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Original Article

Conceptual Design of Unmanned Piston-Propeller Cargo Aircraft

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Abstract. The need for a transportation system to transport goods is increasing, especially for regions in Indonesia that are included in the leading, remote, and disadvantaged (3T) categories. This paper will discuss the conceptual design of an unmanned cargo aircraft with a piston-propeller engine which can later be operated in the 3T area. The design will begin with a discussion of DRO, followed by a concept of the aircraft configuration. After the configuration is formed, the aerodynamic characteristics will be calculated, the weight will be predicted, and flight performance calculations and aircraft stability will be calculated.

Keywords: Unmanned aircraft, cargo, piston engine, cargo aircraft

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1. Introduction

An unmanned piston-propeller cargo aircraft is designed to create and fill the market for General Aviation aircraft in Indonesia and the surrounding region. A Design Requirement and Objectives (DR&O) has been formulated to fulfill the market needs. The DR&O specifies the Maximum Take-Off Weight (MTOW), cruising speed, range, and altitude, maximum service ceiling, take-off distance, landing distance, rate of climb, price, and maximum speed. Our approach is to perform an initial aircraft design from the conceptual design using methods from reference [1], wing and tail design based on reference [2], geometry design and aerodynamics analysis using XFLR5, DATCOM, and OpenVSP platforms, structural design, and weight and balance based on reference [1], calculation of performance and stability based on reference [3], cost analysis based on reference [4], and final geometry drawing using Solidworks platform. CASR 23 is the certification basis of this aircraft.

The paper aims to describe cost analysis of aircraft, which firstly will be presented the DR&O of the designed aircraft, i.e., UCA-13, then the aircraft configuration determination process and prediction of aerodynamics analysis and structural design weight and balance respectively. The calculation of performance, stability and control related to the cost analysis of the aircraft will be drawn later on. It is.

2. Materials and Methods

The General Design Requirement of the new aircraft is stated as follows:

- Unmanned cargo piston propeller aircraft.
- The maximum takeoff weight must not exceed 4000 kg.

The aircraft should have Flight Performance as follows:

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- Maximum range with maximum payload is at least 650 km with cruising speed of 250 km/hrs at 12,000 ft.
- Maximum range with 400 kg payload is at least 2300 km with cruising speed of 250 km/hrs at 12,000 ft.
- Range is calculated by using NBAA IFR Range Profile Standard.
- Maximum service ceiling with maximum takeoff weight is at least 20000 ft.
- Certified takeoff distance with maximum takeoff weight must not exceed 600 m at Sea Level ISA+15.
- Certified landing distance with 0,95 maximum takeoff weight must not exceed 650 m at sea level ISA +15.
- Maximum rate of climb with maximum takeoff weight at sea level must not be less than 1600 fpm.
- For multi-engine aircraft, must comply with one engine inoperative requirements from CASR23.67.

The Design Objectives is specified as follows:

- The aircraft may be designed using the latest and consistent aerodynamics, construction, and system technologies with maximum reliability and maintainability during its lifetime.
- Price per unit may not exceed 2.4 million USD
- Maximum cruising speed at FL120, with 0,95 Maximum takeoff weight may not be less than 300 km/hrs

3. Results

The aircraft uses a monoplane, un-swept, low-wing configuration, with a conventional Horizontal Tail Plane (HTP) and Vertical Tail Plane (VTP) empennage, rounded rectangular fuselage cross-section, nose-mounted tractor engine, and a tricycle fixed landing gear. The aircraft is designed to complete flight missions as stated in the following mission profile.

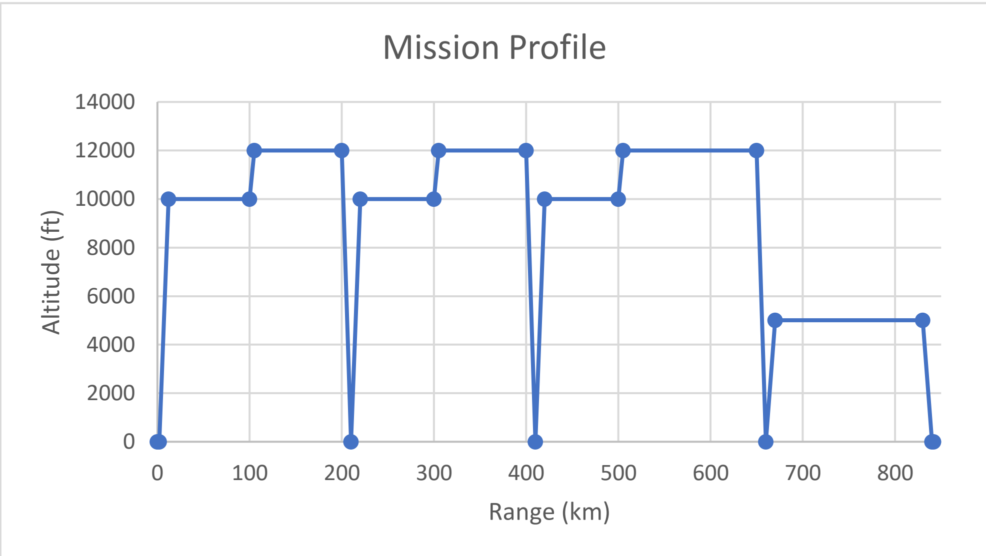

Figure 1. Mission Profile of UCA-13

Based on the designed mission profile, the aircraft weight sizing can be estimated based on methods in the reference [1]. The aircraft weight sizing estimates the Operational Empty Weight (WOE), Fuel Weight (WF), Payload Weight (WP), and Take-Off Weight (WTO).

Table 1. Weight Initial Sizing

W_{OE}	1500	kg
W_F	600	kg
W_P	1600	kg
W_{TO}	**3700**	**kg**

Based on the reference [1], the aircraft's initial sizing was determined by using the matching chart method. The matching chart gives guidelines in determining the design point of Wing Loading (W/S) and Power Loading (W/P). The wing area and engine power can be determined based on the Wing Loading and Power Loading design point. The appropriate engine to equip this aircraft is AC-AERO Eagle E-660 J/G-T with 596 kW engine power. While the appropriate wing area is 35 m2. Fuselage sizing began with the determination of the fuselage cross-section and corner radius. To comply with the DR&O 9 m3 of cabin volume, the cargo section length is determined to be around 4 meters. With a reference of 25% - 35% - 40% portion of nose-cabin-empennage length, the total fuselage length needed is 12 meters. The sizing of horizontal and vertical tails is determined using Tail Volume Coefficient based on previous study [1].

Based on the reference [2], to determine the position of the nose landing gear (xn) and the main landing gear (xm) in the x-axis, the center of gravity in the x-axis (xcg) of our aircraft needs to be estimated. The ground clearance can be determined using the typical value for the propeller ground clearance, which is around 0.5 m from the ground. Added with the information about typical landing gear load distribution, the location of nose landing gear and main landing gear can be determined.

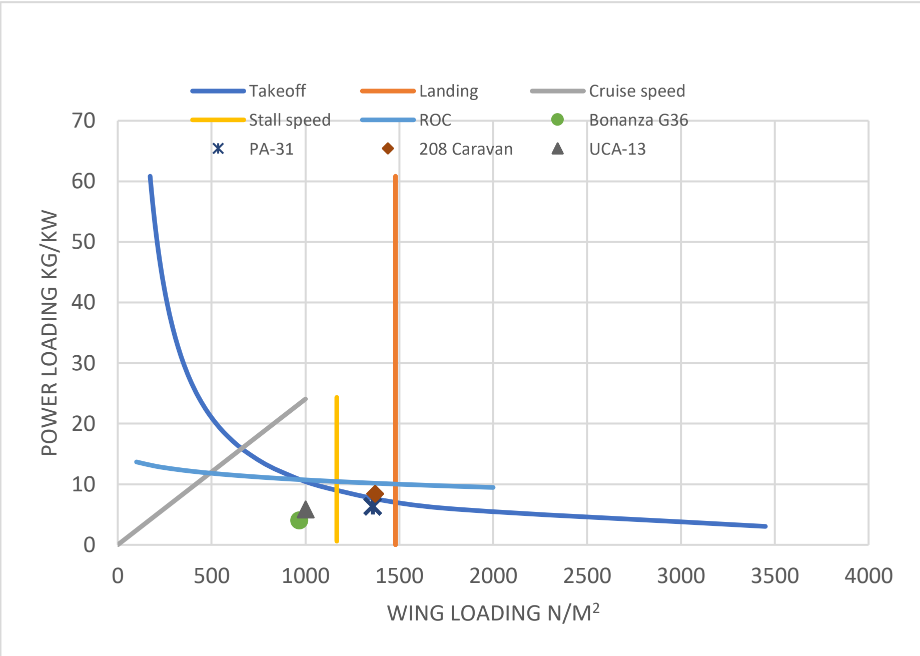

Figure 2. Matching Chart Estimation

The wing design process is based on the reference [2]. Using information of cruise Average Weight (Wavg), Take-Off Weight (WTO), and Stall Speed (Vstall), the suitable airfoil, wing incident angle, wing twist angle, and High Lift Devices can be determined. A winglet is added to the tip of the wing to reduce the induced drag. High Lift Devices are deployed to increase the Lift Coefficient

(CL) during take-off and landing. Trailing edge single-slotted flap is designed with the size of the HTP and VTP designs are also based on the reference [2].

Known the information of Wing Lift Coefficient (CL_wing), HTP location (lh), Wing-Fuselage Coefficient of Moment (Cm_wf), HTP efficiency (ηh), and Coefficient of HTP Volume, the HTP Lift Coefficient (CL_HTP) can be determined. An appropriate HTP airfoil and incident angle can be determined to satisfy the HTP Lift Coefficient. The longitudinal stability of the aircraft with HTP can be calculated using the equation of Angle of Attack derivative Moment Coefficient. The specification of the aircraft configuration is described in the following table and figures.

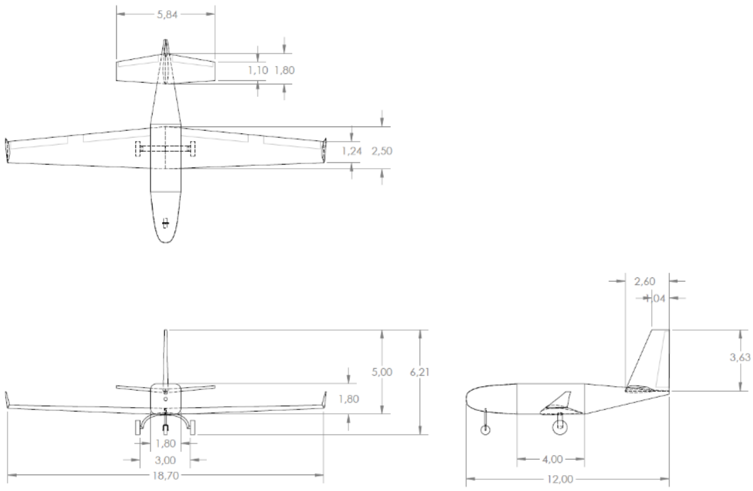

Figure 3. Aircraft Three-View Drawing

Table 2. Aircraft Geometry Configuration

Geometry	Wing	HTP	VTP	Unit
Area	35	8.52	6.6	m^2
Root chord	2.5	1.82	2.6	m
Tip chord	1.25	1.09	1.04	m
MAC	1.94	1.49	1.93	m
Airfoil	NACA4415	NACA0012	NACA0012	

Geometry	Fuselage	Unit
Length	12	m
Width	1.8	m
Height	1.8	m
Corner Radius	0.35	m
Cargo Section Length	4	m

4. Discussion

4.1 Prediction of Aerodynamic Characteristics

XFLR5, DATCOM, and OpenVSP software are used to estimate the aerodynamics characteristics of CL-alpha, drag polar, and L/D. The airfoil Cl-alpha, Cl-Cd, Cl/Cd-alpha, and Wing CL-alpha are obtained as a result of batch analysis of multiple Reynolds numbers using XFLR5. For the whole aircraft configuration, the aerodynamic analysis such as CL, CD, CM for each angle of attack is done

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using DATCOM software. The characteristics of the High Lift Devices also can be obtained using the DATCOM software. The fixed landing gear configuration drag polar is obtained using OpenVSP software. Based on the Drag Polar, the zero-lift coefficient of drag for cruise conditions is around 0.028. Meanwhile, the zero-lift drag for take-off is around 0.04. The zero-lift drag for the landing condition is approximately 0.1. The resulted aerodynamics characteristics are described in the following figuresYOLOv4 algorithm and Google Colaboratory were used in this study, where each model took 8-10 hours to be trained, as presented in Table 1 to Table 8 below.

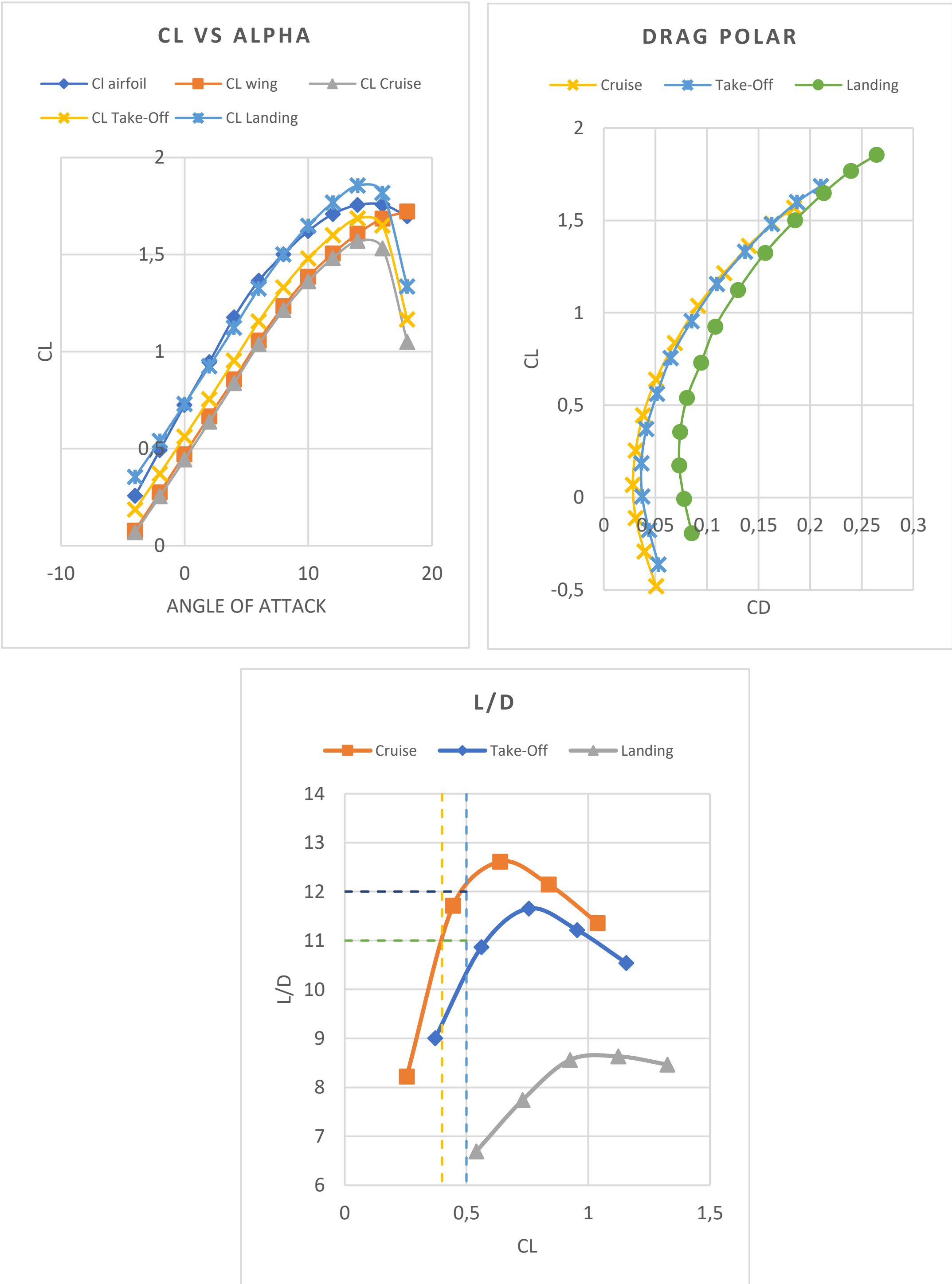

Figure 4. Aerodynamics Curves

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4.2 Structural Design and Weight and Balance

The structural design that will be implemented in this aircraft is a semi-monocoque construction. It is chosen because it offers the best balance between cabin space and structural strength, which is also why it is the most popular structural design among aircraft. When applied to the fuselage, a semi-monocoque design incorporates the same major components namely the skin, frame (formers), stringer/longeron, and Bulkheads. Bulkheads are not required on this aircraft as the cabin is unpressurized but a similar partition between the cargo area, nose, and empennage is still required. Regarding materials, since the fuselage is one of the major load-bearing structures of an aircraft and requires rigid materials, the use of composite material is extensive. Meanwhile, for the joint between the wing and the fuselage, this aircraft will incorporate a bolted joint since this is still a relatively small aircraft and this type of joint is strong and quite resistant to fatigue while still being lightweight. In addition, the bolt joint allows an easy dismantling of the wing.

The number of frames on the structure can be determined by considering the length of the fuselage. With a length of 12 meters, the number of required frames would be 18. With a perimeter of 6.6 meters, the number of required longerons/stringers would be 20. The value of 20 also sets the longeron spacings to around 0.33 meters (13 inches). The spar on the wing, as advised by reference [1], is placed at the 15% and 65% chord. The 65% taken for the rear spar is the most proximal position due to giving space for the flaps towards the end. The 15% taken for the front spar is the most distal position as to space out the distance between the two spars, to offer a bit more rigidity. The ribs will be spaced in 60-cm increments (24 inches) from the tip to the root. With our wingspan being 18.71 m from tip to tip, that will mean that each wing will have 15 ribs. As for the stringer, a spacing of 12 cm is used as it is also commonly used among the competitors as well. As such, 16 stringers will be used on the top and bottom.

For the HTP, the front spar is placed at around 15-25% chord and the rear spar at 70-75% chord. Thus, we have chosen a 15% front spar and a 70% rear spar from aluminum. The ribs will also be placed in 24-inches spacings. With a span of 5.84 m, the ribs will be equipped with 4 ribs placed evenly from root to tip. Also, with 12-cm spacings on the stringer, the HTP will have 11 stringers. For the VTP, the front spar is placed at around 15-25% chord and the rear spar at 70-75% chord. Thus, we have chosen a 15% front spar and a 70% rear spar from aluminum. The ribs will also be placed in 24-inches spacings. With a span of 3.63 m, the ribs will be equipped with 6 ribs placed evenly from root to tip. Also, with 12-cm spacings on the stringer, the VTP will have 16 stringers. In summary, the structure is characterized by the following figures.

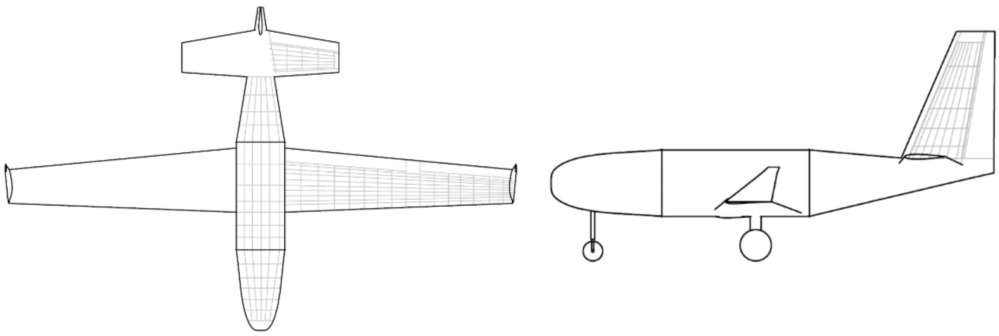

Figure 5. Structural Layout

Weight and balance calculation is done to obtain the position of the Center of Gravity (CG). By grouping, the components into 4 major parts: Fuselage, Wing, HTP, and VTP, the weight, and location of each major part can be calculated into a centroid equation. The Fuselage, Wing, HTP, and VTP contribute to 91%, 7%, 1%, and 1% of the whole aircraft weight, respectively. The longitudinal location of the CG is around 5.26 to 5.47 meters from the nose, or 32% to 43% of MAC, varied according to the payload and fuel configuration. The weight and balance are described in the following table.

Table 3. Weight and Balance

Part	Weight (kg)	X-Location (m)	Y-Location (m)	Z-Location (m)
Fuselage	**3330**	**5.2**	**0**	**-0.2**
Wing	274	4.6	0	-0.6
HTP	41	9.6	0	0.5
VTP	32	9.5	0	1.3
CG	**3676**	**5.26**	**0**	**-0.21**

4.3 Calculation of Performance, Stability, and Control

The aircraft performance criteria occupy the take-off distance, rate of climb, service ceiling, cruise speed and range, maximum speed, and landing distance. The performance calculation is based on the methods in the reference [3]. Previously, the aerodynamics performance has been already predicted. the propulsion performance is estimated using typical characteristics of engine performance in reference [1] as well as data from the engine manufacturer. Considering the aerodynamics and propulsion performance, the aforementioned performance criteria can be calculated.

Table 4. Performance Estimation

Parameter	DR&O	Result	Unit
MTOW	4000	3700	kg
Take-off distance with MTOW	600	500	m
Maximum Rate of Climb with MTOW	1600	1800	ft/min
Maximum Range with maximum payload	810	880	km
Maximum Range with 400 kg payload	2460	2600	km
Maximum service ceiling	20000	35000	ft
Maximum cruising speed	300	305	km/h
Landing distance with 0.95 MTOW	650	480	m

The static stability of the aircraft consists of longitudinal and lateral-directional stability. The longitudinal static stability is mainly supported by the presence of HTP. While the lateral-directional is supported by the wing dihedral and VTP. The calculation of static stability is done using tools from the DATCOM platform and reference [2]. Considering the Wing-Fuselage Lift Coefficient-AOA derivative ($CL\alpha\_wf$), CG location, HTP location concerning the CG, HTP Lift Coefficient-AOA derivative ($CL\alpha\_H$), HTP efficiency, HTP area concerning Wing Area, and the downwash derivative ($d\varepsilon/d\alpha$), the Pitching Moment Coefficient-AOA derivative ($Cm\alpha$) can be calculated. The value of $Cm\alpha$ is calculated -0.95/rad. The lateral-directional stability is calculated using the DATCOM tool, represented in Yawing Moment Coefficient-Sideslip derivative $Cn\beta$. The $Cn\beta$ is calculated at 0.12. The static stability already satisfies the regulation, with a negative value of $Cm\alpha$ and a positive value of $Cn\beta$. The neutral point is located at 50% of the mean aerodynamic chord, with the static margin value in the range of 0.07 – 0.18.

4.4 Cost Analysis

A cost estimation analysis is done based on the methods explained in reference [4]. Considering the aircraft performance data, powerplant data, design and production factors, and cost factors, the RDTE cost and production cost can be estimated. The aimed outputs are the aircraft cost and production quantity in 2025. The results of cost analysis: the breakeven point is at 150 units of aircraft, selling at USD 3,200,000. After reaching 150 units sold, the UCA-13 production will be profiting, as presented on Fig. 6.

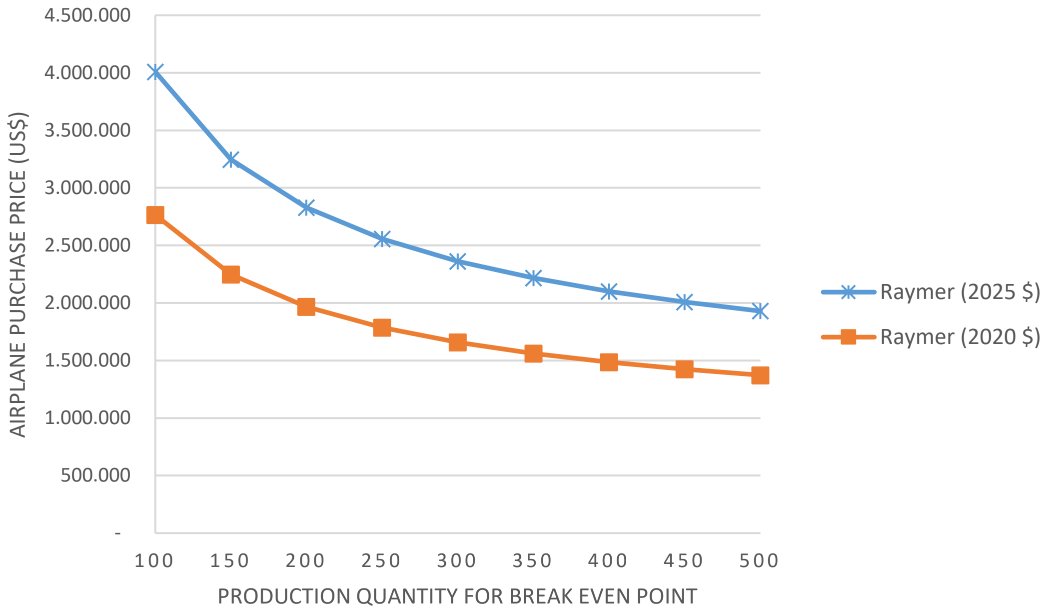

Figure 6. Estimation of Airplane Price and Break-Even Point

5. Conclusions

An unmanned piston-propeller cargo aircraft UCA-13 has been designed to fulfill the DR&O. The conceptual design process results in the concept, weight, and flight performance that satisfy the DR&O. The design complies with the CASR 23 as the certification basis, proven by the values of the static stability. The UCA-13 will be ready to be launched in 2025. The aircraft will be sold at USD 3,200,000, and profiting after 150 units sold.

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