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Structural Design of Fuselage

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Abstract- Aircraft structural design initiates with the consideration of major components such as wings, fuselage and empennage and with a formation of limits of maximum load factor at any flight velocity that an aircraft can withstand i.e., the flight envelope or V-n diagram. Fuselage is the main structural component that carries most of the weight and provides the connection to other component. The current work includes structural design of the light sports aircraft (LSA) single seater for the requirements of +6/-5 G limit loads constraints. The objective of the paper is the structural design of fuselage for the given requirements. Structural design is done such that it should have the enough strength even when the aircraft is performing high angle of attack i.e. at 6g and at negative angle of attack i.e. at 5g performance. Design procedure was followed by each parts structural designing. The geometry of the fuselage is considered as simplified hollow tube without any cut-outs and divided into bays and skin panels. By modelling the aerodynamic, gravity, ground reaction forces and internal pressure a free body diagram and force/moment distribution is created for several flight and ground load cases, like +6/-5 G flight, lateral gust or landing load cases. The critical load cases are used for analysis.

Keywords—Light sport aircraft, Fuselage Structure, stringers, frames

I. INTRODUCTION

With the advent of aircrafts from first controllable human flight by Wright brothers to the present decade there is a tremendous progress in the aviation industry. Light sport aircraft is one of the classes of the aircraft which has FAA regulation restriction on weight and performance. Light Sport Aircraft (LSA) is a small aircraft that flies in a small range with maximum of two passengers. It has less weight, simple-to-operate, low speed and easy-to-fly compared to other commercial aircrafts. The current work is based on the structural design of the Light Sport Aircraft carried out based on the Conceptual and Aerodynamic design of single seater light sport aircraft.

Wings and fuselage are the significant components of the aircraft to be considered while designing an aircraft structurally. Fuselage is the main body of the plane that carries most of the weight, and provides the structural connection to the wing and tail. The fuselage of any aircraft undergoes different types of stresses so it has to be built properly with reasonable safety factor. Due to the presence of high number of equipments in the fuselage, thus it is essential to give adequate number of cut-outs for access and inspection purposes.

These cut-outs and discontinuities result in fuselage design being more complicated, less precise and often less efficient in design. Fuselage being a common structural member of attachment to other significant component of the aircraft transferring the loads, hence it is considered as a long hollow beam. The reaction forces generated by the wing, tail or landing gear are considered as concentrated loads with respective attachment points. The inertial forces from the weight of the fuselage structure and other structural members of fuselage provide the balancing reaction forces.

II. RESEARCH BACKGROUND

Bruhn (1990) has published a detailed procedure regarding structural design of wing, and fuselage. The same procedure was followed for structural design of wing for this current design. The flange thickness of the stringer was chosen. The procedure to calculate the number of stringers and the procedure to calculate margin of safety in bending was followed.

Raymer (1992) has published the sizing of frames and its spacing for the different type of aircrafts. These published values were taken in consideration to locate the frames in the fuselage of the present work.

Megson (2007) has published the detail procedure to calculate the shear flow analysis of skin the same procedure was followed and also the same procedure was followed to calculate critical stress.

Sadraey (2012) has published method to estimate the margin of safety in bending and the procedure to calculate the loads acting on fuselage were taken in consideration in the present work.

Peery (2011) has published the detailed procedure to consider the loads acting on fuselage which was followed.

III. STRUCTURAL DESIGN OF FUSELAGE

1. Material:

7075 T3 and heat treated aluminium alloy is selected for multipurpose light sport aircraft. Aluminium alloys are most commonly used materials in the modern aviation industry. Though these alloys have low structural efficiency compared to steel and also less dense. This is an advantage when they are used for stringers which must sustain bending loads. The engine and the landing gear is mounted and attached in the fuselage respectively which further increases the complexity of the fuselage design.

2. Bulkheads:

The fuselage structure consists of a series of frames which gives the fuselage cross-sectional shape. The fuselage is divided into various intervals at which these bulkheads are placed. Skin transfers the loads to the bulkheads skin acts as a transferring member, Bulkheads carry the concentrated loads as a result of self-weight of power plant, landing gear, wing and tail.

3. Stringers:

These are the longitudinal members which resist axial and bending loads along with skin. . Skin buckling strength is increased by stringers dividing panels into smaller sections. The stringers divide the skin into small panels thereby increasing its buckling strength. The cross-section of the stringer chosen is Z type. The distribution of these stringers on the cross section is explained in the following stages.

4. Skin:

It covers the whole body and makes the external surface smooth to enable uniform flow over the fuselage. It is connected to bulkheads by means of rivets and forms impermeable aerodynamic surface. It transmits aerodynamic forces to frames and stringers. Skin resists shear torsion loads with frames and axial loads with stringers.

IV. STRUCTURAL LAYOUT

The structural analysis of the fuselage is less complex due to symmetrical loading and crossing. Shear load is the significant load acting on the fuselage since it is transferred to the skin of the fuselage from the load acting on the wing. The structural design of fuselage begins with shear force and bending moment diagrams for the respective members. The maximum bending stress produced must be less than the material yield stress chosen for the particular member.

A. Design of fuselage

The primary design objective of fuselage of the current design is to accommodate the payload (pilot and two life jackets), sufficient amount of fuel required for flight and to provide space for wing and empennage attachments. By looking at the fuselage configurations of the existing aircrafts of the same class, the fuselage configuration considered for the current design is represented in figure Fig.1 The geometry of the fuselage was estimated based on the cockpit location and moment arm length required to generate tail moment. Overall fuselage length (Lf) depends on two parameters; namely the fuselage width (W) and fuselage height (H).

Fig.1. Fuselage configuration [4]

B. Loads and its distribution

To find out the loads and their distribution self-weight of different components housed in the fuselage are considered

Weight of the fuselage

- Engine weight
- Wing
- Weight of the horizontal and vertical stabilizers
- Weight of crew
- Payload
- Landing gear

To obtain the exact picture of shear force and bending moment on the fuselage it is necessary to distribute the weights. The length of the fuselage is divided into various stations and nose is considered as the fixed point from which the distance to each station is taken and also it is considered to coincide with location of bulkheads since the bulkheads carry the concentrated loads. The load experienced by the fuselage at high angle of attack is 6 times the actual load. The fuselage is held in equilibrium by landing gear reactions at the landing gear attachment which are calculated and transferred to the adjacent stations.

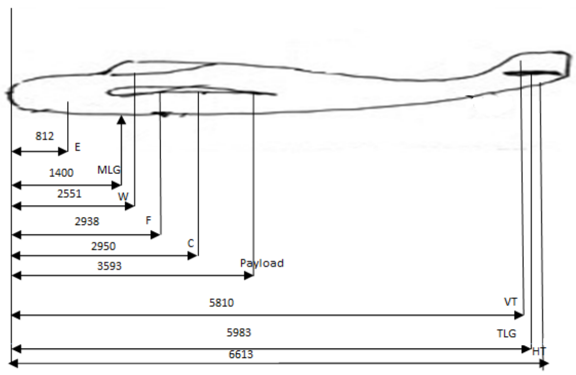

Fig.2. Balance diagram showing loads acting on fuselage

C. Shear force and bending moment

To calculate shear force and bending moment the fuselage is considered as free-free beam. To keep the fuselage in equilibrium wing reactions are determined the front spar reaction at a distance of 2225 mm from the nose is about 5672.91 N and the tail landing gear reaction at a distance of 3964.6 mm from the nose of the fuselage is 1232.23 N. using these landing gear reactions shear force and bending moment is calculated. Max shear force is 3300.37 N and max bending moment is -4.3379*10^6 N-mm. Shear force and bending moment diagram is as shown in the Fig. 3. From the survey the bending moment and shear force diagram for the aircrafts is as shown in Fig.4. The bending moment and shear force diagram for LSA as shown in Fig.3 pattern matches well with Fig.5 so the exact shape of SFD and BMD is obtained.

Wing reactions are calculated as follows

$$\Sigma F = 0$$

$$-2746.8 - 553.57 + F_S - 1263.8 - 804.42 - 981 - 66.75 + R_S - 245.25 - 20.75 - 212.8 = 0$$

$$F_S + R_S = 6905.14$$

$$\Sigma M_{F_S} = 0$$

$$-2746.8 * 1413 - 553.57 * 825 + 1266.8 * 326 + 804.42 * 713 + 981 * 725 + 66.75 * 1368 - R_S * 1739.6 + 245.25 * 3585 + 30.75 * 3758 + 212.8 * 4388 = 0$$

$$R_S = 1232.23\ N$$

Therefore, $$F_S = 5672.91\ N$$

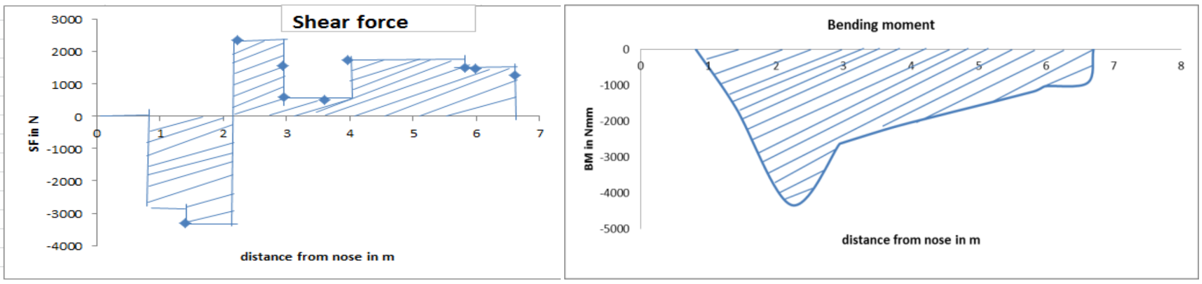

Fig.3. Shear force and Bending moment on the fuselage (free-free beam with one reaction at its centre) at fully loaded condition

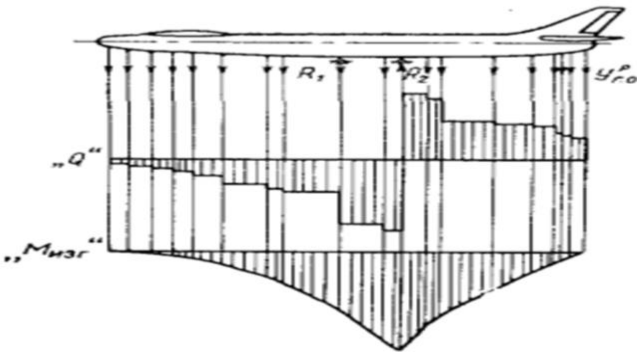

Fig.4. Shear force and the Bending moment [4]

D. Stringer Design

The fuselage design can be initiated by considering maximum bending moment as design bending moment. The cross-sectional area required to withstand the bending stress is found out by using the formula for bending stress. This area is sectioned into several stringers, spaced evenly. The stringers spacing is calculated by considering the buckling portion between adjacent stringers which can be modelled as a plate. The first step is to estimate the required cross-sectional area of the stringers.

Using the formula of bending stress,

$$\sigma = \frac{M * y}{I}$$

$$I = A\left(\frac{d}{2}\right)^2$$

$$C = 2(h + w)$$

A stringer cross section (Z section) is chosen from Bruhn analysis and design of flight vehicle structure satisfying the condition that the actual stress is less than the yield stress of the material. The total circumference of the fuselage cross section is found to be 4900 mm which is distributed with total number of 20 stringers such that the total bending moment is taken up by these stringers effectively and which satisfies the condition. The distribution of stringers around the cross-section of the fuselage is as shown in the above Fig.5.

The properties of the stringer section considered are as follows

Length of stringer l_s=9t

Height of stringer h_s=6t

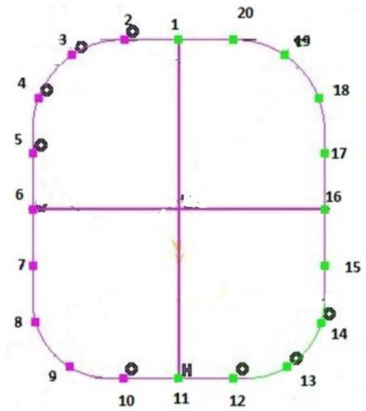

Fig.5. Location of Z shaped Stringer in the fuselage

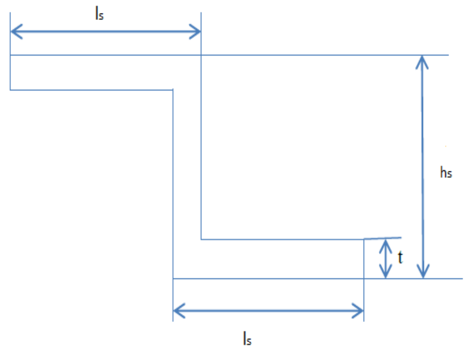

Fig.6. Z Stringer Cross-section

E. Frame Design

Frames major function is to provide end restraints for skin panels and to resist plane deflections when the skins try to buckle. The objective of frame is to select a material and cross sectional area, and spacing to make the local buckling of the skins as likely as the global buckling of the entire structure. Frame spacing can have a substantial impact on compressive skin panel design. The weight of frames and flooring are also affected by frame spacing. The number of frames selected such that it doesn't buckle under 6g load which is achieved by satisfying the condition critical stress must be less than ultimate stress of the material.

$$\sigma_{cr} = \frac{k\pi^2 E}{12(1-\nu^2)}\left[\frac{t}{b}\right]^2$$

$$k = \left(\frac{mb}{a} + \frac{a}{mb}\right)^2$$

a = (Frame spacing), b = (Stringer spacing), m = 1 [Ref 1]

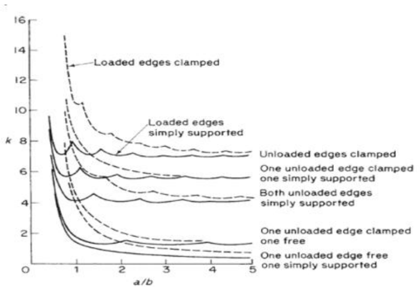

Fig.7. Buckling coefficients for the values of a/b

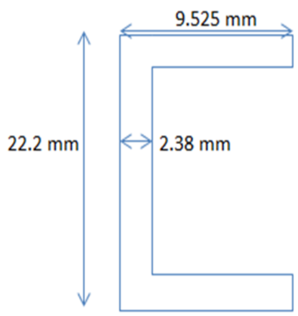

Fig.8. Cross-section of frame

Iterations were carried out to decide the number of frames as shown in Table 1. 20 frames would resist the buckling at 6g force. The type of cross-section chosen is of C-section type from Bruhn analysis and design of flight vehicle structures whose properties are shown in Fig.7.

Table 1 Calculation of Frame Numbers

Frame No	a(mm)	b(mm)	K	t(mm)	Critical Stress	FOS*6g	Total critical stress	Ultimate stress(MPa)
5	1440	213.37	47.56	2.38	383.08	9	3447.75	402.21
10	720	213.37	13.47	2.38	108.51	9	976.63	402.21
15	480	213.37	7.25	2.38	147.76	9	1329.84	402.21
20	360	213.37	5.19	2.38	41.86	9	376.75	402.21

F. Shear Flow Analysis

The estimation of the shear flow distribution in the skin due to shear is basically an analysis of a idealized single cell closed section beam. Considering stringers are numbered in anticlockwise direction and to determine the shear flow between the two stringers a cut is made and calculated using the formula and also effects of shear and torsion are included simultaneously

$$q_s = -\left(\frac{S_x I_{xx} - S_y I_{xy}}{I_{xx} I_{yy} - {I_{xy}}^2}\right)\sum Ax - \left(\frac{S_y I_{yy} - S_x I_{xy}}{I_{xx} I_{yy} - {I_{xy}}^2}\right)\sum Ay + q_{s,0}$$

$$S_x = 0,\ S_y = -3300.37\ N, I_{xy} = 0$$

Therefore equation reduces to

$$q_s = -\frac{S_y}{I_{xx}}\sum Ay + q_{s,0}$$

$$q_b = -\frac{S_y}{I_{xx}}\sum Ay$$

Factor of safety = 1.5

Load factor = 6

$$q_b = \frac{3300.37 * 9}{869240.5063}\sum Ay$$

Cutting one of the skin panels between stringers 1 and 2 and determine shear flow and calculate q_b. The results shown in table 2; column 2 represents the boom crossed when the analysis moves from one panel to the next panel.

$$\sum q_b * l = -4.5e12\ N$$

Table 2 Tabulation of shear flow

Sl.no	y(mm)	A*Y	ΣAY	q_b	q_b *l	q_S
1	213.379	628.337	0	0	0	91.8765
2	426.758	1256.674	1256.674	-28765267.9	-6.1E+09	-2.9E+07
3	640.137	1885.011	3141.686	-71913192.5	-1.5E+10	-7.2E+07
4	853.516	2513.349	5655.034	-129443728	-2.8E+10	-1.3E+08
5	1066.895	3141.686	8796.72	-201356921	-4.3E+10	-2E+08
6	1280.274	3770.023	12566.74	-287652679	-6.1E+10	-2.9E+08
7	1493.653	4398.36	16965.1	-388331139	-8.3E+10	-3.9E+08
8	1707.032	5026.697	18221.78	-417096544	-8.9E+10	-4.2E+08
9	1920.411	5655.034	23876.81	-546540181	-1.2E+11	-5.5E+08
10	2133.79	6283.371	30160.18	-690366520	-1.5E+11	-6.9E+08
11	2347.169	6911.709	37071.89	-848575562	-1.8E+11	-8.5E+08
12	2560.548	7540.046	44611.94	-1021167307	-2.2E+11	-1E+09
13	2773.927	8168.383	52780.32	-1208141525	-2.6E+11	-1.2E+09
14	2987.306	8796.72	61577.04	-1409498446	-3E+11	-1.4E+09
15	3200.685	9425.057	71002.1	-1625238069	-3.5E+11	-1.6E+09
16	3414.064	10053.39	71002.1	-1625238069	-3.5E+11	-1.6E+09
17	3627.443	10681.73	81683.83	-1869742869	-4E+11	-1.9E+09
18	3840.822	11310.07	106817.3	-2445047997	-5.2E+11	-2.4E+09
19	4054.201	11938.41	118755.7	-2718317973	-5.8E+11	-2.7E+09
20	4267.58	12566.74	127552.4	-2919674436	-6.2E+11	-2.9E+09
		131322.5		-2.0452E+10	-4.4E+12	

Taking into consideration as cell twist is zero for the fuselage cross section,

$$-4.5e12 + 4900\ q_{s,0} = 0$$

The constant shear flow added to the cell

$$q_{s,0} = 91.8765\ N/mm$$

The critical shear flow is found to occur in elements between 1 and 11, 6 and 20. The critical shear flow value is 91.8765 N/mm

$$\tau_{cr} = \frac{\pi^2 kE}{12(1-v^2)}\left(\frac{t}{b}\right)^2$$

$a = 360\ mm$ (Frame spacing)

$b = 213.319\ mm$ (Stringer spacing)

$E = 7.17 * 10^4\ MPa$

$\frac{a}{b} = 2.99$

$K_s = 40$

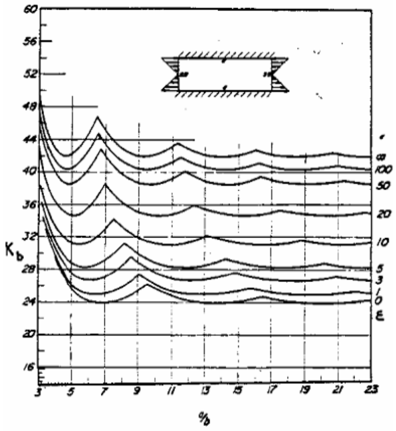

Fig.9. Shear buckling coefficient for values of a/b

Thus we obtain, t=1.27 mm

The skin thickness is thus found to be t=1.27 mm

A fuselage cross-section is a closed section beam. The shear flow distribution produced by a pure torque is given by

$$q = \frac{T}{2 * A}$$

The distribution of shear flow produced by the applied torque

$$q = \frac{T}{2 * 1495000}$$

$$T = 274.71 * 10^6 \text{ N/mm}$$

This value of shear flow is acting in anticlockwise direction around the cross-section. To calculate the stresses acting in each stringer boom areas are calculated

$$B_1 = A + \frac{t_s * d_s}{6}\left(2 + \frac{\sigma_2}{\sigma_1}\right) + \frac{t_s * d_s}{6}\left(2 + \frac{\sigma_{20}}{\sigma_1}\right)$$

B1=B2=B20, B3=B9, B4=B18, B5=B17, B8=B14, B9=B13, B10=B11=B12

Table 3 Stringer boom area

Stringer No	Y (sigma)	B boom area (mm^2)	M_x (N-mm)	Stress
1,2,20	650	274.27	39041100	19.06
3,19	591.58	270.07	39041100	17.35
4,18	424.97	273.49	39041100	12.46
5,17	212.847	187.84	39041100	6.24
7,15	217.5	190.84	39041100	6.38
8,14	434.134	278.17	39041100	12.73
9,13	606.298	273.14	39041100	17.78
10,11,12	650	279.79	39041100	19.06

G. Margin of safety

$$\sigma_{b_{max}} = \frac{M * y}{I_{NA}} = 303.9$$

$$MS = \frac{\sigma_{allowable}}{\sigma_{b_{max}}} - 1$$

$$= \frac{402}{303.9} - 1 = 0.322$$

$$MS = 32\%$$

Margin of safety of stringers is found be 32% which means the selection of number of stringers to withstand 6g forces are sufficient.

H. Weight estimation

The total weight of the fuselage along with stringers and frames is 972.53 N

The weight is calculated using the formula

v=A*l

m=ρ*v

Where v – volume of the fuselage in mm3

A-area of the cross-section

l- Length of the fuselage

IV. CONCLUSION

Margin of safety in bending = 3.7

Number of stringers = 20

Number of frames = 20

Skin thickness = 1.27 mm

In the selection of stringers and skin thickness maximum bending moment while landing with c.g the maximum aft position was considered. Hence the bending moment is higher on the front section. Margin of safety is found be 32% to resist the 6g forces. The 20 number of stringers selected are capable of resisting the forces acting on it, hence the structure is safe.

Number of stringers and number of frames is of the same order and spacing which is implanted in existing aircrafts. To have lesser number of strong members is disadvantageous, because it is better to have more number of structural members.

Nomenclature

Thickness of the stringer - t

Tensile strength of the material -σ

Design bending moment- M

Second moment of area - I

y = d/2

d - Diameter of the fuselage

A - Cross-sectional area of the fuselage stringers

K- Buckling coefficient

q_b - Open section shear flow

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