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FACULTY OF MECHANICAL ENGINEERING
FAKULTA STROJNÍHO INŽENÝRSTVÍ

INSTITUTE OF AEROSPACE ENGINEERING
LETECKÝ ÚSTAV

LIGHT SPORT AIRCRAFT ENGINE REPLACEMENT
REMOTORIZACE LEHKÉHO SPORTOVNÍHO LETOUNU

MASTER'S THESIS
DIPLOMOVÁ PRÁCE

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Pursuant to Act no. 111/1998 concerning universities and the BUT study and examination rules, you have been assigned the following topic by the institute director Master's Thesis:

Light sport aircraft engine replacement

Concise characteristic of the task:

The aim of the diploma thesis is the engine replacement of light–sport aircraft PS–28 Cruiser from the current engine to some of the more powerful type. The solution of the given problem includes a complex range of design tasks.

Goals Master's Thesis:

1) The search of available engines and comparison of performance parameters.
2) Mass and balance analysis for the selected engine.
3) Calculation of flight envelopes according to the selected regulation.
4) Engine mount design.
5) Engine mount test plan.

Recommended bibliography:

NIU, Michael Chun-Yung. Airframe structural design. 2nd ed. Hong Kong: Hong Kong Conmilit Press Ltd. ISBN 962-7128-09-0.

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Abstract

The thesis is focused to find optimal and new engine for PS-28 Sport Cruiser, for more power and thrust.

The Rotax 912 ULS is currently the only engine that the client of this work (Czech Aircraft Works, hereinafter CZAW) installs into Sport Cruiser aircraft. This four-cylinder power unit has a maximum take-off power of 100 hp. For more demanding customers, the offer will be expanded by Lycoming O-235-L2C, which is also four-cylinder engine with maximum take-off power 116 hp. Which means, company will have bigger option and client gonna be much more satisfied

Keywords

Powerplant, Aircraft empty data, ROTAX 912 ULS, Lycoming O-235-L2C, Flight envelope, engine mount.

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I hereby declare that I am the sole author of this master’s thesis and that I have not used any sources other than those listed in the bibliography and identified as references. I further declare that I have not submitted this thesis at any other institution in order to obtain a degree.

BEng. Nikolozi Totogashvili

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1. Introduction

1.1 Objectives which should be achieve:

1) The search of available engines and comparison of performance parameters.

2) Mass and balance analysis for the selected engine.

3) Calculation of flight envelopes according to the selected regulation.

4) Engine mount design.

5) Engine mount test plan.

The aim of the diploma thesis is the engine replacement of light-sport aircraft PS-28 Cruiser from the current engine to some of the more powerful type. The solution of the given problem includes a complex range of design tasks.

The Rotax 912 ULS is currently the only engine that the client of this work (Czech Aircraft Works, hereinafter CZAW) installs into Sport Cruiser aircraft. This four-cylinder power unit has a maximum take-off power of 100 hp. For more demanding customers, the offer will be expanded by Lycoming O-235-L2C, which is also four-cylinder engine with maximum take-off power 118 hp.

Second stage is Cg, mass and balance analysis, for this calculations I used Excel.

On the third stage I calculated Flight envelope and the last level is engine mount design, for this design I needed Catia V5(6) software.

2. DESIGN OVERVIEW

2.1. BASIC DESIGN

The Sport Cruiser is used extensively for flight training around the world. It's a double-seat, all-metal aircraft, arranged as a low-wing monoplane with cantilevered wings and conventional empennage.

The aircraft has a tricycle fixed landing gear of nose type. The Sport Cruiser has been approved for operation in the USA and other countries in the LSA (Light Sport Aircraft) category according to the FAA (Federal Aviation Administration) Accepted ASTM Consensus Standards. [5]

Figure 1. Ps-28 sport cruiser upper view [5]

Figure 2. Ps-28 sport cruiser side view [5]

2.1.1 WING

The wings are of all-metal main spar structure with a rear spar. Each of the two wings is attached to the center section fuselage by means of 6 shear bolts. Fuel tank is located in each wing with a capacity of 57 litres. Unique lockable storage compartment is located in each wing with a load capacity of 10kg each.

The wing tips are made from carbon-glass composite. Both wings are equipped with electrically controlled flaps and ailerons. The right aileron is fitted also with electrical trim.

Position and strobe lights are installed in the wing tips.

2.1.2 TAIL UNIT

The aircraft tail unit consists of the vertical fin, the rudder, the horizontal stabilizer and the elevator with trim tabs. The elevator is well balanced by the electrically controlled trim of the elevator. The elevator is equipped also with a balance tab for aircraft better controllability and balance of the control forces.

2.1.3 FUSELAGE

The fuselage is designed as a semi monocoque conventional all-metal structure. The cockpit fixed frame and the movable frame are made from a composite. The main undercarriage is attached directly to the bottom part of the fuselage. The nose landing gear is attached to the firewall.

2.1.4 COCKPIT

The Sport Cruiser boasts by far the most spacious in its class, which provides utmost level of comfort and highly ergonomic arrangement to the flying crew.

The canopy window is made from plexiglass and is fitted with two openable side windows. There is a luggage compartment behind the pilot seats with the load capacity of 18 kg. The instrument desk consists of three panels. Thanks to dual flight controls and pedals the aircraft is fully controllable from both pilot seats. The cockpit is equipped with adjustable pedals. The first-class choice of materials used for the cockpit upholstery, advanced on-board climate control and defogging systems together with the easy flight control and high endurance of the aircraft provide for the Sport Cruiser being widely recognized also as a true cross-country aircraft.

With the unique baggage lockers integrated into both wings together with the spacious baggage compartment located in the rear part of the cockpit, the Sport Cruiser will always guarantee optimum flexibility for that weekend away. [5]

TECHNICAL SPECIFICATION

Engine :	Rotax 912 ULS2
Power:	100 HP (73.5Kw) at 5800 RPM
Propeller:	Sensenich Fixed 3-blade (*)
Wingspan:	28.22 ft (8.6 m)
Length:	21.72 ft (6.62 m)
Height:	7.60 ft (2.315 m)
Wing surface area:	132.4 sq ft (12.3 m²)
Cockpit width:	3.85 ft (1.17 m)
MTOW:	1,320 lbs (600 kg)
Empty weight:	855.4 lbs (388 kg)
Max. baggage weight in cockpit compartment:	40 lbs (18 kg)
Max. baggage weight in each wing locker:	22 lbs (10 kg)
Cruise speed at 3,000 ft and 75% power:	93 KIAS (172 km/h)
Max. horizontal speed:	220 km/h
VNE :	138 KIAS (255 km/h)
Stall speed VS0:	31 KIAS (55 km/h)
Climb rate:	825 ft/min (4.2 m/s)
Range (30 min. reserve):	516 NM (953 km)
Endurance:	5 hours 25 min

Table 1. PS-28 Cruiser technical specification [1]

2.2 Choice of coordinate system

The coordinate system for easier orientation is chosen in accordance with the aircraft coordinate set (x, y, z)

The origin is placed in the center of gravity of the aircraft. This system is firmly connected to the aircraft. The x-axis is oriented in the direction of the fuselage axis, with its positive sense forward. The z-axis points down fros the aircraft. The y-axis is perpendicular to rhe previous twho axis with a positive sense on the pilot’s right hand side. All three axis are at right angle to each other.

X-axis – longitudinal axis

Y- axis – lateral axis

Z –axis – perpendicular axis

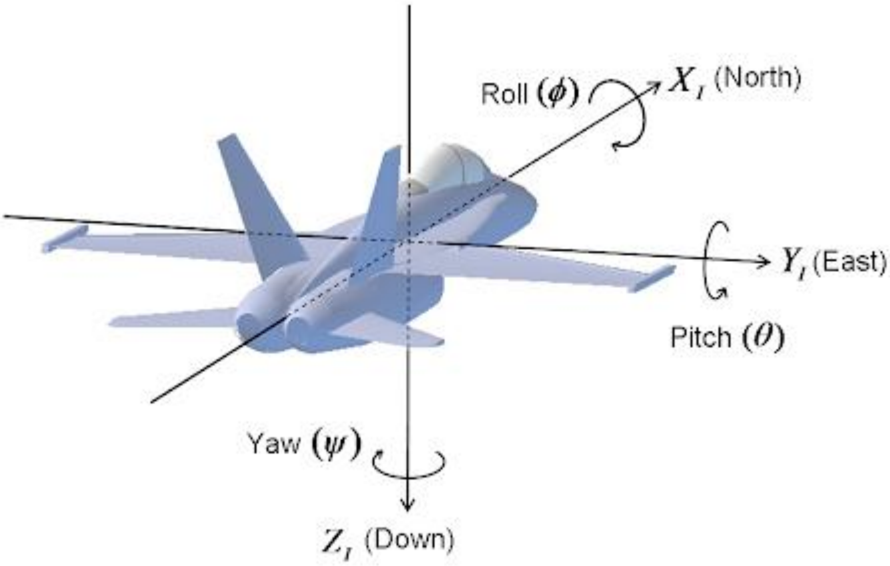

Figure 3. Aircraft coordinate system [11]

2.3 Curent Powerplant

ROTAX 912 ULS

100 hp

Figure 4. Rotax engine [1]

DESCRIPTION :

• 4-cylinder

• 4-stroke liquid/air-cooled engine with opposed cylinders

• dry sump forced lubrication with separate oil tank, automatic adjustment by hydraulic valve tappet

• 2 carburetors

• mechanical fuel pump

• dual electronic ignition

• electric starter

• propeller speed reduction unit

• air intake system

Price : $18,878.

Figure 5. 3 Blade Rotax 3BO ground adjustable propeller [14]

Facts:

Rotax 912 series the 100 hp product line offers more power while keeping the weight. This engine series offers a time between overhauls of 2.000 hrs and the best power to weight ratio in its class - no surprise that this engine is the best selling 4-stroke engine. This series is available as non-certified (Rotax 912 ULS) and certified engine (Rotax 912 S) according to FAR33.

For comparision Lycoming O-235 L2C TBO is 2400 hrs.

Performance			
73.5 kW		100 hp	5800 1/min
Torque			
128 Nm		94 ft.lb.	5100 1/min
	Max RPM		5800 1/min

Table 2. Rotax engine performance [1]

BORE		STROKE	
84 mm	3.31 inch	61.0 mm	2.4 inch
DISPLACEMENT		FUEL	
1352 ccm	82.6 cu in	min. MON 85 RON 95 min. AKI 91	

Table 3. Bore and stroke [1]

WEIGHT	kg	lb
Engine with propeller speed reduction unit i=2.43	56.6	124.5
Overload clutch	1.7	3.7
Exthaust system	4	8.8
External alternator	3	6.6
Engine mount	2	4.4
Air guide hood	0.4	0.8

Table 4. Engine weight [1]

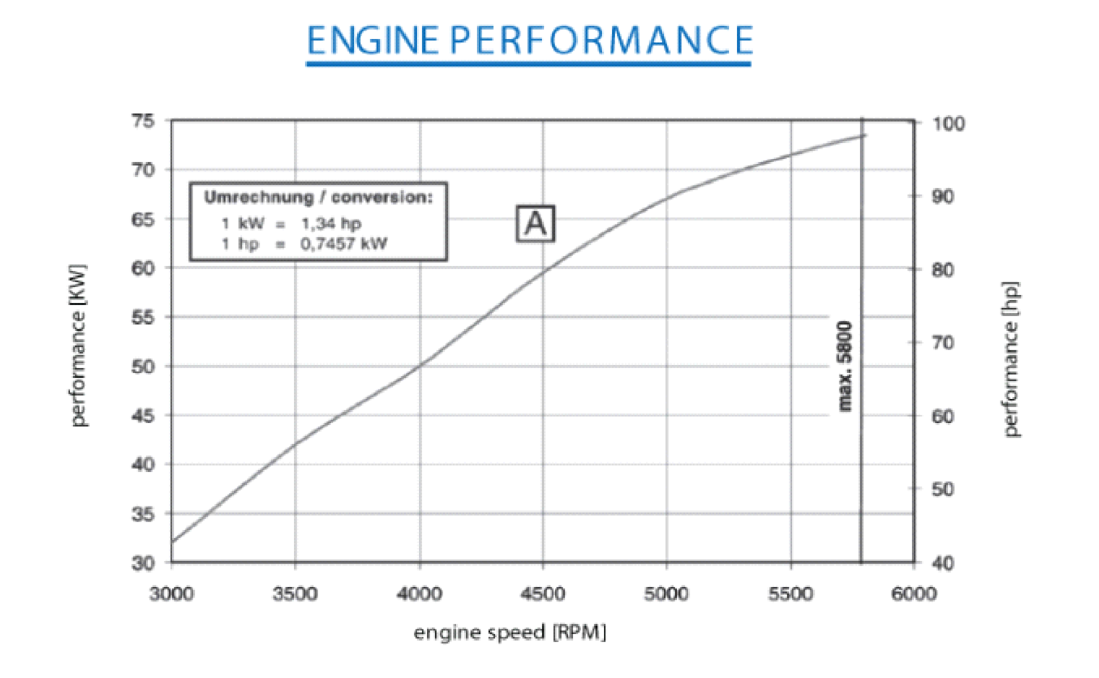

Figure 6. Engine performance [1]

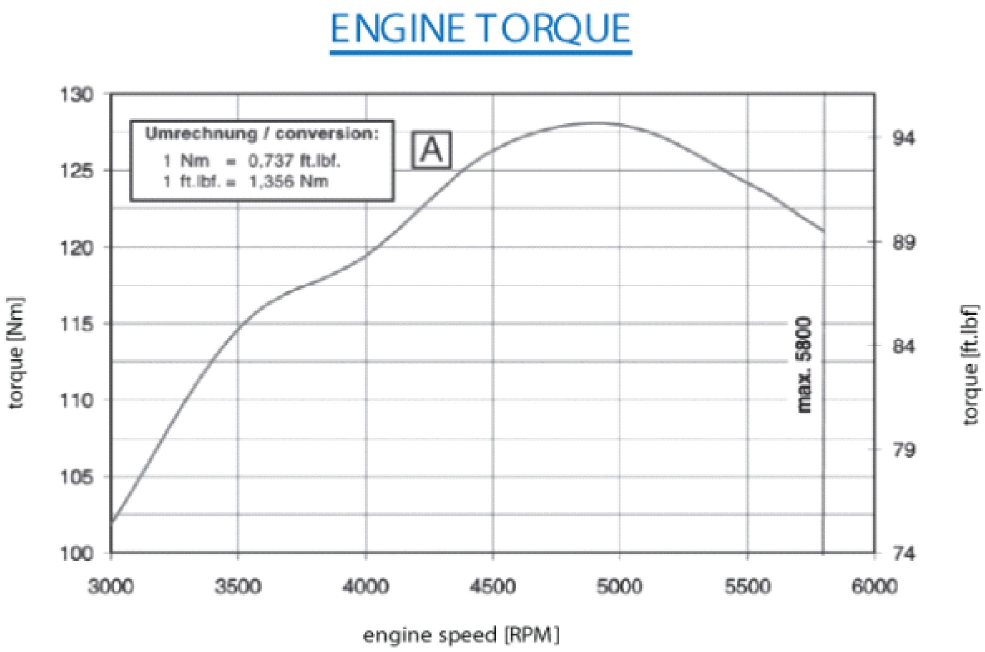

Figure 7. Engine torque [1]

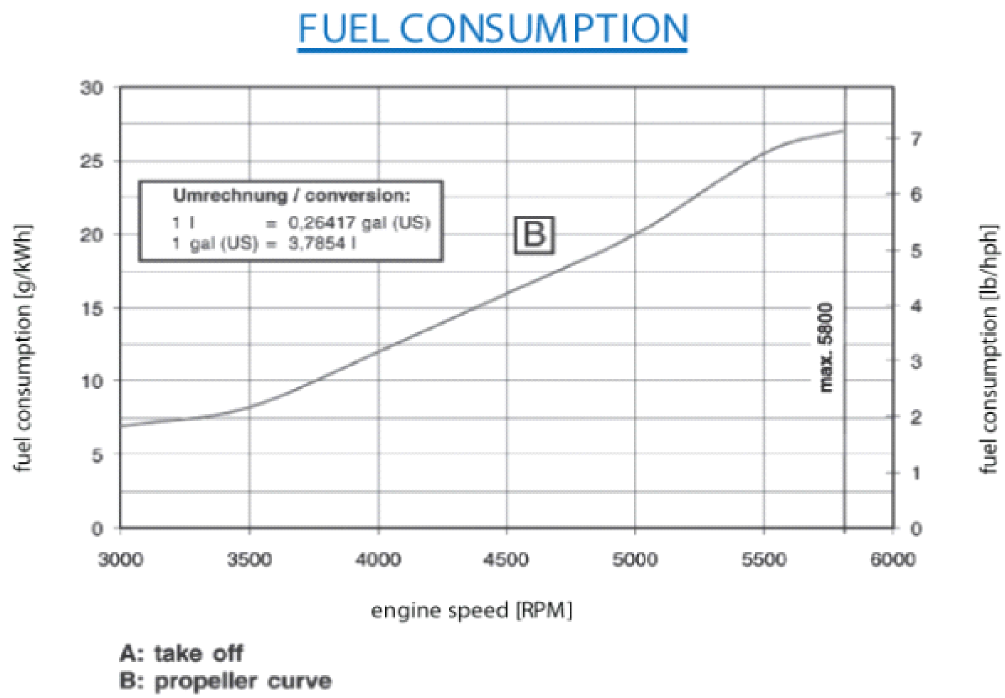

Figure 8. Fuel consumption [1]

2.4 Engine options

	Rotax 912 ULS/S	Rotax 914 UL	Rotax 915 IS	Lycoming O-235-L2C	Continental O-200-D	Continental O-200-A
	4-cylinder 4-stroke liquid-/air-cooled engine with opposed cylinders Dry sump forced lubrication with separate oil tank, automatic adjustment by hydraulic valve tappet 2 carburetors Mechanical fuel pump Dual electronic ignition Electric starter Propeller speed reduction unit Air intake system (Purchased Separately)	4-cylinder 4-stroke liquid-/air-cooled engine with opposed cylinders With turbocharger With automatic waste gate control Dry sump forced lubrication with separate oil tank Automatic adjustment by hydraulic valve tappet 2 carburetors Dual electronic ignition Electric starter Propeller speed reduction gearbox Engine mount assembly Air intake system Exhaust system	4-cylinder 4-stroke liquid/air-cooled engine with horizontally opposed cylinders Dry sump forced lubrication with separate oil tank, automatic adjustment by hydraulic valve tappet Redundant electronic fuel injection and ignition Engine management system (EMS) Electric starter (12 or 24 volt) Propeller speed reduction gearbox Air intake system with intercooler Turbocharger with stainless steel exhaust Service ceiling of 23,000 feet	4 cylinder, boxer, air cooled, lubricated with a dry cabinet, carburator, without reducter	The 200 Series is a family of 4-cylinder, 4-stroke, spark ignited, air-cooled, horizontally opposed, direct-drive reciprocating aircraft engines designed for fixed pitch, ground adjustable, or electric constant speed propeller. The engines are made with a wet oil sump, carburetor with manual mixture control, continuous fuel injection, and Full Authority Digital Engine Control (FADEC) fuel control.	FAA Certified, 200 cubic inch displacement, four cylinder carbureted, air-cooled horizontally opposed piston engine
Weight [kg]	56.6	83.6	92.2	117.5	93.8	95
Power [hp]	98	115	141	118	100	110
TBO [h]	2,000	2,000	1,200	2,400	2,200	2,200
Price [$]	18,900	30,700	37,000	30,500	11,000	25,000

Table 5. Engine options

There are possible engines which was submitted in the company on the table 5.

All that engines had somehow good characteristics, current one is Rotax 912 ULS, highest hp has Rotax 915 IS, but it’s the most expensive and time between overlaul (TBO) is the lowest.

From these engines was chosed Lycoming O-235-L2C because of high horse power performance, price, easy maintanace for company and biggest TBO.

2.5 New Powerplant

Lycoming

O-235-L2C

Figure 9. Lycoming O-235 LC [12]

2.5.1 General characteristics

- **Type:** Certified piston aero-engine
- **Bore:** 103.17 mm

- **Stroke:** 98.425 mm
- **Displacement:** 233.3 in^3 (3.823 L)
- **Dry weight:** 114 kg
- **Full Weight**: 117.5 kg

- **Fuel system:** carburetor
- **Fuel type:** 80/87 avgas
- **Cooling system:** air-cooled

Power output: 116 hp (88 kW) at 2800 rpm [2]

Depending on our task, which means, that we need more powerfull engine we can compare those two engines and will see that Lycoming has better charatceristics, it's cheaper but heavier and bigger, Theoritically, because of this mass and dimensions we have different CG and different nose parts for aircraft, also it's important to reinforce nose landing gear.

3. Mass and balance analysis for the selected engine

3.1. Three-view drawing

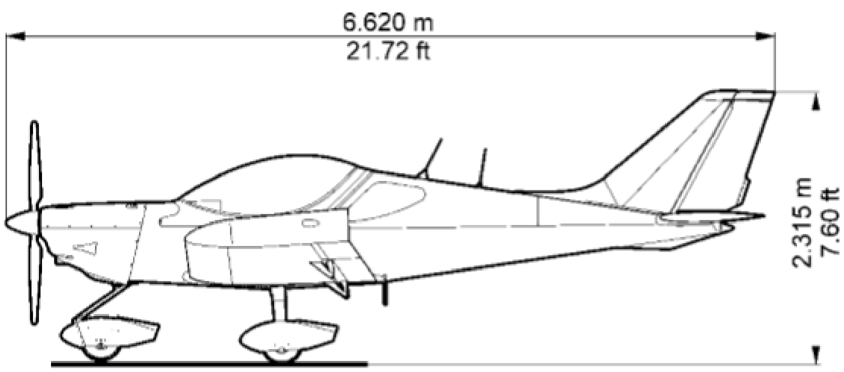

Figure 10. Ps-28 Sportcruiser side view [5]

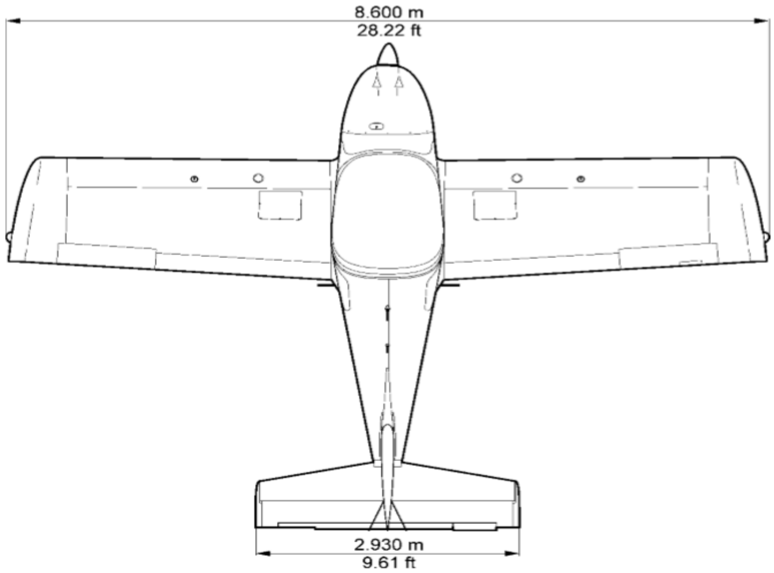

Figure 11. Ps-28 Sportcruiser upper view [5]

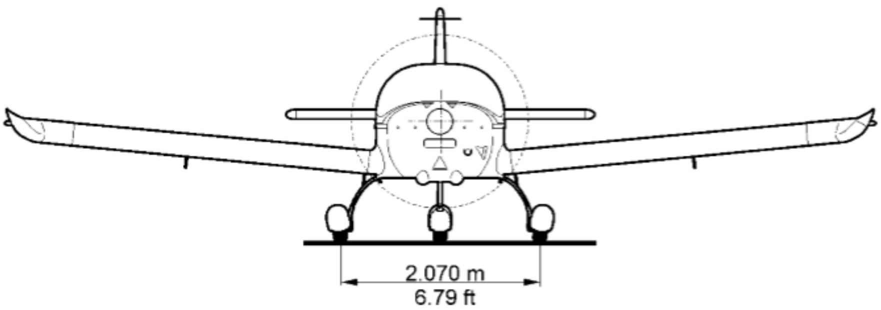

Figure 12. Ps-28 Sportcruiser front view [5]

3.2 Aircraft empty data:

Weight (without engine) ………………………….. 311.4 kg

Arm …………………………………………………. 432.4 mm

Moment …………………………………………….. 142 864.96 kg mm

MAC ………………………………………………… 1 500 mm

Operating weights :

Pilot …………………………………………………. 85 kg

Passenger …………………………………………….. 65 kg

Baggage in cockpit …………………………………. 5 kg

Baggage in wing lockers …………………………….. 0

Fuel in tanks ………………………………………… (16.2 – 21 kg)

For Cg calculation we need to find weights, arm and moment.

Arm can be determine with measurement. There are two kind of possibilities, 1) to measure from 0 point (in our case propeller) 2) to measure from leading edge.

I used second variant and got arms. Weights are given, so for moment and CG determination we can use

Formulas :

Moment (kg mm) = Weight (kg) ARM (mm) ,

$CG = \frac{M(total)}{W(total}$ (mm).

For empty aircraft with Rotax engine arm is 432.2 mm. For new engine CG calculation we need to calculate empty aircraft (without engine) arm :

$$arm_{empty\ aircraft} = \frac{387 \cdot 432.4 - (56.6 \cdot (-930))}{330} = 666.3\ mm$$

CG Calculation with new engine for maximum loads :			
Item	Weight (kg)	ARM (mm)	MOMENT (kg mm)
Empty Aircraft	311.3	666	207325.8
Pilot	85	700	59500
Passenger	65	700	45500
Baggage	5	1310	6550
Wing lockers	0	600	0
Fuel in tanks	16.2	180	2916
Engine	117.5	-930.35	-109316.125
Total	600	354.126125	212475.675
		23.608 %	

Table 6. CG calculation with new engine for max. loads.

You can see CG percentage (in green) for maximum loads on the table 6.

For Light Sport Aircraft (LSA) MTOW is 600 kg. In our case engine is 117.5 kg, which means twice more than Rotax. Because of this weight I desided to canceled wing lockers on maximum loads and decreased also baggage amount, which was 18 kg.

With 16.2 kg of fuel pilot can fly 1 hour with passanger.

	1 Pilot [kg]	1 Passanger [kg]	Fuel [kg]	Baggage [kg]	Weight (kg)	Xt (mm)	Zt (mm)	Center (%)
A	85	65	16.2	5	600	2089.1	980.1	23.6
B	85	65	0	0	578.8	2101.6	985.4	23.3
C	85	0	60	0	573.8	2089.3	980.6	19.5
D	65	65	16.2	5	580	2063.1	972.1	22.8
E	65	65	0	0	558.8	2091.5	984.2	22.5
F	85	0	16.2	5	535	2024.2	961.9	20.8
G	85	0	0	0	513.8	2067.5	981.5	20.4
H	65	0	16.2	5	515	2017.1	960.7	19.8
I	65	0	0	0	493.8	2060.5	980.8	20
J	55	0	16.2	5	505	2007.5	959	19.2

Table 7. All possibilities

You can find all possibilities for flight below 600 kg on table 7.

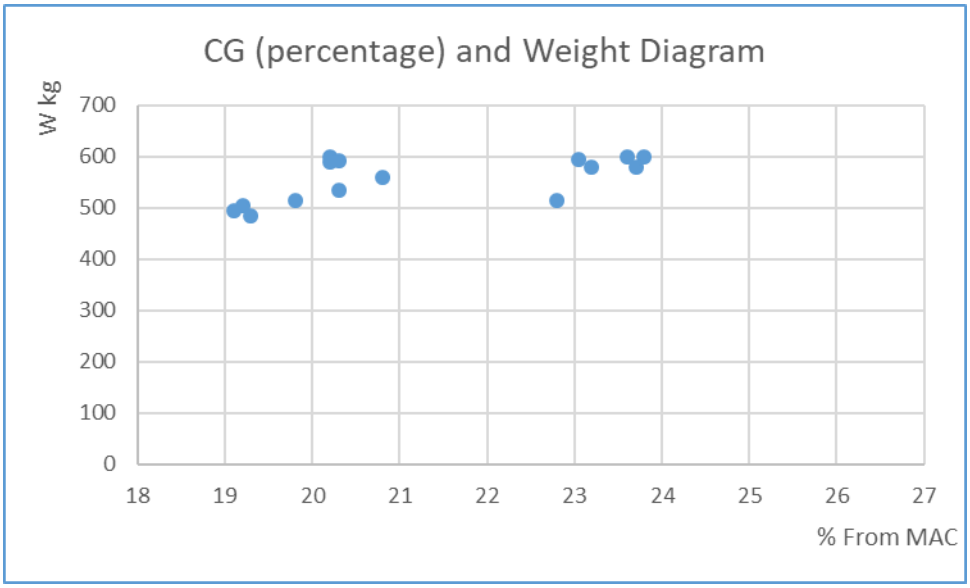

Figure 13. CG and weight diagram

On the Figure 13. X axe is percentage and Y is Weights.

Which means that CG varies between 19.1 – 23.8 %

15.1 % CG is for empty aircraft only with Lycoming engine and 23.8 % is CG on max. load.

CG Calculation with new engine for maximum loads :			
Item	Weight (kg)	ARM (mm)	MOMENT (kg mm)
Empty Aircraft	311.3	666	207325.8
Pilot	85	700	59500
Passenger	65	700	45500
Baggage	5	1310	6550
Wing lockers	0	600	0
Fuel in tanks	16.2	180	2916
Engine	117.5	-930.35	-109316.125
Total	600	354.126125	212475.675
		23.608 %	

Table 8. Cg calculation for aircraft with fuel, pilot and passanger

CG Calculation with Pilot, Passanger and 75% of Fuel			
Item	Weight (kg)	ARM (mm)	MOMENT (kg mm)
Empty Aircraft	311.3	666	207325.8
Pilot	85	700	59500
Passenger	65	700	45500
Baggage	0	1310	0
Wing lockers	0	600	0
Fuel in tanks	12.15	180	2187
Engine	117.5	-930.35	-109316.125
Total	590.95	347.2318724	205196.675
		23.149 %	

Table 9. Cg calculation with pilot, passanger and 75 % of fuel

There is shown 75% amount of Fuel and other max. loads on the table 9.

4. Calculation of performance (Power and thrust, Drag polar estimation, flight regimes) and flight envelopes according to LSA-F2245 regulation.

4.1 Input data

Aircraft data	
Wing span m	8.6
Wing area m^2	12.3
MTOW kg	600
empty weight kg	311.3
gravitaional acceleration m/s^2	9.80665
power kW	74.57
cruise speed [km/h]	167
stall speed [km/h] with flaps	52
max speed [km/h]	270
rate of climb [m/s]	5
ni. Prop	0.85
pi	3.141592654
Oswald	0.8
Cd0	0.039
stall speed [km/h] without flaps	59.264
Load factor	4
Fuel weight [kg]	16.2
Specific fuel consumption c [l/h]	20
With out fuel W1 [kg]	583.8

Table 10. Aicraft input data.

4.2 Power and Thrust estimation

altitude [m]	0	2000	4000	6000	8000	10000
density kg/m3	1.225	1.000	0.819	0.660	0.525	0.413
Max power [kW]	74.57	59.09	46.64	35.68	26.43	18.69
stall speed at altitude km/h	59.26	65.59	72.48	80.76	90.51	102.10
max speed throught alltitude km/h	204	201	196	190	177	140

Table 11. Aircraft speed properties on different altitutes

Max power is calculated by:

$$P = P_{max} \cdot \left(1.13 \cdot \frac{\rho(h)}{1.225} - 0.13\right)$$

.

	Available power [kW]					
Altitute [m]	0	2000	4000	6000	8000	10000
59.3 km/h	47.175	37.384	29.50719	22.575	16.722	11.826
78.3 km/h	53.788	42.624	33.64377	25.74	19.066	13.484
97.4 km/h	58.074	46.021	36.32434	27.791	20.585	14.559
110 km/h	59.303	46.995	37.09344	28.379	21.021	14.867
116 km/h	60.771	48.158	38.01121	29.081	21.541	15.235
136 km/h	62.392	49.443	39.0255	29.857	22.116	15.641
155 km/h	63.226	50.104	39.5471	30.256	22.412	15.85
174 km/h	63.334	50.189	39.61467	30.308	22.45	15.878
193 km/h	62.552	49.57	39.12567	29.934	22.173	15.682
212 km/h	60.491	47.936	37.83633	28.947	21.442	15.165
231 km/h	56.535	44.801	35.36166	27.054	20.04	14.173
250 km/h	49.842	39.497	31.17546	23.851	17.668	12.495

Table 12. Power in different altitute

Available power is calculated by formula:

$$P_{av} = P \cdot Prop.eff$$

Thrust is calculated by formula:

$$F = \frac{P_{av}}{V}$$

4.3 Drag polar estimation

Next step calculation of Lift and drag coefficients through height and velocities by formulas:

$$C_L = \frac{2m_{tow}g}{\rho v^2 S}$$

$$C_D = C_{d_0} + \frac{C_L^2}{\pi \cdot e \cdot AR}$$

		Lift coeficient					
Altitute [m]		0	2000	4000	6000	8000	10000
59.3	[km/h]	2.881932	3.530367	4.310583	5.351474	6.721948	8.554319
78.3	[km/h]	1.649397	2.020511	2.467047	3.062773	3.847127	4.895836
97.4	[km/h]	1.066714	1.306724	1.595512	1.980786	2.488051	3.166282
98.4	[km/h]	0.918094	1.124665	1.373217	1.704813	2.141403	2.72514
116	[km/h]	0.74598	0.913826	1.115782	1.385214	1.739957	2.214261
136	[km/h]	0.550824	0.674759	0.823882	1.022828	1.284767	1.634988
155	[km/h]	0.423318	0.518565	0.633168	0.786062	0.987366	1.256518
174	[km/h]	0.335458	0.410936	0.501753	0.622913	0.782437	0.995726
193	[km/h]	0.272361	0.333643	0.407378	0.505749	0.635268	0.808439
212	[km/h]	0.225526	0.27627	0.337326	0.418781	0.526028	0.66942
231	[km/h]	0.18981	0.232517	0.283903	0.352459	0.442721	0.563404
250	[km/h]	0.161952	0.198391	0.242236	0.300729	0.377744	0.480715

		Drag coeficeient					
Altitute [m]		0	2000	4000	6000	8000	10000
59.3	[km/h]	0.588586	0.863723	1.268533	1.934026	3.028914	4.881161
78.3	[km/h]	0.219019	0.309141	0.441738	0.659723	1.018358	1.625069
97.4	[km/h]	0.114295	0.151989	0.207449	0.298623	0.448625	0.702388
105	[km/h]	0.094775	0.122698	0.163781	0.231319	0.342435	0.530412
116	[km/h]	0.075823	0.094258	0.121381	0.16597	0.23933	0.363434
136	[km/h]	0.059077	0.069128	0.083916	0.108227	0.148224	0.215888
155	[km/h]	0.050858	0.056794	0.065528	0.079887	0.10351	0.143473
174	[km/h]	0.046446	0.050174	0.055659	0.064676	0.07951	0.104607
193	[km/h]	0.043909	0.046366	0.049982	0.055925	0.065704	0.082248
212	[km/h]	0.042366	0.044051	0.04653	0.050605	0.05731	0.068653
231	[km/h]	0.041384	0.042577	0.044333	0.04722	0.05197	0.060004
250	[km/h]	0.040736	0.041604	0.042883	0.044984	0.048442	0.054291

Table 13. lift and drag coefficient in different altitude and speed

From drag coefficient, Drag can be calculated by formula:

$$D = \frac{\rho v^2 S C_D}{2}$$

Next step is to calculate required power:

$$P_{req} = D \cdot V$$

4.4 Analyzed flight regimes:

4.4.1 Horizontal flight

Now we can graph available power-velocity and required power-velocity graph. From where we can find maximum service ceiling altitude speed by finding required and available power intersection. Minimum speed can be calculated by formula:

$$V_s = \sqrt{\frac{2m_{TOW}g}{\rho(H)C_L S}}$$

4.4.2 Climb

Climb speed is calculated by formula:

$$\omega = \frac{P_{av} - P_{req}}{W}$$

		Climb velocity m/s					
	Altitute [m]	0	2000	4000	6000	8000	10000
59.3	km/h	4.66	2.33	0.17	-2.11	-4.58	-7.38
78.3	km/h	6.25	3.91	1.82	-0.31	-2.52	-4.93
97.4	km/h	6.97	4.67	2.66	0.64	-1.38	-3.53
110	km/h	7.04	4.85	2.94	1.07	-0.79	-2.72
116	km/h	6.57	4.55	2.80	1.09	-0.59	-2.31
136	km/h	5.59	3.81	2.28	0.78	-0.69	-2.21
155	km/h	4.08	2.64	1.38	0.14	-1.09	-2.37
174	km/h	2.00	0.98	0.08	-0.83	-1.77	-2.78
193	km/h	-0.77	-1.24	-1.69	-2.19	-2.77	-3.46
212	km/h	-4.38	-4.13	-4.01	-4.00	-4.12	-4.42
231	km/h	-9.00	-7.85	-7.00	-6.33	-5.90	-5.72
250	km/h	Maximum climb speed in altitude [m/s]					
	7.04	4.85	2.94	1.09	-0.59	-2.21	

Table 14. Climb speed in different altitude

4.5 Gliding (descend)

To calculate Aircraft gliding regime we need to start lift coefficient with small value like 0.1 and linearly increase until it reaches maximum lift coefficient on stall speed without flaps, calculate gliding ratio, velocities and gliding angle for every Lift coefficient.
Formula for gliding ratio:

$$k = \frac{C_L}{C_D}$$

Formula for gliding angle:

$$arc\tan\left(\frac{C_D}{C_L}\right)$$

If during flight there will be opposite direction wind coordinate system will move to right same length as wind speed. It will be opposite in directional flight. To find best gliding ratio we need to make tangent line from origin of coordinate system to Hor. Speed vs Vert. speed curve

4.5.1 Turning

Parameters limitating turns:

- Structural (load factor)-$n = 4$
- Aerodynamic ($C_{L_{max}}$)-$n = \left(\frac{V}{V_S}\right)^2$
- Propulsive (thrust)-$n = \frac{C_L}{C_D} \cdot \frac{T}{W}$

Velocity [m/s]	Turning minimum radius [m] (1)
16.50	7.17
18.00	8.53
20.88	11.48
24.85	16.26
28.83	21.88
32.81	28.34
36.78	35.62
40.76	43.74
44.74	52.69
48.71	62.48
52.69	73.10
56.67	84.55

Table 15. Velocity and turning min. radius

4.5.2 Range and Endurance

Next step is to calculate Range and Endurance. In this case height is constant (2000m).

Formula for range: $R = \frac{n \cdot C_L}{g \cdot C_p \cdot C_d} \cdot ln\left(\frac{W_{TOW}}{W_{fuel}}\right)$

Formula for endurance: $E = \frac{C_L^{3/2} \cdot n}{C_D \cdot C_p} \sqrt{2 \cdot \rho \cdot S} \cdot \left(\frac{1}{\sqrt{W_{TOW}}} - \frac{1}{\sqrt{W_{fuel}}}\right)$

Speed for max endurance can be seen in req. power vs Speed graph where req. power is minimum

Speed for max range can be seen in Thrust vs speed graph, where thrust is minimum, because when there is minimum thrust there will be minimum fuel consumption.

From this graph we can determine maximum range velocity which is minimum drag in drag vs velocity. Maximum endurance will be minimum power in power required vs speed in altitude.

Speed [km/h]	Endurance at 2000m [h]	Range at 2000m [km]
59.3	1.72	102
78.3	2.37	186
97.4	2.71	264
110	2.67	311
116	2.37	322
136	1.97	305
155	1.58	274
174	1.23	238
193	0.95	200
212	0.71	163
231	0.50	126

Table 16. Endurance and range at different speed

Speed km/h	Thrust [N]
59.3	2865.641961
78.3	2471.835204
97.4	2146.218424
105	1878.138301
116	1656.941512
136	1471.974736
155	1312.584652
174	1168.117938
193	1027.921273
212	881.3413356
231	717.7248043

Table 17. Thrust

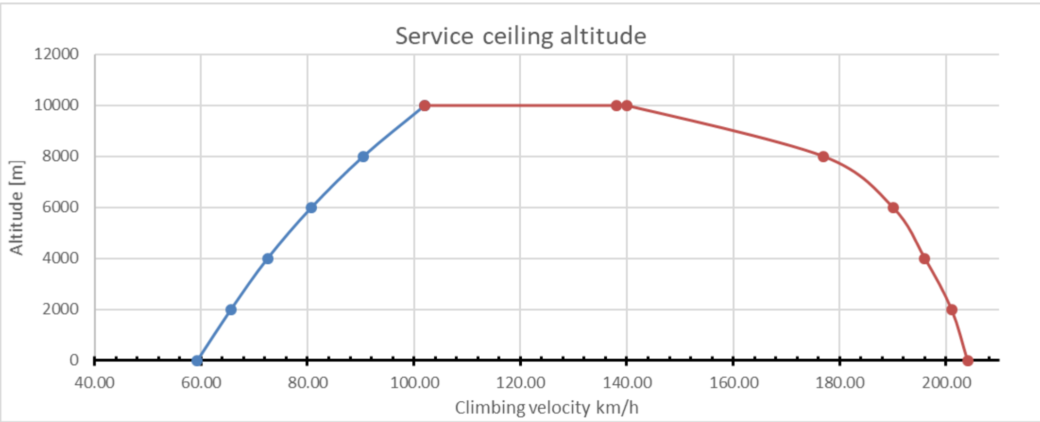

Figure 14. Service ceiling altitude

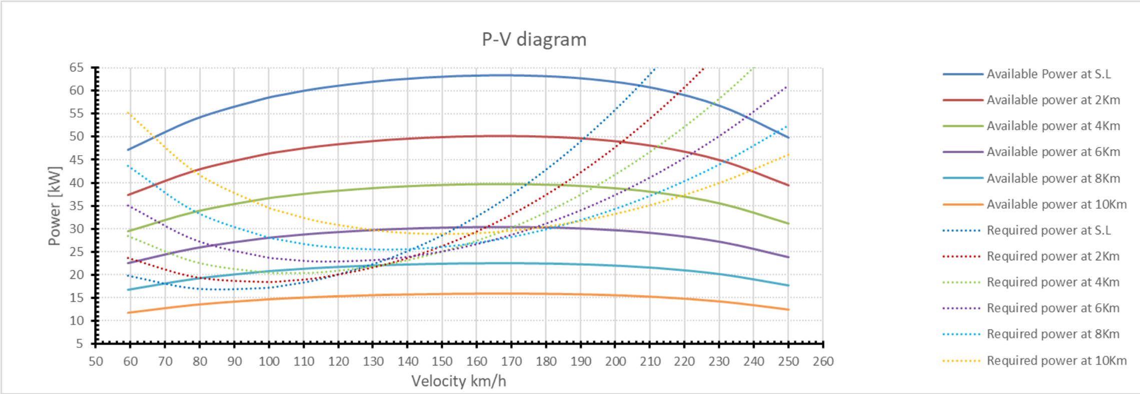

Figure 15. P-V Diagram

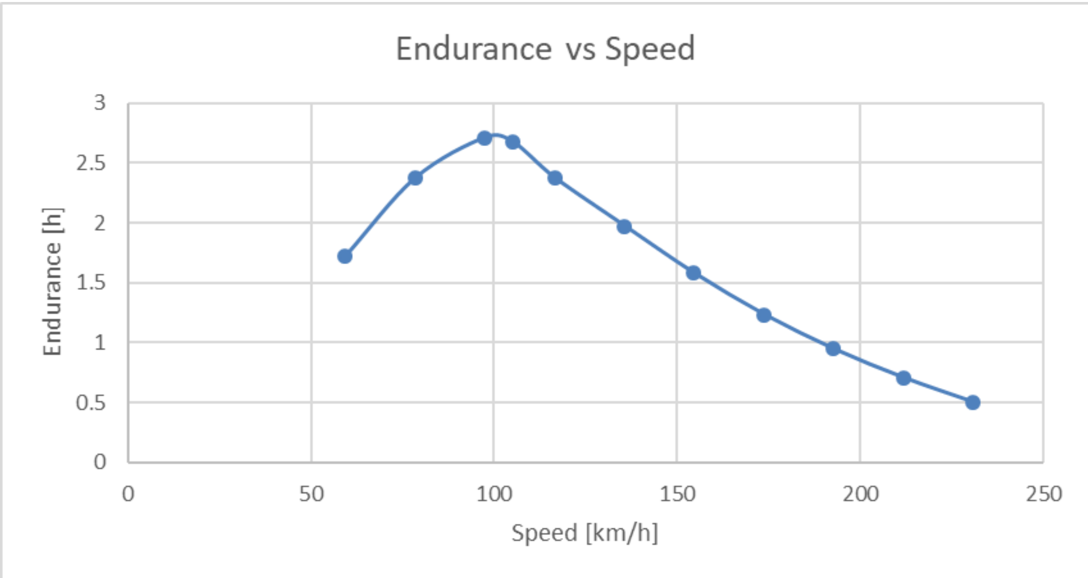

Figure 16. Endurance vs Speed

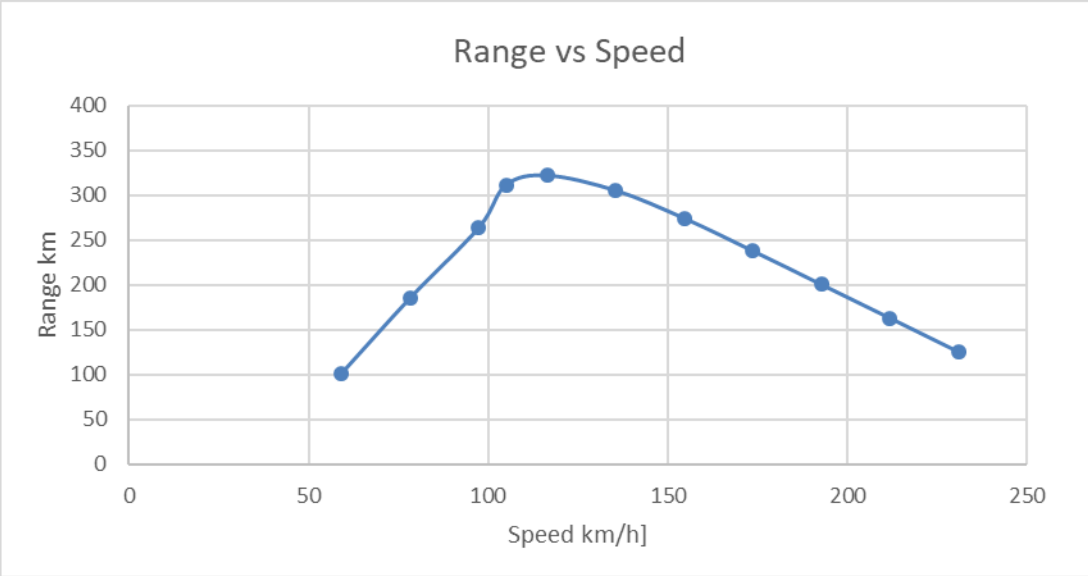

Figure 17. Range vs Speed

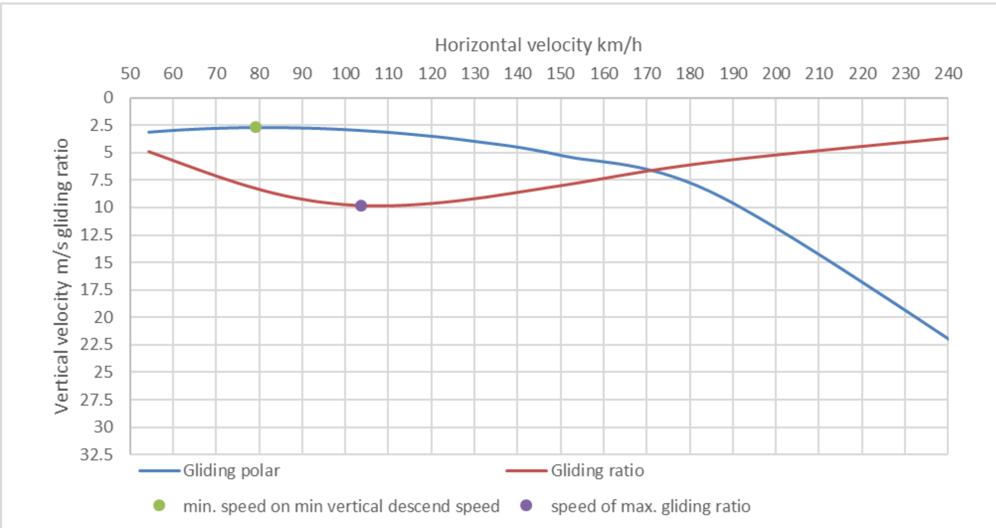

Figure 18. Gliding polar and glidi flight diagram

4.6 Flight Envelope Diagram

maneuver	calculated	choosen in knots	m/s	ft/s
Vc not less than	104.30834	105.00	54.017	177.2199
Vc not more than	116.63071			
Vc min	107.4692			
Vd not less then	150.45689	155	79.739	261.6103
Va not less then	73.674856	75.00	38.583	126.5856
Va not more Than	105.00			
Vf	49.740839	80		

Table 17. Maneuver and speed diagram

Limit maneuvering load factor for LSA category airplane:

The positive limit maneuvering load factor n_1 may not be less than 4.0.

The negative limit maneuvering load factor n_2 may not be greater than −2.0

Loads with wing flaps extended: (1) if flaps or other similar high lift devices are used, the airplane must be designed for n_1 = 2.0 with the flaps in any position up to V_F and (2) n_2 = 0.

Loads with speed control devices: (1) if speed control devices such as speed brakes or spoilers are used, the airplane must be designed for a positive limit load factor of 3.0 with the devices extended in any position up to the placard device extended speed.

n gust	4.1740226	3.342730948
	-2.174023	-1.342730948

Table 18. Positive and negative gust load

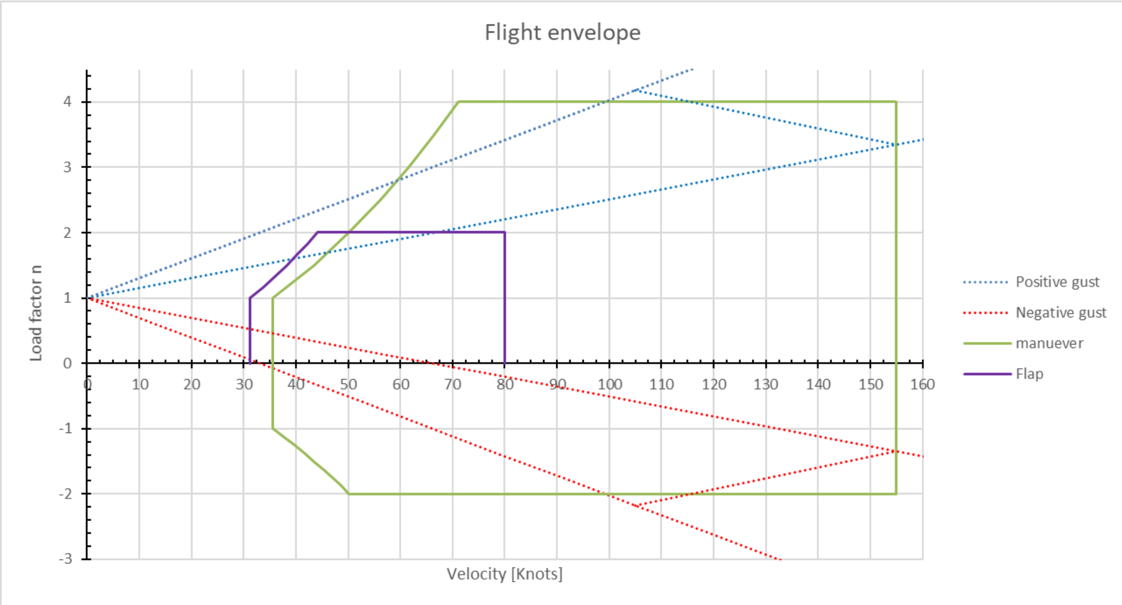

Figure 19. Flight envelope

On the figure 19 you can see flight envelope. Green curve is for maneuver, the positive gust load is blue points and n maximum is 4.17, for negative gust load (red points) n = -2

5. Engine mount CAD models

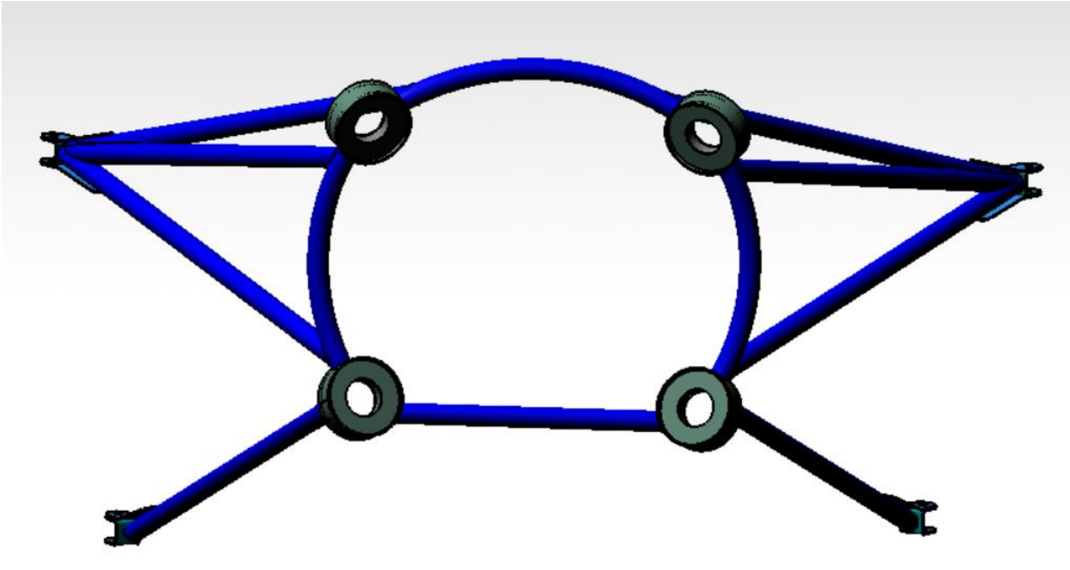

Figure 20. Engine mount front view

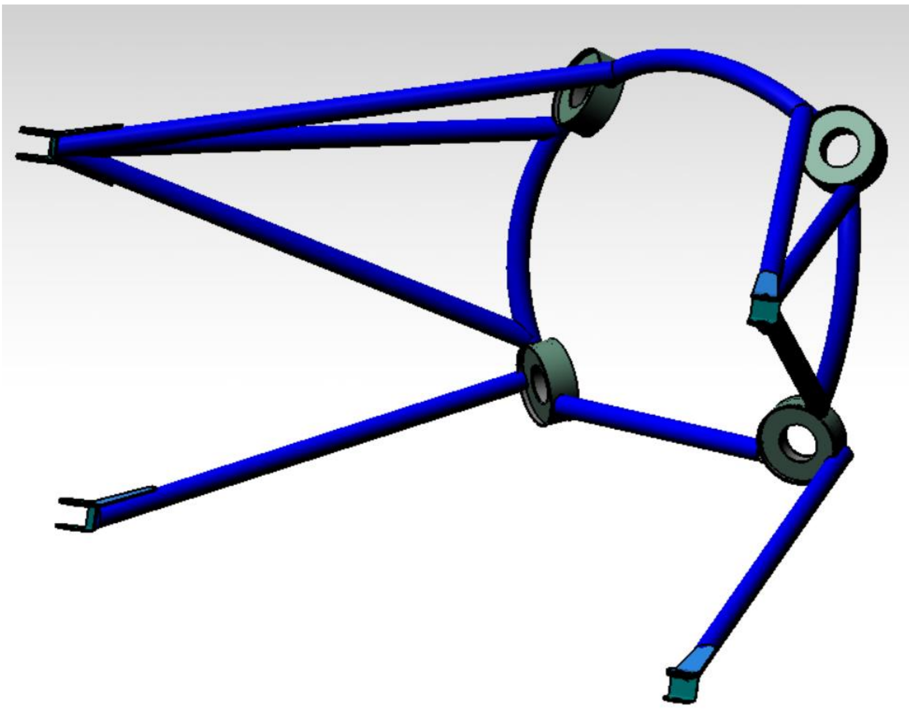

Figure 21. Engine mount back view

On the figure 21 you can see back view. From where need to be fixed on the first cross section.

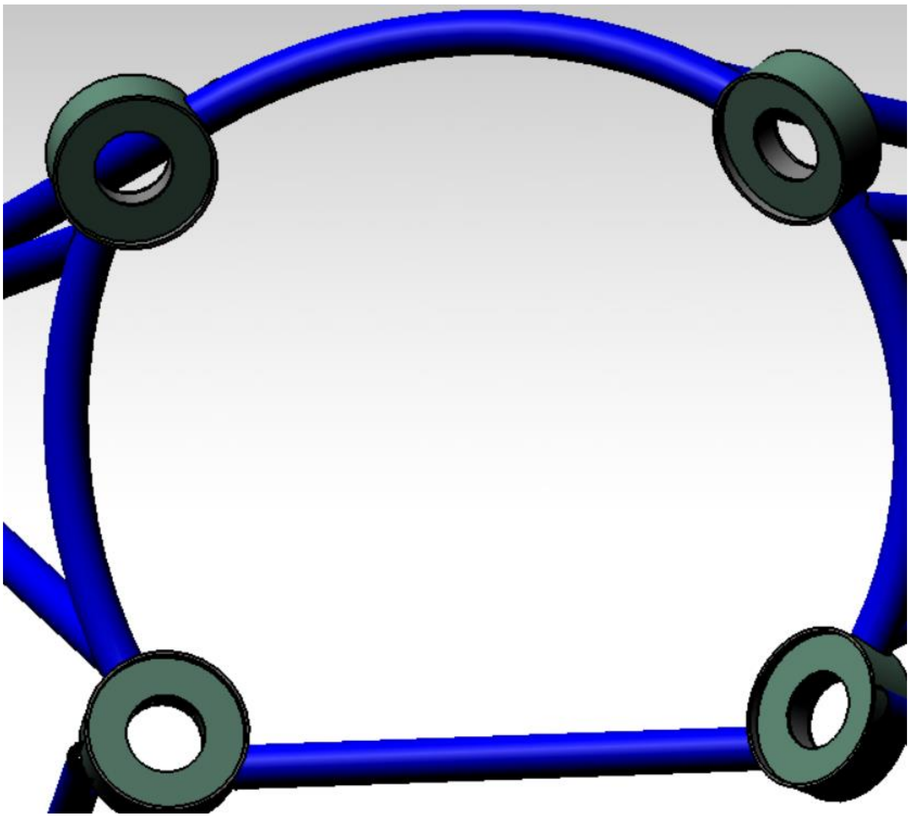

Figure 22. Engine mount main hinges

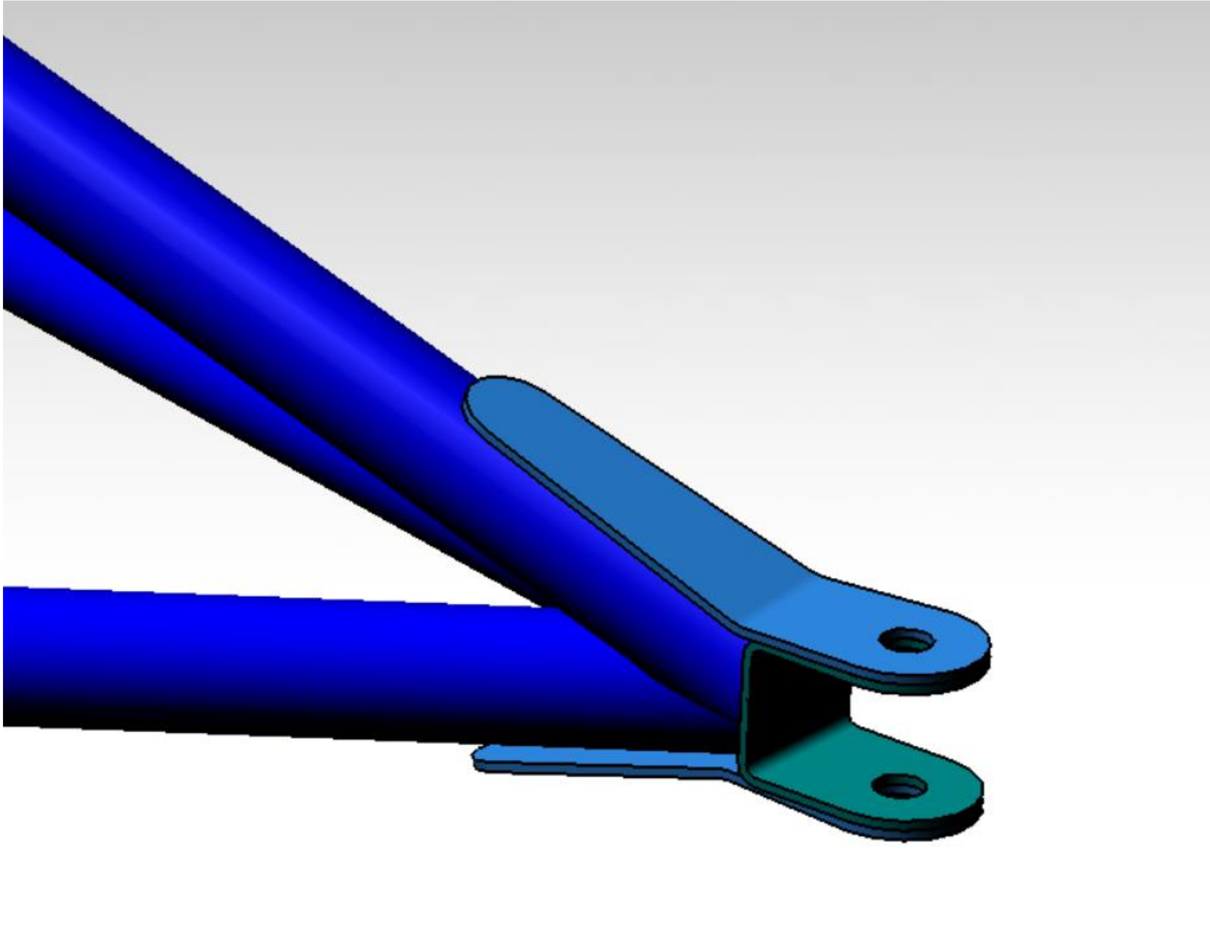

Figure 23. Engine mount hinge for fixation on the cross section

Hinge, from where need to fix engine mount on the first cross section with M8 bolts.

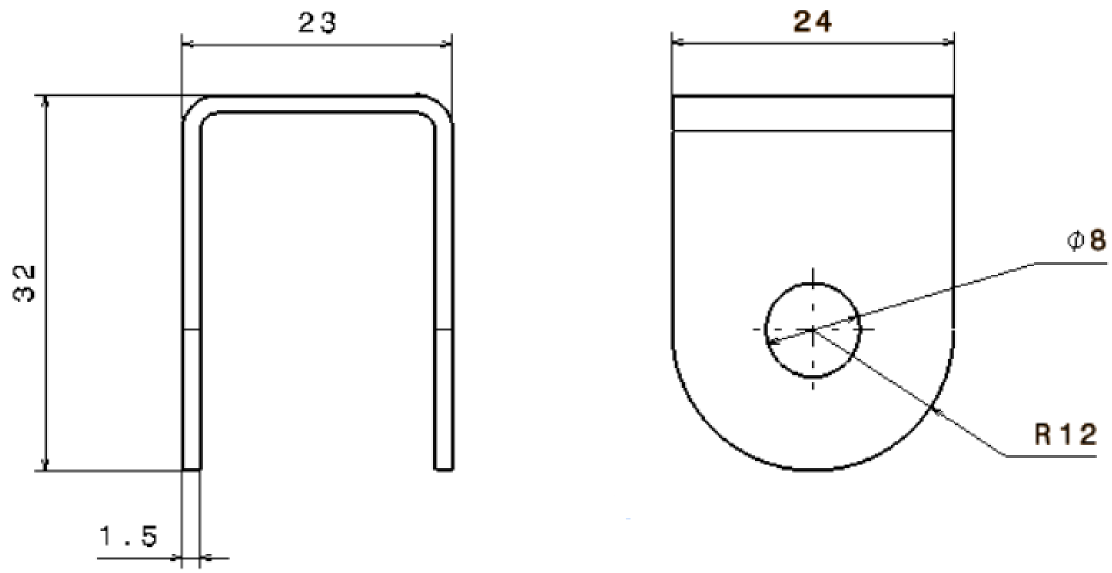

Figure 24. Hinge drawing

Note:

All tubes has a same diameter : inside radius r = 8.8 mm, thickness : 1 mm.

Engine mount material is AISI 4130 steel alloy.

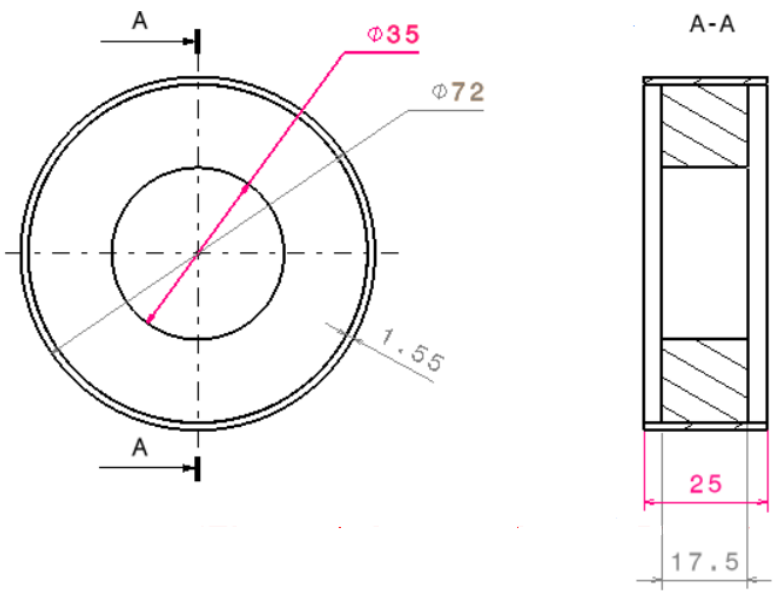

Figure 25. Engine mount main hinge.

7. Engine mount Test Plan

7.1 Introduction

The engine mount strength test plan were performed by a theoretical calculations which are based on to F-2245-04 regulations.

The resulting weight of all components of the engine is 117.5 kg.

Max. continuous power $P_{Flight} =$ 80.5 kW

Speed $V_A = 61 m/s$ = 220 km

Thrust at V_c F = 1878 N

7.2 Tensile Load

$$F_T = \frac{P_{Flight} \cdot \eta_{prop}}{V_A} = \frac{80500 \cdot 1}{61} = 1319.9\ N$$

For test plan all loads must be multiplied on safety factor – f=1.5

I load case:

Side load (lateral load) - $F_L = 1.5g$

II Load case:

Vertical load

Thrust

Torque

III Load case

Positive gust

Thrust

Torque

IV Load case

Negative gust load

Thrust

Torque

I. Load case (according to F-2245 5.2.10.1):

Lateral Load

With a load case of 1/3· n_A = 1/3 · 4 = 1.5 =n_B

F_L= $m \cdot g \cdot n_B$= 117.5· 9,81 · 1.5 =1729N

Lateral load – limit load - $F_L = 1729\ N = 176kg$

Ultimate load $F_L = 1729 \cdot 1.5 = 2593.5N = 264.4\ kg$

Load	F_L [Kg]
Limit	176
Ultimate	264.4

Table 19. Limit and ultimate load for lateral load.

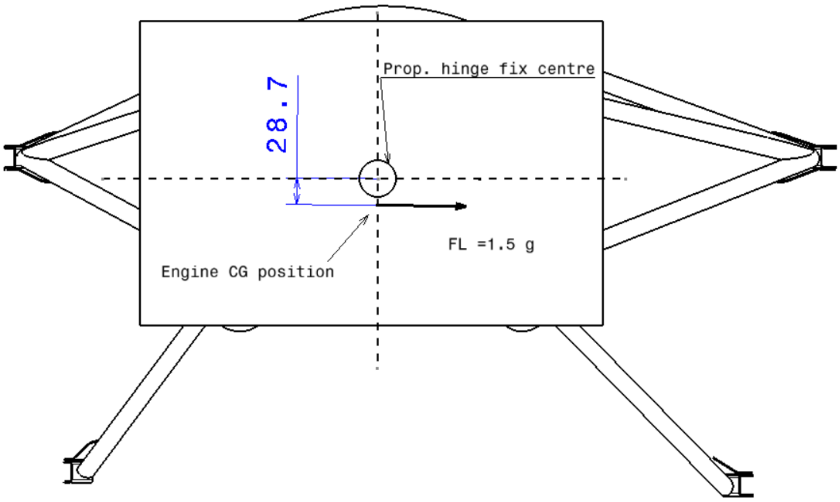

Figure 28. Engine mount test plan for lateral load

For all load cases fuselage must be install on a steel support. Engine imitation should be installed on the engine mount. Loading must be performed by a load bags and torque forces must be performed by chain.

II. Load case

Inertia load (according to the requirements of the regulation F2245 5.2.9.1)

Note:

Load in case of load factor 75% is not calculated, because according to engine specification max. continuous and take of power are same.

Load in case of load factor :

$G_A = m \cdot g \cdot n_A$ = 117.5 · 9.81 · 4 = 4610.7N

(n_A multiple according to the requirements of the regulation)

Vertical load – Limit load	$G_A = 4610.7\ N = 470\ kg$
Ultimate load	$G_{Au} = 4610.7 \cdot 1.5 = 6916N = 705kg$
Max. continuous power	P_{Flight} = 80.5 kW
Thrust at V_c	F = 1878 N

Load	GA [kg]
Limit	470
Ultimate	705

Table 20. Limit and ultimate load for vertical load.

Torque load from the engine

Engine speed at max. continuous power at standart sea level conditions

n = 2800 rpm   (2800 rpm for max. 116 hp)

According to F2245 chapter 5.2.9.3, the torque from the engine (for a 4 cylinder 4 stroke) is multiplied by a factor n=2

$$Mk_{Flight} = \frac{P_{Flight}}{2\pi n} = \frac{80500}{2 \cdot 3.14 \cdot \frac{2800}{60}} \cdot 2 = 549\ Nm$$

Torque load for flight regime – Limit load $Mk_{Flight} = 549N = 59kg$

Ultimate load $Mk_{Flightu} = 97.5kg$

Load	Mk_{Flight} [kg]
Limit	59
Ultimate	97.5

Table 21. Limit and ultimate load for torque.

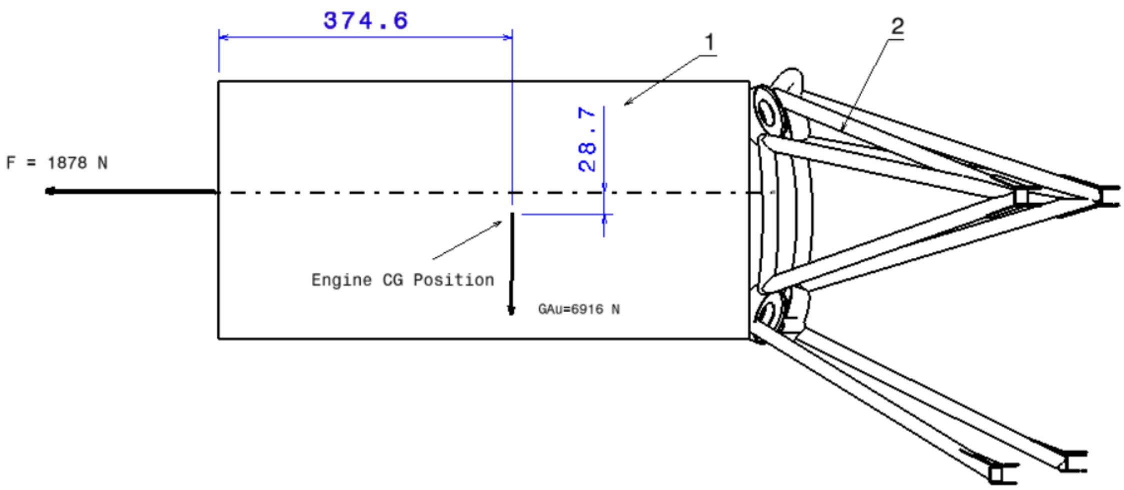

Figure 29. Engine mount test plan for vertical load

1) Engine imitation

2) Engine mount

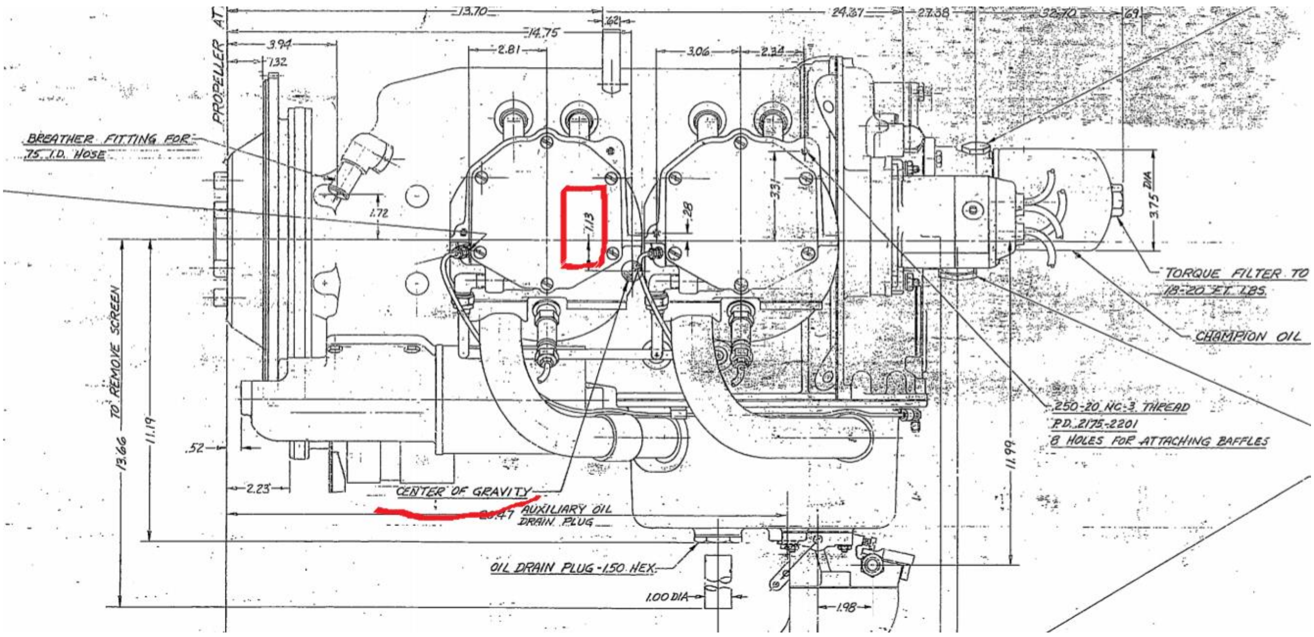

Figure 30. Lycoming O-235 Center of Gravity position from side [15]

On the figure 24 is marked Lycoming O-235 Cg position for side view which is 1.13 inch - 28.7 mm ($\pm 0.25\ inch$) below the centerline of the crankshaft and 14.75 inch - 374.6 mm ($\pm 0.25\ inch$) from the front face ot the propeller mounting flange

III. Load Case

Positive gust load:

$$F_{gp} = (m_{engine} + m_{prop}) \cdot g \cdot n_A = (117.5 + 6.8) \cdot 9.81 \cdot 4 = 4877.5\ N$$

Positive gust load - limit load $F_{gp} = 4877.5\ N = 497\ kg$

Ultimate load $F_{gp} = 4877.5 \cdot 1.5 = 7316\ N = 746kg$

Torque load for flight regime – Limit Load $Mk_{Flight} = 549N = 59kg$

Max. continuous power P_{Flight} = 80.5 kW

Thrust at V_c F = 1878 N

Load	F_{gp} [kg]
Limit	497
Ultimate	746

Table 22. Limit and ultimate load for positive gust load.

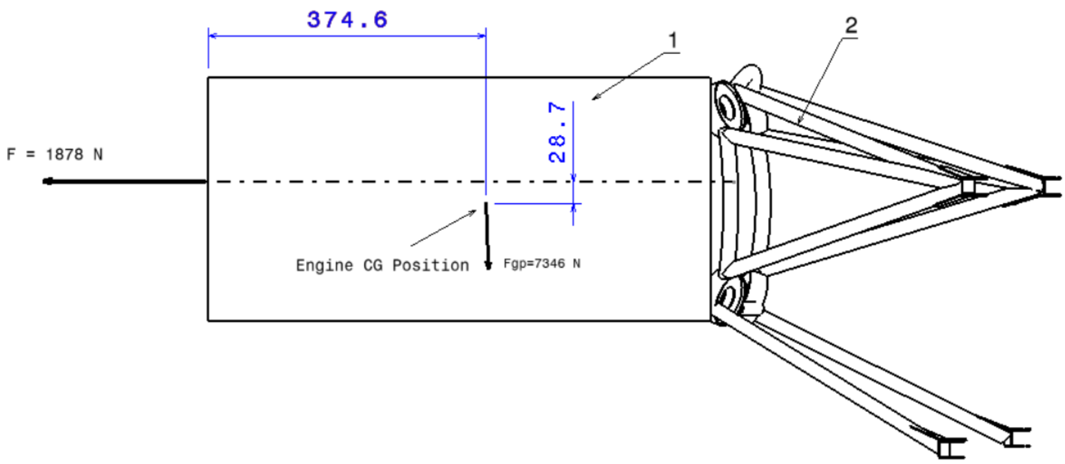

Figure 31. Engine mount test plan for positive gust load

1) Engine imitation

2) Engine mount

$$a = \frac{Mk_{Flight}}{F_{gp}} = \frac{823}{7316} = 0.11 \text{ m}$$

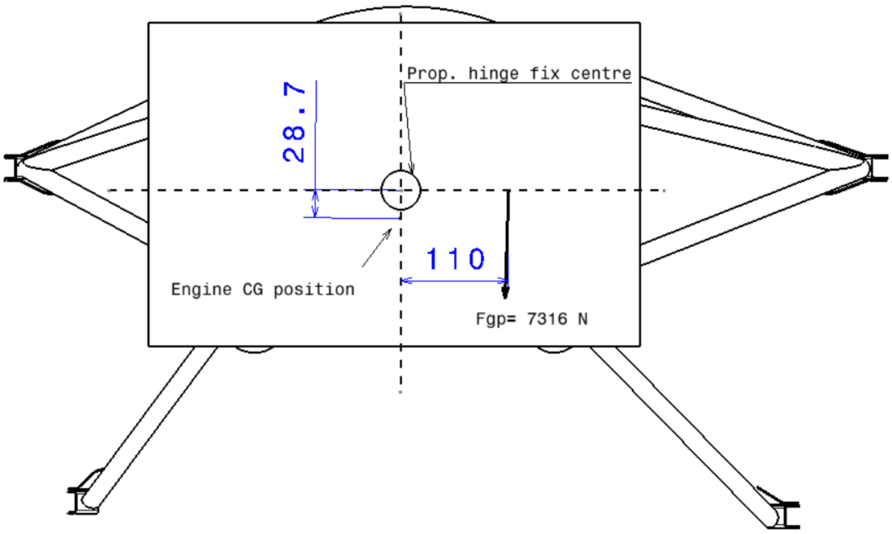

Figure 32. Engine mount test plan for positive gust load (front view)

IV. Load Case

Negative gust load:

$$F_{gn} = (m_{engine} + m_{prop}) \cdot g \cdot n_A = (117.5 + 6.8) \cdot 9.81 \cdot (-2.3) = -2796.7$$

Negative gust load - Limit load $F_{gn} = -2796.7N = -285\ kg$

Ultimate load $F_{gnu} = -2796.7 \cdot 1.5 = -4195N = -427.7kg$

Torque load for flight regime – Limit Load $Mk_{Flight} = 549N = 59kg$

Ultimate load $Mk_{Flight} = 549 \cdot 1.5 = 823.5N = 87.5kg$

Max. continuous power P_{Flight} = 80.5 kW = 80500 Nm/s

Thrust at V_c F = 1878 N

Load	F_{gn} [kg]
Limit	285
Ultimate	427.7

Table 23. Limit and ultimate load for negative gust load

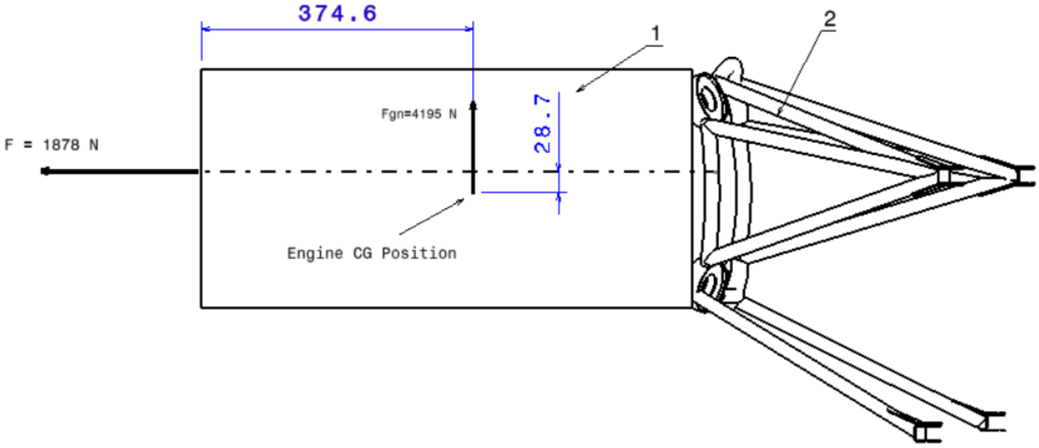

Figure 33. Engine mount test plan for negative gust load

$$a = \frac{Mk_{Flight}}{F_{gn}} = -\frac{823}{4195} = -0.19 \text{ m}$$

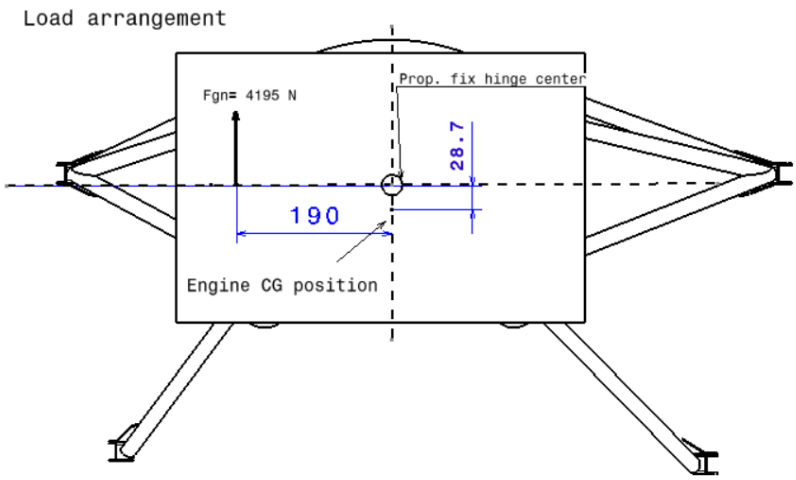

Figure 34. Engine mount test plan for negative gust load (front view)

Conclusion

“Czech sport aircraft” needed to change engine on aircraft PS-28 Sport cruiser, which would be more powerfull.

Curent engine was Rotax 912 ULS, which is very common aircraft 4 cylinder engine with 100 hp.

Because of this company choosed Lycoming O-235 also 4 cylinder engine with 118-120 hp.

The main problem was weight, which is in total 117.5 kg (twice more than Rotax).

According to EASA, LSA max. take of weight should be 600kg, in our case only powerplant was 117.5 kg, only construction was 311 kg. So I decided to made some changes, which means, that I canceled wing lockers and decreased baggage comparision in the aircraft, also decreased fuel amount and all these gave me possibility to fixed Lycoming in the aircraft.

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