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Retrofitting of an ultra-light aircraft for unmanned flight and parachute cargo dropping

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ABSTRACT

Despite the increasing interest in unmanned aerial vehicles (UAV), their adoption in commercial flight operations invariably meets with skepticism, mainly on the base of safety and reliability concerns, as well as poor payload and endurance of available - generally rotary-wing - platforms. However, there exist specific missions where higher-weight UAVs may be employed, specifically to serve wild or disadvantaged areas, far from crowded regions, and transporting medical aids or food. Clearly a niche too small to be considered profitable as a market for a new design by industry, this requirement can be fulfilled through the partial re-design of an aircraft in the light sport aircraft (LSA) weight class. Based on a set of specifications discussed with Médecins Sans Frontières (MSF), the present paper analyzes the feasibility of a mission where medical aids are carried over a prescribed route by means of an UAV, and parachute-dropped on the target area. A candidate for the proposed mission is found in an existing LSA. Its optimal use and the corresponding retrofitting steps to fit within the prescriptions of the mission are proposed and critically discussed.

1 INTRODUCTION

In recent years, unmanned aerial vehicles (UAV) have experienced a tremendous growth in several sectors of the aviation market (Nonami et al. 2010). Historically adopted firstly for military purposes, where successful use dates back at least to the 1990s conflicts in former Yugoslavia, conventionally-powered UAVs, i.e. typically featuring a propeller and an internal combustion engine, or a turbofan, are today widely deployed for PHOTINT or SIGINT missions, as well as to a more limited scale for aggressive actions by many Air Forces (Walsh and Shulzke 2018). Most military drones are designed in a fixed-wing configuration, and when capable of carrying a larger non-disposable (i.e. not expendable) payload, they are typically optimally designed for altitude, range and endurance. Whilst less mechanically complex and cheaper to operate than military cargo or attack aircraft, these machines are generally high-technology platforms, in a range of acquisition and operation cost far beyond commercial use.

On the other end of the spectrum, rotary-wing, electrically powered drones, aggressively put on the commercial market more recently, are extremely cheap to acquire and fly, but have the shortcoming of limited payload, and poor endurance or range performance, the latter being limited by low values of battery energy density today available - similarly to electric aircraft in any weight class (Riboldi and Gualdoni 2016). As a matter of fact, similar UAVs are typically relegated to the entertainment flight sector, or to commercial activities involving either low payload or short flight time.

In commercial flights with cargo (i.e. not passenger) payload, UAVs are currently facing regulatory issues, in turn stemming from safety concerns when it comes to overfly crowded areas (Motlagh et al. 2016). Consequently, UAVs in the weight and payload range of light sport aircraft (LSA) or light general aviation (GA) aircraft still do not make for a readily exploitable market, and such designs are currently seldom proposed from scratch despite the current push in this sector, which is mostly connected with the adoption of novel propulsion systems (Riboldi et al. 2020b) and related technologies (Riboldi et al. 2020a). However, the option to partially re-design existing LSA or light GA machines for the task is indeed interesting for some specific missions, where purchase/operation cost is at a premium, payload/range requirements fit those of such category, and operations are to be carried out away from crowded areas.

This paper explores the latter scenario, focusing on a specific case study. In particular, discussions with Médecins Sans Frontières (MSF) have highlighted the need for an aircraft capable of reaching human settlements at a distance of some hundred kilometers from the nearest airstrip, and poorly linked by road connections. The payload would be medical supplies in a quantity limited to the periodical needs of a disadvantaged remote community, assisted by MSF staff. The lack of available landing airstrip at destination entails the need for a parachute drop of the cargo.

On the base of severe cost constraints for such humanitarian mission, and potential risk for a pilot in overflying wild areas where guerrilla operations may be taking place, an UAV would be a valuable option, also matching the low complexity of such cargo-deployment mission. Similarly, the safety risk connected with catastrophic control loss would be minimal when overflying a wild area.

Clearly, an aircraft designed from scratch for this specific niche would capture little interest from aircraft manufacturers. Therefore, the re-design of an existing platform may be a valid alternative. This can be met through a delicate retrofitting operation on a suitable machine.

In the body of the work, a detailed analysis of the mission is proposed, negotiating the requirements in an optimal way. The analysis takes into account the parachute-dropping phase. Then an existing LSA is considered, matching the mission requirements, and key aspects of the re-design are analyzed, dealing with the most substantial modifications needed to make the aircraft capable of autonomous flight.

In the conclusion, the actual feasibility of the retrofitting and the suitability of the resulting UAV platform for the intended mission are critically discussed.

2 MISSION STUDY FOR RETROFITTING: SPECIFICATIONS, CHOICE OF PLATFORM AND OPTIMIZATION

In the discussions with MSF, the following mission requirements have emerged

1. a payload mass of 250 kg, pharmaceuticals and medical aids
2. a target range between 200 and 600 km, i.e. doubled for a round trip
3. parachute-dropped cargo, automated
4. autonomous flight, chance of remote control in terminal flight phases (no landing aids expected)
5. take-off/landing from unprepared runway
6. non-military affiliation of manufacturer/civilian certification of aircraft, to ease import in target Countries

Many LSA aircraft would suit the requirements, especially considering the increase in payload obtained from the retrofit of a suitable cargo bay instead of the passenger compartments. A good example, which also provides the advantage of a classical metal tubes and sheet construction allowing for easier modifications than a composite airframe, the Groppo G70 has been selected as a testbed. Ongoing contacts with the manufacturer have allowed to study the modifications and assess the feasibility of the retrofit more in depth. Basic technical specifications of the G70 and a portrait are shown in Fig. 1 and Tab. 1 (G70 POH 2018). For the scope of preliminary design, the mission of interest can be described as a cargo transport cruise, with a parachute drop of the cargo, associated to a sudden drop in the weight of the machine. The corresponding mission profile is composed by a climb, a cruise to the drop point, a new climb to the return cruising altitude, and a return cruise. Take-off and descent are not considered, since little impacting the weight (either for the short duration of the former, or the reduced fuel consumption in the latter).

Considering a generic existing fuel-burning LSA, based on its weight, power-train characteristics and aerodynamic polar, it is possible to first assess its suitability for a certain range. By adopting an optimal approach in the verification of the compliance with requirements, it is possible to simultaneously impose constraints pertaining to the new mission (e.g. equal outbound and return range), specifying fixed and non-negotiable aero-propulsive parameters (e.g. aircraft polar, engine power), and obtaining - in case a solution compliant with the constraints is found at all - the values of flight mechanics parameters for optimally exploiting aircraft characteristics on the new mission specifications. In the present work, a simple optimization algorithm is employed, in order to assess the maximum range of the mission, by suitably selecting the airspeeds and altitudes for both the outgoing and return cruising phases (which correspond to two significantly different values of weight, due to payload dropping on target). In the proposed implementation, range will not be constrained explicitly, and will be optimized instead. By leaving range free to vary as an outcome of computations, the compliance with respect to the range requirement specified in the new mission profile needs to be checked *a posteriori*. However, this approach allows to carry out sensitivity analyses on range more easily, as will be shown in the application section ref 2.3.

To better explain this approach, we introduce an analytic expression for range $\mathcal{R}$ as

$$\mathcal{R} = \int_{W_2}^{W_1} \frac{\eta_p}{g c_p} \frac{C_L}{C_D} \frac{dW}{W}, \tag{1}$$

where η_p is the propeller efficiency, g is gravitational acceleration, c_p is the brake specific fuel consumption of the engine, C_L and C_D lift and drag coefficients respectively, and W is the weight of the aircraft, decreasing over the mission due to fuel consumption. Conditions 1 and 2 in Eq. 1 refer to a generic initial and final condition of the cruise. For the outgoing cruise, they will correspond to the condition at the end of the climb phase and up to the parachute dropping over target, whereas for the return cruise they will correspond to the after-drop condition and the start of the descent phase respectively.

Now, range can be subject to an optimization considering that engine characteristics (represented by c_p and η_p in Eq. 1) can be seen as variables. Furthermore, the weight profile over time is a further variable, function of several quantities including airspeed, altitude and power. A numeric optimization algorithm will be employed, after making the dependencies just cited explicit.

2.1 Mission Modeling for Range Optimization

The specific mission of interest here is composed by four phases, two climbs and two cruises, for which slightly different models apply.

Climb

For climb, it is assumed that the airspeed $V = V(h)$ and climb rate $V_v = V_v(h)$ are assigned functions of the altitude h, obtained from the performance of an existing aircraft such to maximize the rate (fastest climb condition), usually published on flight manuals. The target cruising altitude for the first cruising leg h_{cr1} is the optimal variable in this phase, and is assigned by the optimizer in the following computation. By discretizing the altitude domain between the initial (h_0) and final (h_{cr1}) altitudes through a suitable set of n_{climb_1} intervals, the corresponding fractional time to climb can be written as

$$\Delta t_i = \frac{\left(h_{cr1} - h_0\right)/n_{climb_1}}{V_{v_i}}, \tag{2}$$

where $V_{v_i} = V_v(h_i)$ is the vertical speed corresponding to the current i-th altitude during climb. At the same i-th altitude, the lift and drag coefficients can be computed, the former from force equilibrium in the direction of gravity, the latter from an assigned polar of the aircraft, yielding

$$C_{L_i} = \frac{m_i\left(g + \frac{V_{v_{i+1}} - V_{v_i}}{\Delta t_i}\right)}{\frac{1}{2}\rho_i V_i^2 S}, \qquad C_{D_i} = C_D\left(C_{L_i}\right), \tag{3}$$

where m_i is the mass of the aircraft at h_i, obtained from the initial mass of the aircraft, reduced by the integral of the fuel flow, as will be clear at the end of this paragraph. In climb, the power balance is reported in Eq. 4,

$$P_{a_i} = P_{r_i} = \frac{1}{2}\rho_i V_i^3 S C_{D_i} + \frac{m_i\left(V_{i+1}^2 - V_i^2\right)}{\Delta t_i} + W_i \cdot V_{v_i}, \tag{4}$$

where the power available from the propeller P_{a_i} is set equal to the power required for flight, P_{r_i}, itself composed of a term due to drag, one such to produce an acceleration along the trajectory, and a last one for climb (as on the r.h.s. of Eq. 4). Now, the power required from the engine P_{b_i} can be computed as

$$P_{b_i} = \frac{P_{r_i}}{\eta_{p_i}}, \tag{5}$$

where it is assumed to know the propeller efficiency $\eta_{p_i} = \eta_{p_i}(J_i)$ as a function of the propeller advance ratio $J_i = \frac{V_i}{\omega_i \bar{r}}$. Here $\bar{r}$ is the radius of the propeller. The advance ratio can be computed for an assigned V_i and from the knowledge of $\omega_i = \omega_i(V_i)$, which can be obtained for equilibrium conditions for a specific aircraft, when the engine and propeller characteristics are known. Finally, assuming to know the engine characteristics, it is possible to compute the fuel flow $\dot{F}_i$ as a function $\dot{F}_i = \dot{F}(P_{b_i}, h_i)$, so that the mass decrease for the i-th step is $\Delta m_i = \dot{F}_i \Delta t_i$, and $m_{i+1} = m_i - \Delta m_i$, as required in Eq. 3 and 4. Specifically, the total decrease of mass during climb F_{climb_1} can be evaluated as

$$F_{climb_1} = \sum_{i=1}^{n_{climb_1}} \dot{F}_i \cdot \Delta t_i. \tag{6}$$

The computations pertaining to the second climb, taking the aircraft from the outbound cruising altitude to the return one, follow the very same passages just outlined for the first climb.

Cruise

The fuel required for cruise is split over the two cruising phases. They will be different due to the difference in weight after cargo dropping. As an operative choice, the altitude of the aircraft during each cruising leg is kept fixed, whereas speed is allowed to change, and is therefore treated as an optimal variable (see later Eq. 8). The value of the fuel required for flying the first cruise, F_{cr1}, is computed through Eq. 7,

$$F_{cr_1} = \eta_f \cdot (F_{tot} - F_{climb_1} - F_{climb_2}), \tag{7}$$

where η_f is an optimization parameter, considered known in the computations to follow, and allows to define the share of the total fuel available for the outbound and return cruises (represented by the term between parentheses in Eq. 7) corresponding to the outbound cruise leg. From Eq. 7, the value of the fuel mass for the outbound cruising phase is computed. Similar to climb, cruise can be discretized into n_{cruise_1} intervals, so that the fuel for each of them is $F_k = \frac{F_{cr_1}}{n_{cruise_1}}$, with $k = 1, 2, \ldots, n_{cruise_1}$. Lift and drag coefficients can be computed from equilibrium in steady flight, and similarly the power balance for cruise can be computed accounting only for drag and speed change, i.e. null climb rate, in the power required figure, yielding

$$C_{L_k} = \frac{m_k g}{\frac{1}{2}\rho_{cr_1} V_k^2 S}, \quad C_{D_k} = C_D\left(C_{L_k}\right), \quad P_{r_k} = P_{a_k} = \frac{1}{2}\rho_{cr_1} V_k^3 S C_{D_k} + \frac{m_k\left(V_{k+1}^2 - V_k^2\right)}{\Delta t_k}. \tag{8}$$

In Eq. 8 the nodal values of the airspeed are set by the optimizer, which therefore assigns the speed profile over the cruise according to a range-optimal seek. Brake power and fuel flow are obtained similarly to the climb phase (see Eq. 5 and corresponding comments). The time corresponding to each discretized segment can be computed as $\Delta t_k = \frac{F_k}{\dot{F}_k}$, therefore the range corresponding to the first cruise leg is

$$\mathcal{R}_1 = \sum_{k=1}^{n_{cruise_1}} V_k \cdot \Delta t_k. \tag{9}$$

Clearly, the weight of the aircraft impacting Eq. 8 is updated on account of the loss F_k pertaining to each segment, similar to climb. The computation of the return cruise follows exactly the same procedure, but is based on a different initial mass and altitude.

2.2 Optimal Problem

The equations introduced in the previous subsection are structured so as to allow the computation of the range of the two cruising legs (outbound and return), namely $\mathcal{R}_1$ and $\mathcal{R}_2$, based on the assignment of the cruising altitudes h_{cr1} and h_{cr2}, the arrays of airspeeds $\boldsymbol{V}_{cr1}$ and $\boldsymbol{V}_{cr2}$, featuring respectively n_{cruise_1} and n_{cruise_2} elements, and the fuel ratio parameter η_f. All other parameters, including initial weight and altitude, the aerodynamic polar and the power-train specifications are assigned constants. The minimization of range can be performed by writing the optimal problem as

$$\min_{\boldsymbol{p}} \left(- \left(\mathcal{R}_1^2 + \mathcal{R}_2^2 \right) \right), \; s.t. \, \boldsymbol{q} \tag{10}$$

where the set of optimization variables $\boldsymbol{p} = \left(h_{cr1}, h_{cr2}, \boldsymbol{V}_{cr1}, \boldsymbol{V}_{cr2}, \eta_f \right)$, and $\boldsymbol{q}$ is a set of constraints. The latter is specified to assure that the ranges of the outbound and return legs are equal, which reflects the structure of the mission, and that the power required keeps within the limits of the assigned engine, yielding

$$\begin{aligned} q_1 &: |\mathcal{R}_1 - \mathcal{R}_2| \leq \text{tol} \\ q_{2,k} &: P_{\min} \leq P_{b_k} \leq P_{\max}, k = 1, \ldots, n_{cruise_1} + n_{cruise_2} \end{aligned} \tag{11}$$

The optimal problem in Eq. 10 has been solved in the present work making use of a gradient-based algorithm, on account of a good regularity of the cost function and constraints.

2.3 Application: Optimal Mission Profile for Test-Bed

Fed with the characteristics of the G70, the optimal problem in Eq. 10 with constraints 11 has been solved. In particular, the engine and polar of the aircraft are known, and the tolerance in q_1 has been set to 1 km. The resulting optimal mission is reported in Fig. 2, left plot. The optimal altitudes for the outbound and return cruises are h_{cr1} = 1'320 m and h_{cr2} = 1'640 m respectively, whereas parameter η_f = 0.58 in fuel balance equation Eq. 7. The distance from the airport to the cargo drop point is close to $\mathcal{R}_1$ = 400 km, which is therefore compliant with the requirements specified at the beginning of section 2. The optimal outbound range result complies with the requirement for a minimum target range of 200 km previously specified. Actually, the margin with respect to the specification may be employed to fly missions according to a sub-optimal profile. The adoption of the latter - for instance an altitude different from the optimal one - may result from specific on-the-day flight conditions (e.g. forest fires, risk of flight interdiction, etc.).

In the scope of a retrofit design, it is interesting to perform a sensitivity analysis of the optimal range vs. payload, in view of the ability of the selected test-bed to sustain a significant maximum normal load factor (see Tab. 1). In case the top load factor is reduced, a larger weight can be loaded, thus strongly increasing the payload capacity of the retrofit. The right plot on Fig. 2 displays the outcome of such analysis, where each point on the plot has been obtained as the result of an optimal mission design. The take-off weight of the aircraft has been computed based on the scaling law $m_{TO}(n_{\max}) = m_{TO,\text{design}} \frac{n_{\max,\text{design}}}{n_{\max}}$, for different values of the maximum assumed load factor $n_{\max}$ between 2.5 and 4 (with $n_{\max,\text{design}} = 4$ as per Tab. 1). Clearly, the results corresponding to higher values of $n_{\max}$ assumed in this sensitivity analysis have a limited significance, since for values of $n_{\max}$ too much above the original $n_{\max,\text{design}}$ value, the corresponding increase in m_{TO} (according to the law binding $n_{\max}$ to m_{TO} just introduced) would require to redesign other parts of the aircraft (in particular the landing gear), and also an increase in installed power, especially to grant satisfaction of take-off requirements. These effects in turn would imply a further increase in m_{TO}, making the results for higher $n_{\max}$ of partial practical validity, as said. However, the right plot in Fig. 2 is interesting for showing the expected trade-off between range and payload for an assigned value of $n_{\max}$, as well as the potential of the optimally-oriented approach adopted as a numerical tool to find the range of the retrofitted aircraft, as anticipated in the introduction to section 2.

3 TECHNOLOGICAL PROBLEMS IN RETROFITTING

The choice of an aircraft based on a metal tubes frame and skin is a key enabler of the retrofitting process. However, modifications in two major areas are required in view of a conversion for unmanned use, namely

1. the manual control chain of command needs to be converted, installing servo-actuators for all control axes. Sizing servo-actuators is now required, and an autopilot needs to be installed as well.
2. the inside of the cabin needs a conversion to host pallets, taking over as much volume as possible (profiting also from the suppression of the instrument panel and mechanical control levers), but without altering the load-bearing structure, keeping the weight and CG excursion within the same limits prescribed for equilibrium performance and stability, and allowing unimpeded pallet dropping.

The two issues and corresponding retrofitting methodologies will be explained in the following paragraphs.

3.1 Control Chain

In view of the adoption of an electronic flight controller, the control chain of the elevator, rudder and ailerons can be suppressed and substituted with a set of electric-mechanical servo-actuators. The actuation scheme for each of the these surfaces is reproduced in Fig. 3. The choice is suggested by the use of the same technology for flap actuation on the selected model. Here the electric torque M_{el}, produced by a torque generator of fixed length b_1 acting between the two hinges to the left of the scheme, is applied on an anchor point on the airframe (the leftmost in the picture). The hinge in point $\boldsymbol{H}$ is similarly attached to the airframe, but the control surface is pivoting around it. The control surface is attached to arm b_2, whereas the actuator physically takes over arm b_1 as said, and is responsible for imparting the electric torque M_{el}.

Due to the choice of the geometry in Fig. 3, the following kinematic equivalences apply

$$\begin{cases} b_1 \cos\left(\frac{3}{2}\pi + \omega_1\right) + l_1 \cos(-\theta) = l_2 \cos(-\gamma) + b_2 \cos\left(\frac{3}{2}\pi + \omega_2\right) \\ b_1 \sin\left(\frac{3}{2}\pi + \omega_1\right) + l_1 \sin(-\theta) = l_2 \sin(-\gamma) + b_2 \sin\left(\frac{3}{2}\pi + \omega_2\right) \end{cases}, \tag{12}$$

respectively in the horizontal and vertical directions on the sketch. In Eq. 12, the free parameters in a design phase are the length of the rods (b_1, b_2 and l_1), the relative position of the two fixed joints (described by l_2 and γ) and the reference values of the angular coordinates ω_1 and ω_2. The kinematics in Eq. 12 can be evaluated in specific conditions, thus introducing some design constraints. In particular, according to the range of rotation of the actuator and of the control surface to be rotated, ranges for ω_1 and ω_2 can be specified, matching extreme values. In analytical terms, considering the maximum deflections of the control surface achievable in both directions, this bears

$$\begin{cases} b_1 \sin\left(\omega_{1_{\min|\max}}\right) + l_1 \cos\left(\theta_{\min|\max}\right) = l_2 \cos(\gamma) + b_2 \sin\left(\omega_{2_{\min|\max}}\right) \\ b_1 \cos\left(\omega_{1_{\min|\max}}\right) + l_1 \sin\left(\theta_{\min|\max}\right) = l_2 \sin(\gamma) + b_2 \cos\left(\omega_{2_{\min|\max}}\right) \end{cases}. \tag{13}$$

Considering Eq. 13, a system of 4 equations has been written, in the 6 unknowns b_2, l_1, l_2, γ, $\theta_{\min}$ and $\theta_{\max}$. Actually, the value of b_1 cannot be considered as a free design variable, since it must cope with the physical length of the actuator, and is therefore assigned. Since the system is not determined, it can be solved imposing further conditions. The satisfaction of an optimality condition is selected, explained in the following.

Optimal sizing of the control chain

Considering an assigned set of b_2, l_1, l_2, γ, it is possible to express both θ and ω_2 as functions of ω_1, respectively $\theta = \theta(\omega_1)$, $\omega_2 = \omega_2(\omega_1)$, exploiting Eq. 12. Similarly, the moment arms a_1 and a_2 in Fig. 3

may be expressed as functions of ω_1, as $a_1 = a_1(\omega_1)$, $a_2 = a_2(\omega_1)$. Such moment arms can be employed to get an expression of the moment $M_{\boldsymbol{H}}$ transferred to the hinge of the control surface from the actuator, yielding

$$M_{el} = \frac{a_1}{a_2} M_{\boldsymbol{H}}. \tag{14}$$

Now, in Eq. 14 the reaction of the two arms is a function of the value of ω_1, and the variability of that quantity is a function of the geometry, as just explained. Therefore, an optimal sizing problem can be configured, where the measure to be penalized tries to capture that variability, bound to the extreme excursion of the arms ratio, yielding

$$J = \left(\min\left(\frac{a_1(\omega_1)}{a_2(\omega_1)}\right) - \max\left(\frac{a_1(\omega_1)}{a_2(\omega_1)}\right)\right)^2, \tag{15}$$

which produces the associated optimal problem

$$\min_{\boldsymbol{g}} J,\ s.t.\ \boldsymbol{s} \tag{16}$$

where $\boldsymbol{g} = (b_2, l_1, l_2, \gamma, \theta_{min}, \theta_{max})$, and the set of constraints $\boldsymbol{s}$ is represented by Eq. 13.

3.2 Application: Re-Sizing of the Control Mechanisms and Actuators

A numerical implementation of the problem in Eq. 16 is carried out computing the functional in Eq. 15 over a discretized domain of ω_1 between assigned extreme values $\omega_{1_{\min}}$ and $\omega_{1_{\max}}$. This is treated via a gradient-based algorithm, accounting for a set of equality constraints from Eq. 13 (as explained). For the G70, the design algorithm is applied to the elevator, rudder and ailerons. Table 2 summarizes the sizing results (partly in normalized form for secrecy).

Assigned values of the length b_1 have been specified for the three control surfaces, according to the sizing of candidate existing actuators (see Tab. 4, explained in more detail later). Similarly, the extreme values of the deflections have been assigned, according to the range of motion of the actuator and of each of the moving surfaces, thus assigning the parameters $\omega_{1_{\min}}$, $\omega_{1_{\max}}$ and $\omega_{2_{\min}}$, $\omega_{2_{\max}}$ needed for the optimization (the corresponding values cannot be disclosed). As an example, the geometry of the actuator for elevator deflection is shown in Fig. 4.

With an assigned geometry for each of the control surfaces, the choice of a corresponding actuator should be carried out based on an evaluation of the hinge moment produced by the surface, and compliant with the outcome of geometrical sizing. In order to estimate the hinge moment on the elevator, rudder and ailerons, a standard linear representation of the hinge moment coefficient has been assumed, where the hinge moment at $\boldsymbol{H}$, C_{M_H}, is expressed as

$$C_{M_H} = C_{M_{H_\sigma}}\sigma + C_{M_{H_\delta}}\delta + C_{M_{H_0}}, \tag{17}$$

where σ represents the angle of attack for the elevator and ailerons, and the sideslip angle for the rudder, whereas the control variable is that corresponding to the specific control surface. In order to estimate the three coefficients in Eq. 17 for each of the three control surfaces, two methods have been applied, namely a semi-empirical method based on regressions (Roskam 2004) and a numerical method, based on an inviscid computation (Drela 1989). Both need as an input the sizing of the mean chord of the corresponding assembly (i.e. the horizontal tail for the elevator, vertical tail for the rudder and wing for the ailerons), the mean chord of the deflectable control surface and its span, as well as a representative aerodynamic profile and its two-dimensional properties. These properties, as well as maximum and minimum deflections, are shown in Tab. 3 (in normalized form and except airfoil identity due to secrecy).

The outcome of the estimation of the hinge coefficients is employed to compute the hinge moments for three airspeed settings, obtained from the flight manual and corresponding to the maximum speed values for a deflection of the corresponding control surface. These are $V_{FE} = 120$ km/h for the rudder, and $V_A = 150$ km/h for the elevator and ailerons. Based on the estimation of the hinge moments and of the optimal kinematic sizing, three compatible candidate servo-actuators are proposed, with characteristics listed in Tab. 4. Elevator actuators are two, operating in parallel on the two lateral halves of the horizontal tail respectively.

The compliance of the selected actuators with respect to the requirements is demonstrated by the comparisons in Fig. 5, where the hinge moments produced by the actuators in Tab. 4 through the kinematics previously designed are obtained exploiting Eq. 14, and compared to the estimation model in Eq. 17 for a series of deflections of the control between minimum and maximum. The plots of Fig. 5 refer to the airspeed $V_A = 150$ km/h, for an easier comparison among cases. It can be observed that the rated hinge moment from the selected actuators and control mechanisms is always above the expected value, except for extreme deflections of the rudder (top-right plot). The latter is not surprising, since for compliance with design limits the rudder deflection should not reach the extreme mechanically achievable values at V_A, but only at the inferior V_{FE} airspeed.

3.3 Payload Bay Re-Design

As pointed out in the introduction to the current section, a major aspect in the process of passing from a manned aircraft to an UAV in this size is the redesign of the payload bay, i.e. the cabin. The plants and corresponding masses which are deleted in the UAV aircraft are the mechanical control chain for all moving surfaces, which are aluminum tubes and steel cables in the considered testbed, as well as the pilot's and passenger's seat and instrument panel. As explained in the previous section, it is possible to replace the control chain with actuators mounted in proximity of the control surfaces. These bring in corresponding mass components. Table 5 displays the normalized positions of the masses taken out or added to the aircraft (mass values for aircraft parts cannot be disclosed).

From the last line of Tab. 5, the overall effect on mass distribution, according to the configuration of the aircraft and the placement of the components added or removed, is equivalent to subtracting a total mass of $\Delta m = 15.4$ kg from a station located 1.59 chords upstream of the wing leading edge.

According to this computation, and from mass data for the specific test-bed in the original manned version, the total mass of the payload for the UAV version of the LSA can be estimated at $m_{PL} = 245.9$ kg. The moderate change with respect to the original payload and loading configuration are compatible with the original structural design, allowing to avoid any redesign of the load bearing structure.

A redesign of the cabin accounting for the room gained from the deletion of the interiors (seats, control commands and panel) has been carried out in two configurations, considering two major conceptual inputs:

1. for assuring ease of conversion and reducing design cost, the structure of the aircraft should not be altered. This implies keeping the load-bearing strut between the legs of the main undercarriage, which ideally splits the pavement of the passenger bay in two longitudinal sections
2. the payload needs to be parachute dropped according to the requirements of the medical transport mission, thus implying the need to put the payload in pallets, to be released through the pavement of the cargo bay

According to these two drivers, in a first configuration (configuration A), the payload is arranged in two pallets as shown on the left plot in Fig. 6. The size of the back pallet (#1) allows for a volume $V_{PL,1} = 45$ lt, whereas that of the cut-trapezoid forward one (#2) is $V_{PL,2} = 49$ lt. Assuming for the payload a uniform nominal density and the overall mass m_{PL} just computed, the mass for each of the two pallets will be proportional to the volume. Inertial values and the respective centers of gravity positions are reported in Tab. 6.

The second cargo configuration (configuration B) accounts for a small range-extending fuel tank. This is placed in top position, on account of the gravity-based feeding of the engine (no fuel pumps in the tanks). This configuration allows to further simplify the manufacturing of the two cargo pallets, which take a more basic parallelepiped form. The corresponding cargo and fuel arrangement is reported on the right plot of Fig. 6. The geometric and inertial characteristics of the two pallets (#1, #2) and auxiliary tank are reported in Tab. 7. Considering the density of MOGAS fuel, the mass stored in the range extending tank is $m_{RE} = 49.5$ kg.

It should be noted that the position and size of pallet #1 are the same in both considered configurations A and B.

The parcelling of the cargo is potentially critical for longitudinal static stability. For assuring the aircraft is inherently stable in the longitudinal sense in any phase of the flight, including the cargo dropping phase, an analysis of the center of gravity position ξ_G and of the static margin in all cargo loading conditions has been carried out in both configurations A and B. The results are reported in Tab. 8.

It can be observed from Tab. 8 that for configuration B (with range extender) instability is encountered in case pallet #1 is dropped before pallet #2. This defines the dropping sequence, which needs to be #2 (forward pallet) first and #1 last.

4 UNMANNED MISSION EXECUTION: DYNAMICS, CONTROL AND LAUNCH PRECISION

Following re-design, a simulated analysis of the mission is carried out, to two major aims:

1. assessing the required features for a flight control system, capable of autonomously controlling the flight of the retrofitted UAV, including the cargo dropping phase, coping with the sudden and significant change in the inertial features of the aircraft.
2. forecasting the characteristics of the drop phase, including the achievable precision of the launch

Concerning the first point, a simulator for the dynamics of the aircraft in the longitudinal plane is sufficient for the task. For the second, the dynamics of the aircraft are flanked by those of the parachute and cargo, for which a standard model will be recalled and employed. The two points are treated in the next subsections, leading to a final assessment of the suitability of the proposed aircraft for the intended mission, provided the modifications defined in this work are adopted.

4.1 Flight Control System and Cargo Dropping

A significant perturbation in the longitudinal plane is expected as a result of cargo dropping, due to the sudden change in the position of the center of gravity, and the ensuing alteration in moment balance. In order to study this effect, a model for longitudinal dynamics has been set-up, based on available data for the selected test-bed. In particular, as explained also in section 2.1, the aerodynamic polar (modeled as a parabolic function through coefficients C_{D_0} and K) and the lift curve (in linear form, assigned through coefficients C_{L_0}, C_{L_α}, $C_{L_{\delta_e}}$, α_0, δ_{e_0}) are assigned, as well as the position of the neutral and control points ξ_N and ξ_C. The coefficients $C_{M_{G_\alpha}}$, $C_{M_{G_{\delta_e}}}$ and $C_{M_{G_0}}$ in the expression of the barycentric aerodynamic moment C_{M_G}, modeled as linear with α and δ_e, are functions of the position of the center of gravity ξ_G, computed based on the results of section 3.3 for the proposed cargo configurations, either with or without range extender, as per flight mechanics definitions (Pamadi 1998)

$$C_{M_{G_\alpha}} = -\left(\xi_N - \xi_G\right) C_{L_\alpha}, \quad C_{M_{G_{\delta_e}}} = -\left(\xi_C - \xi_G\right) C_{L_{\delta_e}}, \quad C_{M_{G_0}} = -\left(C_{M_{G_\alpha}} \alpha_0 + C_{M_{G_{\delta_e}}} \delta_{e_0}\right). \qquad (18)$$

Table 9 shows the effect on the moment coefficient components (normalized with respect to the absolute value $|C_{M_\alpha}|$ for the pre-drop configuration without range extender) in Eq. 18, before cargo dropping (stage 1), after the first pallet drop (stage 2) and after the second drop (stage 3).

The system is trimmed in static equilibrium, solving the static trim problem in Eq. 19

$$\begin{cases} L + T \sin\alpha - mg = 0 \\ T \cos\alpha - D = 0 \\ M_G = 0 \end{cases} \qquad (19)$$

for all stages and configurations, yielding the results in Tab. 10. The flight condition considered for trim is that assumed for the airdrop, i.e. an airspeed of 25 m/s and an altitude of 300 m (more on this in section 4.2).

The equilibria for stages 1, 2 and 3 in the dropping phase correspond to three different inertial characteristics, as explained, due to a motion of the center of gravity and a change in mass and pitch inertia. A control system capable of dealing with the transient is designed, to reduce the potentially severe oscillations or divergence triggered by the drops between phase 1 and 2, or 2 and 3. The longitudinal dynamics of the system are modeled via a standard non-linear representation in the longitudinal plane, based on four scalar equations - two equations for momentum balance (along the trajectory and normal to it), one for moment of momentum balance, and a kinematic relationship for rotational rates (Pamadi 1998). These are reported in

Eq. 20

$$\begin{cases} \dot{V} = \dfrac{F_{x_w}}{m} - g\sin\gamma \\ \dot{\alpha} = \dfrac{F_{z_w}}{mV} + q + \dfrac{g}{V}\cos\gamma \\ \dot{q} = \dfrac{M_{G_{y_b}}}{I_{yy}} \\ \dot{\gamma} = q - \dot{\alpha} \end{cases} \tag{20}$$

where F_{x_w}, F_{z_w} are the components along the wind frame first (x) and third (z) axis of the aircraft of aerodynamic and propulsion force, $M_{G_{y_b}}$ is the pitching moment due to aerodynamics and propulsion in the center of gravity, α the angle of attack, γ the climb angle, q the pitch rate.

Control system for longitudinal flight dynamics

The proposed control system is thought to make use of a minimal set of measurements, as recommendable for the specific, low-budget and low-technology architecture. A double loop parallel architecture is envisaged.

A first SISO controller measures the error between the reference and current value of the angle of attack α, and targets it with a deflection of the elevator δ_e. This is implemented as a PID regulator, therefore requiring the integration and differentiation of α. This is typical to most longitudinal SAS architectures.

The second loop takes a MISO structure, which makes use of the thrust setting δ_t as a control, and takes as inputs the airspeed V, vertical speed $V\sin\gamma$ and altitude h. In particular, the regulator makes use of the integrals (i.e. I-type law) of airspeed and altitude, and implements a PI law on the vertical speed. Tuning of the control law has been carried out via a trial-and-error procedure, iteratively testing the control system over the drop maneuver, making use of the dynamics in Eq. 20 and including the dynamics of the actuators in Tab. 4, which act as second order systems. The system is numerically integrated with a Runge-Kutta scheme. The result of an example tuning are shown in Fig. 7 in terms of elevator control, angle of attack, pitch rate and normal load factor, for the case without range extender (qualitatively very similar results are obtained with range extender tank). The response in Fig. 7 is deemed satisfactory, since that the elevator deflection δ_e is always largely between excursion limits, the angle of attack α keeps safely below stall values, and the normal load factor tops largely below the maximum allowable for this aircraft. Figure 8 shows the effect of the same tuning of Fig. 7 on further quantities, namely thrust setting, airspeed, altitude and engine RPM. It can be observed that the altitude is the most sensitive to the drop, as expected. However, the controller is capable of keeping it within a 15 m boundary from the target value. The airspeed is visually oscillating, but the actual amplitude values are limited under 1 m/s. Furthermore, the rotation of the engine changes smoothly and without non-physical oscillations.

4.2 Cargo Parachute Drop Maneuver: Simulation and Precision Assessment

In order to assess the suitability of the proposed aircraft for the parachute cargo drop in terms of achievable precision, given the speed, altitude and mass characteristics of the aircraft and cargo, the parachute drop dynamics is accurately simulated.

The adopted model is that of a single-riser parachute (Guglieri 2012). This is an approximation, since the parachute is typically not made with a single riser. However, this model is suitable for the scope of the analysis, mostly centered on cargo dynamics. It assumes that the parachute is always aligned with the airspeed, and treats it like a drag force generator. Since the latter is not applied to the center of gravity of the cargo box, also a barycentric moment will be introduced in the dynamics of the cargo. The (single) riser connecting the box to the parachute is modelled as a spring-damper system. The analytic scheme considered for the implementation is shown in Fig. 9. A preliminary sizing of the parachute has been carried out based on a static model, where parachute drag equals the weight of the parachute and cargo load. According to the requirement for a maximum terminal speed of $V_t = 3$ m/s, compatible with the type of payload, and considering a nominal parachute drag coefficient of $C_D = 1.5$ (Gelito et al. 2006), the area of the parachute has been estimated at $S = 142$ m^2. Correspondingly, a Mills G-14 commercially available parachute has been selected, which is slightly larger than required, and designed for a cargo mass of $m_p = 226$ kg. It features a mass of 16 kg, which is cut from the payload mass previously estimated for each pallet, thus not adding to the overall weight of the retrofitted aircraft. Furthermore, the parachute is designed for a drop from a minimum altitude of h= 300 m, which is therefore assumed as the target flight altitude for the drop phase.

The equations governing the motion of the parachute and pallet system are reported in Eq. 21 (Guglieri

2012).

$$\left\{\begin{aligned} \dot{V}_p &= \frac{F_r - D_p - mg\sin\gamma}{m_p + m_a} \\ \dot{u} &= \frac{F_b(1)}{m} - qw + rv \\ \dot{v} &= \frac{F_b(2)}{m} - ru + pw \\ \dot{w} &= \frac{F_b(3)}{m} + qu - pv \\ \dot{p} &= \frac{\left(I_{yy} - I_{zz}\right)qr + M_b(1)}{I_{xx}} \\ \dot{q} &= \frac{\left(I_{zz} - I_{xx}\right)pr + M_b(2)}{I_{yy}} \\ \dot{r} &= \frac{\left(I_{xx} - I_{yy}\right)pq + M_b(3)}{I_{zz}} \end{aligned}\right. \quad , \quad \left\{\begin{aligned} &\begin{bmatrix} \dot{q}_0 \\ \dot{q}_1 \\ \dot{q}_2 \\ \dot{q}_3 \end{bmatrix} = \frac{1}{2}\begin{bmatrix} 0 & -p & -q & -r \\ p & 0 & r & -q \\ q & -r & 0 & p \\ r & q & -p & 0 \end{bmatrix}\begin{bmatrix} q_0 \\ q_1 \\ q_2 \\ q_3 \end{bmatrix} \\ &\dot{s}_p = V_p \\ &\dot{x} = u \\ &\dot{y} = v \\ &\dot{z} = w \end{aligned}\right. \tag{21}$$

The system to the left in Eq. 21 represents the dynamic balance. The first equation governs the motion of the reference point $\boldsymbol{G}_P$ of the parachute, associated to airspeed V_p, where the term $m_p + m_a$ represents the sum of the mass of the parachute and dragged air respectively. The latter has been estimated as a function of the porosity of the parachute, assumed at $\bar{p} = 0.2$ (Maydew and Peterson 1991). The parachute deployment transient is accounted for by assigning a time evolution of the mass of dragged air, going from null to the nominal value according to a smooth cubic interpolation, over a time frame estimated at 13 s (Cockrell 1987). Forces F_r and D_p represent the riser tension and parachute drag respectively. The next six equations model the translation and rotation dynamics of the pallet in its body frame. Components u, v, w and p, q, r pertain to the speed and rotation rate of the pallet in the body frame $(\cdot)_b$ in Fig. 9. Here the force components F_{b_1}, F_{b_2} and F_{b_3} are the resultants of aerodynamic force, riser force and gravity (i.e. all active forces). Correspondingly, components M_{b_1}, M_{b_2} and M_{b_3} pertain to the resultant moment, measured in the pallet center of gravity. The mass of the pallet is m, and the components I_{xx}, I_{yy} and I_{zz} pertain to the barycentric tensor of inertia of the pallet (it is assumed that the non-diagonal terms are not relevant, in consideration of the symmetries in the pallet construction).

The system to the right of Eq. 21 introduces kinematic relationships. The first four are rates of quaternion components q_0, q_1, q_2 and q_3, as functions of the body components of the rotational speed of the pallet p, q and r. The fifth equation refers to the parachute, where it is assumed (as said) that the airspeed is equal to the time rate of the position of the reference point $\boldsymbol{G}_p$. The last three relate the components of translational speed of the center of gravity of the pallet and the time rates of position.

In modeling the aerodynamic force acting on the system, a model for the wind was included, since a significant impact of the actual wind condition on the evolution of the trajectory of the parachuted pallet during the descent is expected. The wind field has been modeled through the superimposition of a deterministic power-law, representing the average ground boundary layer, and a stochastic component. The former is based on the equation

$$V_w(h) = V_{w_{h_0}} \cdot \left(\frac{h}{h_0}\right)^{\alpha_w} \tag{22}$$

where h is the altitude from ground, h_0 the airdrop altitude, and $V_{w_{h_0}}$ the corresponding speed of the wind. The power law exponent α_w depends on the type of terrain, and assuming low-rise vegetation as acceptable for a dropping area, a value of 0.14 is assumed (Ray et al. 2006). It is noteworthy that the wind vector in Eq. 22 has components only in the horizontal plane. The stochastic component is added according to the Dryden atmospheric turbulence model.

Drop simulation and precision assessment

The system in Eq. 21 can be integrated with a Runge-Kutta scheme. For simplicity, only the first pallet is considered for a precision assessment of the airdrop. No significant qualitative difference is expected from the outcome of the airdrop of the second part of the cargo. The initial conditions for the simulation are set as the flight conditions. In particular, the aircraft is assumed to be flying north (assumed as the orientation of the x axis in the navigational frame). A Monte-Carlo analysis is carried out where the descent of the first pallet is simulated, considering 300 combinations of parameters. Four parameters have been selected in this study. The first is $C_D S_p$, i.e. the drag coefficient times the area of the parachute, representing a shape parameter of the latter. The second is $V_{w_{h_0}}$ (see Eq. 22). The third is the experimental parameter k_{a_0}, which modulates the value of the dragged air m_a (Maydew and Peterson 1991). The last one is the misalignment ψ between the direction of flight (north) and the direction of the average wind at altitude h_0. Table 11 displays the average and standard deviation of the parameters, for which a Gaussian distribution model is assumed.

Figure 10 displays to the left an example integrated trajectory of the cargo pallet, and to the right the outcome of the Monte-Carlo analysis. The rectangle on the right plot defines a 2σ range for both the flight (x axis) and cross-flight direction (y axis). The x-by-y size of the rectangle under the assumed trial conditions is 102-by-57 m, which is compatible with the intended practical mission purpose, since collecting the pallet from an area of that size should not impose an unacceptable pick-up burden. To better understand the most sensitive drivers potentially impacting the landing precision performance, a correlation analysis has been performed, determining the most intensely influencing factors (among the four considered in the analysis) on three relevant performance indices, i.e. the coordinates of the landing point, x_{lnd} and y_{lnd}, as well as the top value of the riser force $F_{R_{\max}}$. The latter is inherently bound to the top acceleration sustained by the cargo pallet in flight. Under the considered trial scenario, it has been determined that

1. x_{lnd} is much influenced by $V_{w_{h_0}}$, and less significantly by $C_D S_p$
2. y_{lnd} is influenced by the wind misalignment ψ, by a significant extent
3. $F_{R_{\max}}$ is influenced by $V_{w_{h_0}}$

Correspondingly, Fig. 11 graphically displays the correlations. It can be noticed that the wind direction ψ and intensity $V_{w_{h_0}}$ are relevant drivers in enabling a better landing precision.

These quantities can be estimated according to a GPS-PEC technique (D'Aniello 2021), which besides an average GPS tracker and Pitot vane, calls for one or more flight circuits around the cargo landing target point. However, this is not incompatible with the budget and technology level of the considered retrofit, especially in view of the need to mount an autopilot to make the design unmanned.

4.3 Choice of autopilot

Based on the outcome of the analysis, a suitable autopilot can be selected, considering that it should allow a remote control feature, so that the manual take-off and landing can be performed as per specification, and it should feature an option to define custom control laws with specifically designed software to manage the behaviour of the aircraft as well as the payload throughout the mission. Table 12 lists the main characteristics of shortlisted autopilots.

All feature redundancy, with the Veronte 4X being the safest, thanks to three complete autopilot cores, plus one dissimilar arbiter board. Sense & Avoid functions are also possible by installing an obstacle data source, such as ADS-B, a radar or a LIDAR sensor. However, the Micropilots assures compatibility with Volz servos, and is therefore recommendable for reducing complexity and cost in the retrofitting process.

5 CONCLUSIONS

This paper investigates the feasibility of the retrofitting of an existing LSA aircraft for an unmanned, cargo transport mission. Based on a set of specifications formulated together with Médecines Sans Frontières, an optimal mission profile has been obtained, complying with the need to parachute-drop an assigned load on a target point and return to the origin. This was employed on a test-bed, shown according to the compliance with the mission requirements (range, payload) and ease of transformability (traditional metal construction), further showing its applicability for the mission in terms of flight performance.

The modifications required for the retrofitting process have been identified in three major areas, namely the redesign of the command chain, the redesign of the cargo bay, and the implementation of an automatic control system. Correspondingly, servo-actuators requirements have been studied through a sizing problem and proposed for all control surfaces. Two re-design option of the cargo bay have been accurately formulated, carefully assessing the effects on longitudinal inertia (and therefore static stability and dynamic performance), on account of the actual size of the test-bed. A flight control system have been envisaged, implemented and tested in virtual environment on a dynamic model of the system, capable of managing the target mission profile.

Finally, the mission has been simulated, including the parachute drop of the cargo, trying to assess the precision of the launch outcome, and showing that through a suitable selection of the launch parameters (airspeed, altitude) totally compatible with usual flight of the selected test-bed, an acceptable accuracy can be obtained, also in presence of wind, thus showing that the proposed retrofit might meet the specifications for the mission. The analysis of the flight control system, and more generally of the mission requirements, have allowed to investigate the autopilot suite to put on board.

Beside a retrofitting methodology, the paper shows for a specific case study that a retrofit of the proposed test-bed would well cover the mission at hand. Therefore, retrofitting would be for this type of mission a cost-effective way of recycling existing designs, without the need to start a new design from scratch, which due to cost and the limited size of the targeted market, may starkly reduce the interest of manufacturers.

The work might be further developed assessing the cost of the retrofitting process, for which models are under investigation with the collaborating Company manufacturing the test-bed.

DATA AVAILABILITY STATEMENT

Some or all data, models, or code that support the findings of this study are available from the corresponding author upon reasonable request.

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FIGURE CAPTIONS

Fig. 1. Portrait of Groppo G70.

Fig. 2. Flight mechanics variables in optimal mission profile (left). Payload-range diagram for optimal missions with a changing value of $n_{\max}$ (right).

Fig. 3. Actuator and control surface kinematics.

Fig. 4. Arrangement and sizing (normalized) of the elevator actuator. Maximum negative deflection.

Fig. 5. Actuator torque vs. hinge moment for the range of deflections. Top-left: elevator. Top-right: rudder. Bottom: ailerons. Moments measured in Nm.

Fig. 6. The cargo bay of the aircraft in configurations A (left) and B (right), the latter with range-extending tank.

Fig. 7. Fast dynamics parameters before and after the first airdrop (stage 1 to 2, on the left) and the second airdrop (stage 2 to 3, on the right), without range extender.

Fig. 8. Slow dynamics parameters during the airdrop operation, without range extender.

Fig. 9. Dynamics of the single-riser parachute with cargo pallet.

Fig. 10. Example landing trajectory (left) and landing points resulting from Monte-Carlo analysis (right).

Fig. 11. Distribution of results and trend lines of the analyzed quantities with respect to the most relevant variable parameters.

TABLES

TABLE 1. Groppo G70 basic specifications.

Parameter	Value	Unit
Wingspan	8.9	m
Length	6.22	m
Width	2.74	m
Wing surface	10.56	m^2
Cabin width	1.22	m
Tank capacity	2x50	l
Empty mass	297.5	kg
Maximum project mass	600	kg
V_{FE}	110	km/h
V_{NE}	210	km/h
V_A	150	km/h
V_{SO}	65	km/h
V_S	71	km/h
Max/Min normal load factor	+4/-2	g

TABLE 2. Results of kinematic optimization for each control surface (partly normalized).

Parameter	Elevator	Rudder	Aileron
l_1/l_1	1.0	1.0	1.0
l_2/l_1	0.94	0.96	0.94
b_2/l_1	0.32	0.25	0.36
γ	8.97	10.1	7.58

TABLE 3. Geometric characteristics of control surfaces (normalized over either mean chord or maximum deflection).

Parameter	Elevator	Rudder	Aileron
Mean chord assembly	1.0	1.0	1.0
Mean chord control surface	0.43	0.46	0.20
Span control surface	3.34	1.01	1.18
δ_{max} control surface	1.0	1.0	1.0
δ_{min} control surface	-1.15	-1.0	-1.9

TABLE 4. Characteristics of selected actuators.

Parameter	Elevator	Rudder	Aileron
EMA model	2 x DA 30-HT-Duplex	DA 30-HT-Duplex	DA 26 Duplex
Travel angle [°]	±45	±45	±45
Rated torque [Nm]	2 x 36	36	5
Peak torque [Nm]	2 x 64	64	12
Actuator mass [kg]	2 x 1.85	1.85	0.64
Rated speed [°/s]	115	115	170
Length (b_1) [mm]	24	24	20

TABLE 5. Mass and positioning changes, normalized with respect to mean aerodynamic chord and measured with respect to the wing leading edge, positive downstream.

Element	Mass variation [kg]	CoG position
Control chain	-(*)	0.066
Seats	-(*)	0.397
Instrumentation	-(*)	-0.4
Elevator EMAs	+3.70	3.75
Rudder EMAs	+1.85	3.75
Rudder EMAs	+1.28	0.8
Total	-15.37	-1.59

TABLE 6. Geometry and inertia of pallets in configuration A.

Parameter	Pallet #1	Pallet #2
Volume [m^3]	0.45	0.49
m_{PL} components [kg]	117	128 kg
Center of gravity position	0.675	0.025
I_{yy} [kgm^2]	11.72	13.46

TABLE 7. Geometry and inertia of pallets in configuration B.

Parameter	Pallet #1	Pallet #2	Fuel tank
Volume [m^3]	0.45	0.39	0.069
Mass [kg]	105	90	49.5
Center of gravity position	0.675	0.013	0.203
I_{yy} [kgm^2]	10.50	9.57	3.40

TABLE 8. Payload configuration and position of the center of gravity.

Case	ξ_G		Compliance with limits
	Configuration A	Configuration B	
With pallet #1 and #2	0.2796	0.2711	Yes
Without pallet #1	0.2108	0.2069	No
Without pallet #2	0.3502	0.3330	Yes
Without pallet #1 and #2	0.2799	0.2663	Yes

TABLE 9. Values (normalized) of the coefficients in the linear expression for barycentric aerodynamic moment, for all stages in the airdropping procedure.

Description	Symbol	Value		Unit
		Without RE	With RE	
Before drop	$C_{M_{G_{\alpha_1}}}$	-1	-1.052	
	$C_{M_{G_{\delta_{e_1}}}}$	-0.9115	-0.9148	
	$C_{M_{G_{0_1}}}$	-0.0155	-0.0206	rad
After drop of first pallet	$C_{M_{G_{\alpha_2}}}$	-0.6475	-0.7370	
	$C_{M_{G_{\delta_{e_2}}}}$	-0.8899	-0.8953	
	$C_{M_{G_{0_2}}}$	0.0187	0.0100	rad
After drop of second pallet	$C_{M_{G_{\alpha_3}}}$	-1	-1.0737	
	$C_{M_{G_{\delta_{e_3}}}}$	-0.9115	-0.9161	
	$C_{M_{G_{0_3}}}$	-0.0155	-0.0227	rad

TABLE 10. Results of the trim in all configurations.

Description	Symbol	Value		Unit
		Without RE	With RE	
	α_1	12.09	12.15	deg
Angle of attack	α_2	9.29	8.35	deg
	α_3	6.82	5.38	deg
	δ_{e_1}	-14.24	-15.28	deg
Elevator deflection	δ_{e_2}	-5.56	-6.23	deg
	δ_{e_3}	-8.46	-7.72	deg
	δ_{t_1}	0.548	0.547	
Throttle setting	δ_{t_2}	0.467	0.438	
	δ_{t_3}	0.393	0.366	
	ω_{eng_1}	4196	4197	rpm
Engine rotational speed	ω_{eng_2}	3854	3698	rpm
	ω_{eng_3}	3478	3251	rpm

TABLE 11. Variable parameters for Monte-Carlo analysis, nominal values and standard deviation.

Variable parameter	Nominal value	Std.deviation
$C_D S_p$	213 m^2	10.68 m^2
$V_{w_{h_0}}$	3.5 m/s	0.24 m/s
k_{a_0}	0.75	0.038
ψ	0 deg	4.10 deg

TABLE 12. Comparison of autopilot features.

Data	MP128^{HELI2}	AP-10.1	600	AUTOPILOT 4X
Producer	Micropilots	UAVOS	Vector	Veronte
Mass [g]	40	170	180	660
Dimensions [mm^3]	100x40x15	119x47x72	45x78x75	117x70x82
Temp. range [°C]	-40 to +85	-40 to +60	-40 to +85	-40 to +65
IP rating	-	IP67	IP66	IP67
Max. airspeed [kts]	500	-	220	206
Max. altitude [m]	12000	50000	12000	7500
Attitude accuracy [°C]	<1	<1	<0.5	<1.5