some further projects [8][9]. The equations of motion are derived from fundamental laws of mechanics - the balance of momentum theorem (1) and the balance of the moment of momentum theorem (2).

$$\frac{\partial \bar{\Pi}}{\partial t} + \bar{\Omega} \times \bar{\Pi} = \bar{F} \quad (1)$$

$$\frac{\partial \bar{K}_0}{\partial t} + \bar{\Omega} \times \bar{K}_0 + \bar{V}_0 \times \bar{\Omega} = \bar{M}_0 \quad (2)$$

The necessary coordinate systems and the loads components are presented in Fig. 5. The assumed body axis system is typical for flight dynamics discipline [39][40][41]. The equation of motion derived from equations above can be written in the form (3).



Fig. 3. UAV aircraft NORD – Numerical Optimization Results Demonstrator.

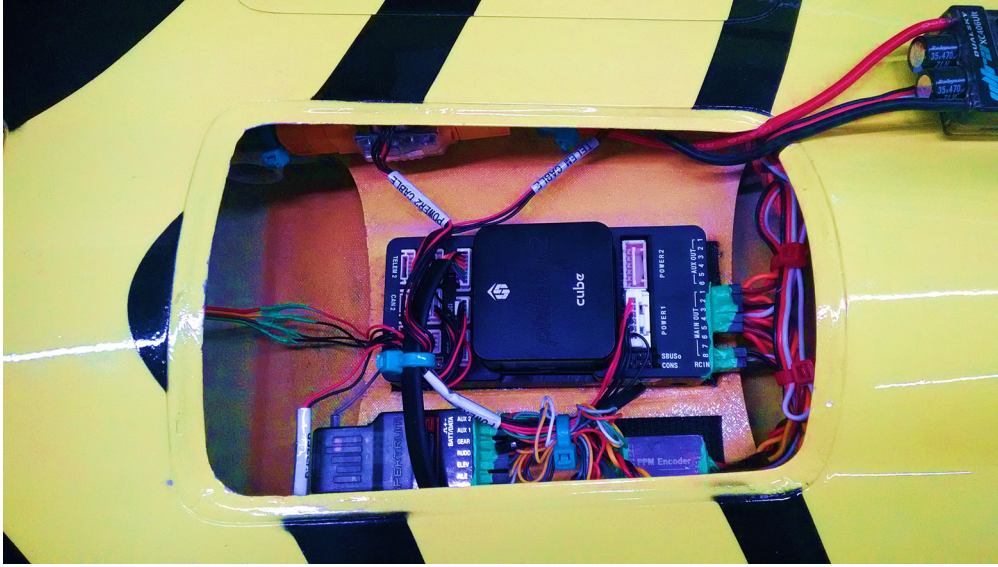


Fig. 4. Autopilot installed in the aircraft.

$$\frac{ds}{dt} = A^{-1}\bar{F}(s, \dot{s}, t)$$

where: $s = \{u, v, w, p, q, r\}^T$

$$A = \begin{bmatrix} m & 0 & 0 & 0 & S_z & -S_y \\ 0 & m & 0 & -S_z & 0 & S_x \\ 0 & 0 & m & S_y & -S_x & 0 \\ 0 & -S_z & S_y & I_{xx} & -I_{xy} & -I_{xz} \\ S_z & 0 & -S_x & -I_{xy} & I_{yy} & -I_{yz} \\ -S_y & S_x & 0 & -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix}$$

The set of six dynamic equations must be completed with six kinematical equations that link global position coordinates with velocity components and attitude angles with angular rates. For stability analysis only two kinematical equations are needed, those that link roll angle (ϕ) and pitch angle (θ) with angular rates [42]. The nonlinear set of equations (3) must be linearized, which is done by direct linearization of the right side of equation (3) by calculation of the Jacobian (J_8) for the defined state of the flight (4). Index “8” in the following formulas is used to notice that the dimension of considered problem is extended to 8).

$$(3) \quad \frac{ds_8}{dt} = A_8^{-1}\bar{F}_8(s_8, \dot{s}_8, t) \Rightarrow \frac{ds_8}{dt} = J_8 s_8 \quad (4)$$

where: $s_8 = \{u, v, w, p, q, r, \phi, \theta\}^T$

$$J_8 = \left\{ \frac{\partial (A_8^{-1}\bar{F})_{i,j}}{\partial s_j} \right\}$$

Now, the eigenvalue problem (5) can be formulated.

$$[J_8 - I\lambda]s_8 = 0 \quad (5)$$

The solution of the eigenvalue problem gives directly the frequency and damping coefficients. The eigenvector problem is also solved to identify the motion modes.

The solution of the eigenvalue problem has the general form (6).

$$\lambda = \xi + i\eta \quad (6)$$

The damping ratio and undamped frequency could be obtained from formulas (7) and (8) respectively.

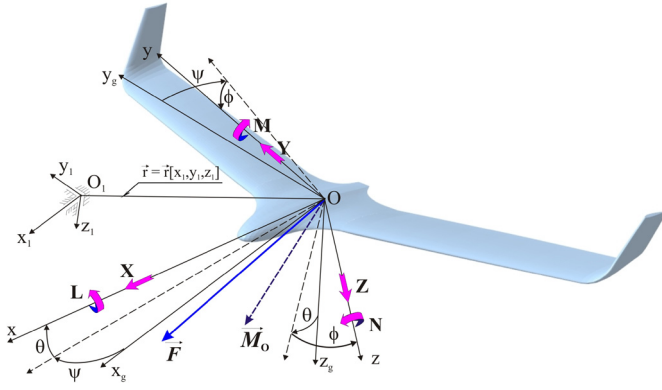


Fig. 5. Coordinate systems and forces components.

$$\zeta = \frac{-\xi}{\sqrt{\xi^2 + \eta^2}} \quad (7)$$

$$\omega_n = \sqrt{\xi^2 + \eta^2} \quad (8)$$

The period and damping halftime (or double time in unstable case) could be computed from the formulas (9) and (10).

$$T = \frac{2\pi}{\eta} \quad (9)$$

$$T_{1/2} = -\frac{\ln 2}{\xi} \wedge \xi < 0 \quad (10)$$

$$T_2 = \frac{\ln 2}{\xi} \wedge \xi > 0$$

4. Aircraft moments of inertia measurement

The equation (3) shows that the most important data that influences flight dynamic characteristics are aerodynamic characteristics and mass distribution, which affect moments of inertia. Aerodynamic characteristics were computed using PANUKL solver and mass breakdown was estimated using simplified methods. To have better comparison of computed flight dynamic stability characteristics with the one measured in flight, moments of inertia of the aircraft, in the flight configuration, were measured on the ground. Using the same computational procedures, which were used for analyses of dynamic stability characteristics during the design [38][43], new dynamic stability characteristics were computed with measured values of the moments of inertia.

Moments of inertia were measured on the test stand presented in Fig. 6. Annulus for aircraft placement was made of laminated foam and suspended on three ropes. The aircraft was placed on the ring in three different orientations depending about which axis moment of inertia was measured. The annulus with positioned aircraft had to be slightly twisted from the neutral balance position. When released the aircraft rotated forth and back like a pendulum. The CG of the annulus and aircraft had to be collinear with the vertical axis. The geometry of the ring and its moments of inertia were known. By measuring period of the oscillations it was possible to calculate the moment of inertia about vertical axis, which was going through the center of the ring.

Taking into account that all ropes are equally loaded and taking notations from Fig. 7, the equation of angular oscillation can be written in the form (11).

$$I\ddot{\varphi} = mgR \sin \gamma = mg \frac{R^2}{l} \varphi \quad (11)$$

after dividing side by side by I the general oscillations equation is derived (12).

Fig. 6. The test stand with airplane prepared for the I_{yy} measurement.

$$\ddot{\varphi} = \omega^2 \varphi \quad (12)$$

where: $\omega = R \sqrt{\frac{m}{I} \frac{g}{l}}$.

The period of the oscillations of the measured system is defined by formula (13).

$$T = \frac{2\pi}{\omega} \quad (13)$$

Rearranging equation (13) through (14) to the form (15) moment of inertia of the aircraft could be calculated. To improve accuracy of the period measurement, the time of twenty oscillations was measured for every axis of rotation of the airplane and the period was computed using formula (16). The second method of measurements was the registration of the angular velocity ω of the setup by use of accelerometers. The general oscillations equation (17) was a base to approximate the registered $\omega(t)$. The equation values were tuned for the best fit of the model to the recorded data using minimum square method. Comparison of the actual angular velocity of the setup and model prediction matches very good. Fig. 8 shows example of the comparison of the measured and estimated angular velocity for the setup in the z axis of the airplane.

$$T = C \sqrt{\frac{I_{\text{aircraft}} + I_{\text{ring}}}{m + m_{\text{ring}}}} \quad (14)$$

where: $C = \frac{2\pi}{R} \sqrt{\frac{l}{g}}$

$$I_{\text{aircraft}} = \frac{T^2 (m + m_{\text{ring}})}{C^2} - I_{\text{ring}} \quad (15)$$

$$T = \frac{t}{n} \quad (16)$$

$$x(t) = x_0 + a_1 \cdot e^{-\zeta \cdot \omega_n \cdot t} \cdot \sin(\omega \cdot t + \varphi) \quad (17)$$

Table 2 compares mass properties of the aircraft calculated during the design and within the measurement tests. The aircraft was designed for 1 kg of payload, which should be placed in the front part of the fuselage. During the test flights the payload bay was

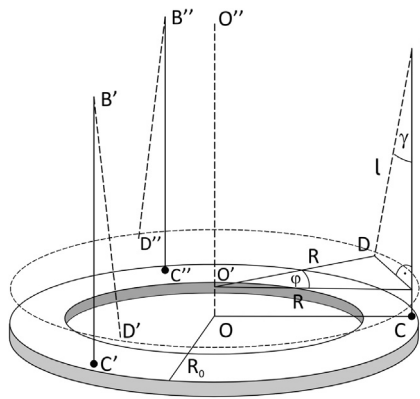


Fig. 7. The annulus scheme used for moments of inertia measurement.

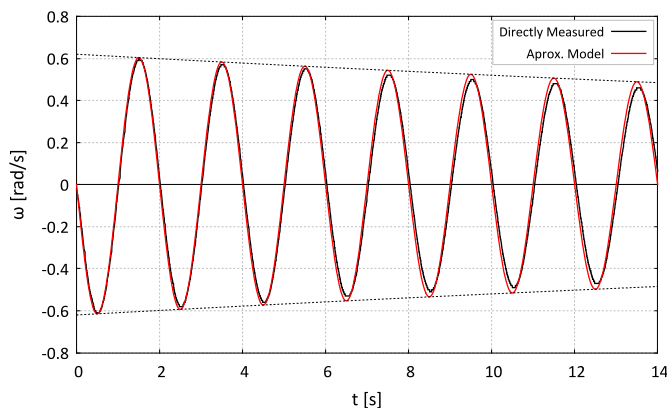


Fig. 8. Oscillations of the angular velocity of the setup for moments of inertia measurement – axis z of the airplane.

Table 2
Mass properties comparison.

			Design	Aircraft			
				min x_CG	% diff	max x_CG	% diff
m	[kg]	5	5.038	0.8%	4.742	5.2%	
CG pos.	[m]	0.657	0.630	4.1%	0.664	1.1%	
lxx	[kg m ²]	1.465	1.587	8.4%	1.587	8.4%	
lyy	[kg m ²]	0.289	0.571	97.3%	0.511	76.7%	
lzz	[kg m ²]	1.693	1.994	17.8%	1.935	14.3%	

empty, but additional mass was added on the nose of the aircraft to gather data for different levels of aircraft's longitudinal stability. Additional mass varied from 300 g for the most forward position of center of gravity and the most stable aircraft to no additional mass on the nose, which made aircraft still flyable, but almost neutrally stable. Measured I_{xx} and I_{zz} moments of inertia match quite well with the design predictions, but I_{yy} moment of inertia is much less accurate. UAV length of flying wing configuration is short, which influences I_{yy} moment of inertia estimation. This indicates that more careful methods of moments of inertia estimation are needed. Incorporating larger number of mass components and their own moments of inertia would improve the accuracy of moments of inertia calculation.

Comparison of the dynamic stability characteristics with the updated moments of inertia is shown in the following chapters.

5. Dynamic stability validation

To validate the flight dynamic stability characteristics data was collected during test flights and compared to the design predictions. The data was recorded with average recording frequency

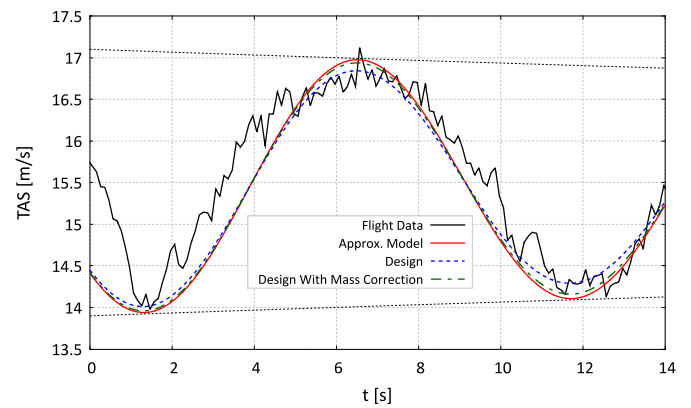


Fig. 9. Phugoid - TAS oscillations.

25 Hz using installed onboard autopilot Pixhawk 2.1 Cube. During flights the autopilot stability and navigation functions were disabled to allow only for aircraft's own natural dynamic stability characteristics.

All test flights were conducted in very calm weather conditions with wind speed ranging from 0 to 2 m/s throughout most of the test flights time. To initiate oscillations of the particular stability modes, different control surfaces were deflected in a singlet, or doublet input maneuvers [44]. After applying both types of the disturbance input, controls remained in neutral positions to observe how the oscillations are naturally damped. All the data for stability analyses and comparisons was recorded in the time periods where no control surfaces were deflected, or throttle changed.

Initially, within the design process, the modes of motion were computed using aerodynamic data from PANUKL computation and estimated mass and inertia data. In order to have better accuracy of the predictions, moments of inertia were measured for the aircraft ready to fly and taken to compute modes of motion. Therefore two computed values of frequency and damping coefficients could be compared with those obtained from the flight tests. The recorded data of selected state variables were approximated using the general oscillations equation (20). The comparison is presented in Table 3. Detailed considerations for each of the periodical modes are presented in the following subchapters.

5.1. Phugoid

The phugoid oscillations are characterized by long period and can be mildly damped. The oscillations were excited by gentle increase of the airspeed by impulse deflection of the elevons. Recorded oscillations of the airspeed, its approximation and curves obtained within numerical analysis are presented in Fig. 9. Table 3 presents values of undamped natural frequencies and damping ratios identified from recorded flight parameters, obtained from numerical analysis performed during the design stage and after correction of the mass and inertia data. The compliance of the results is pretty good for frequency. The computed damping characteristics are slightly overestimated, which means that oscillations are not so strongly damped as in the real flight. The damping characteristics match more better in the case of the corrected moments of inertia, which shows sensitivity to the mass breakdown.

5.2. Short period

Short period oscillations characterized by fast changes of the angle of attack and pitch rate while the airspeed is almost constant. This mode is usually very strongly damped and therefore it is difficult to identify it within the recorded flight data. The time history of the pitch rate during two maneuvers that excited this

Table 3
Flight dynamic stability characteristics.

	Aircraft's		Design mass properties				Measured aircraft's mass properties			
	ω_n	ζ	ω_n	ζ	% ω_{n_err}	% ζ_err	ω_n	ζ	% ω_{n_err}	% ζ_err
Phugoid	0.605	0.018	0.627	0.032	3.4%	44.3%	0.583	0.023	-3.8%	23.1%
Short Period	9.5	0.5	12.632	0.562	24.8%	11.0%	10.422	0.534	8.8%	6.3%
Dutch Roll	2.8	0.11	2.861	0.082	2.2%	-34.1%	2.711	0.103	-3.3%	-26.0%

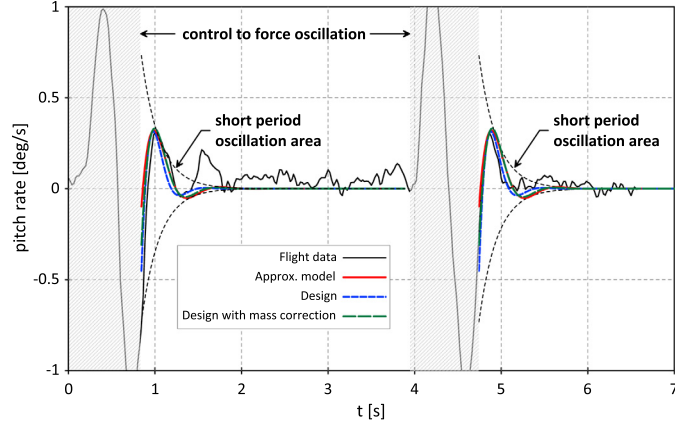


Fig. 10. Short period – pitching oscillations.

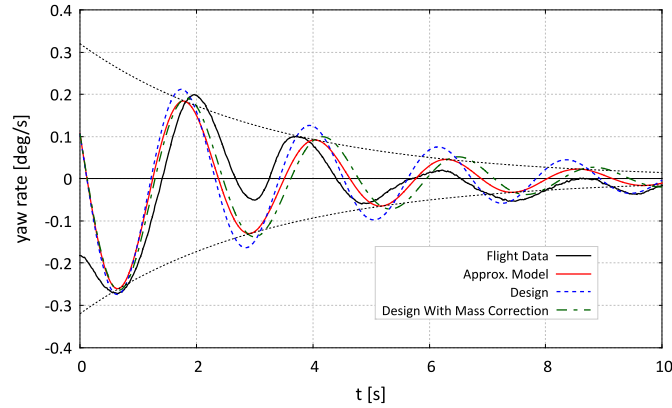


Fig. 11. Dutch Roll – yawing oscillations.

mode is presented in Fig. 10. The comparison of oscillation parameters is presented in Table 3. Parameters computed after the moments of inertia correction show good compliance with flight data in terms of frequency and damping. It should be noted that oscillations forced by controls have stopped after about 1.5 s. Subsequent disturbances are not associated with the short period oscillation and are caused by minor gusts.

5.3. Dutch roll

The requirements related to combined lateral-directional oscillations called Dutch roll are precisely defined in the airworthiness regulations [1]. Dutch roll oscillations must be damped to 1/10 amplitude in 7 cycles. Fig. 11 shows oscillations of yaw angular rate with fitted oscillations model. The aircraft's flight data oscillations frequency and damping ratio parameters and design computations corresponds quite well. The computation results with corrected inertia characteristics are slightly better than the first Dutch roll estimation within the initial design.

Table 4
Measured L/D ratios for different flights.

L/D Characteristics			
V	CL	L/D	% error
15.85	0.32	11.64	16.9%
15.41	0.34	10.31	26.4%
17.13	0.27	10.38	25.9%

5.4. Discussion

The comparison of stability characteristics obtained from test flights and computed within the design stage of the project shows good compliance. The relative error is less than 10% for all frequency characteristics and slight bigger for damping ratio in case of phugoid and Dutch roll. The results comparison indicates that it would be advisable to increase precision in determining moments of inertia.

It has to be noted that RC pilot's assessment of the aircraft and visual judgment of oscillations during the flights were positive. All observed oscillations were quickly and heavily damped. The heavy damping was especially pronounced with the lack of visible lateral oscillations, which are very common for small UAV in flying wing configuration.

6. Aerodynamic performance measurements

Beside flight dynamic stability validation, an aerodynamic performance was undertaken. Aircraft made three separate, straight fly-overs through the airfield with no control surfaces movement and with the motor turned off. The aircraft was equipped with the propeller with folding blades, which adds less to minimum drag in the gliding configuration. Average airspeed was calculated using fragments of the flight data with approximately linear drop of altitude (18). Distance traveled within the considered time frame was calculated using expression (19). The resulting L/D ratio is calculated as the traveled distance divided by altitude drop (20). The height is calculated as the difference of the recorded altitude at the beginning and at the end of the flight segment.

$$V_{\text{average}} = \frac{\sum_{i=1}^n V_i}{n} \quad (18)$$

$$d = V_{\text{average}} \cdot t \quad (19)$$

$$L/D = \frac{d}{h} \quad (20)$$

Obtained values of L/D ratio for different test flights are gathered in Table 4. The results are also compared to aerodynamic results from CFD – Fluent [45] computations shown in Fig. 12. It can be observed that resulting L/D values for three flights gave similar values, all of which are slightly below CFD predictions. The difference between the CFD computations and three closely located measured points was expected and can be explained by surface imperfections on the actual flying aircraft e.g. covers, control surfaces gaps and additional objects generating drag like e.g. radio antennas.

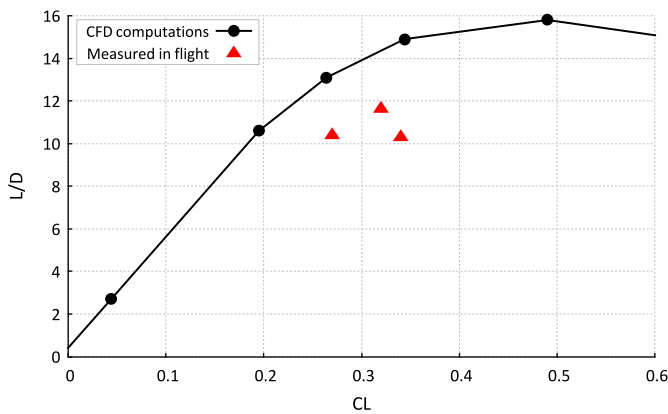


Fig. 12. L/D characteristics.

7. Conclusion

The outcome of this validated optimization framework has resulted in very stable aircraft. Flight dynamic characteristics were confirmed by comparison of flight parameters, RC pilot's aircraft assessment and in flight observation. In few cases, stability parameters have noticeable differences, what indicates possible further simulation models improvement of the aerodynamics and mass distribution predictions. Aerodynamic performance of the aircraft in flight was also measured and compared with the CFD results, which revealed good data correlation.

Pilot's introduced aircraft oscillations were heavily damped, both in lateral and longitudinal axis. Both singlet and doublet inputs were performed during test flights to collect the necessary data points for further analysis. The oscillations were barely noticeable throughout visual observation. It is worth noting that the aircraft does not have any excessive adverse yaw behavior. The oscillations in yaw axis are not visible as well. It is easy to maintain a correct heading which is usually problematic in the flying wing airplane configuration. During the last phase of approach a strong ground effect can be observed, which makes the aircraft reluctant to land. Good aerodynamic efficiency together with lack of spoilers or flaps has contributed to this effect. This shortcoming is not causing much trouble due to the aircraft good stability. It is easy to maintain level flight, even through prolonged flare phase of the landing.

The overall platform stability gives pilot a trust and confidence during maneuvers. All of that makes the platform suitable for use in photogrammetry, even in a harsh weather environment as well as makes it easy to integrate with most of the commercial autopilot systems.

The results presented in this article prove the applicability usefulness of the methodology proposed in previous works [19][20]. The application of the multidisciplinary optimization process gives exceptionally good results in the design of an aircraft in unconventional configuration, where traditional approach is not sufficient to design an airplane with the correct flight characteristics. The presented flight tests results of the aircraft built as a flying wing and their comparison with computational results, prove not only the advantages of multidisciplinary optimization methodology implemented at the initial stage of the project, but also good quality methods used to solve problems of particular disciplines [22][38].

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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