STRESS ANALYSIS OF AIRCRAFT FUSELAGE STRUCTURES

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الملخص

تتناول هذه الورقة كتابة برنامج حاسوب باستخدام لغة "بايثون"، لحساب الخصائص الهيكلية المتمثلة في مركز الثقل، وعزم القصور الذاتي ولإيجاد إجهادي القص والثني لمقطع الهيكل.

تسمح الصيغة المعممة للبرنامج بإجراء تحليل أي مقطع لجسم طائرة مكون من قشرة واضلاع طولانية سواء كان دائريًا أو غير دائري، متماثلا أو غير متماثل. وقد اخُتِير مقطعين لجسم طائرة كدراستي حالة من مصدر منشور وذلك لغرض التحليل والتحقق من نتائج البرنامج. وتم التحقق أيضًا من دقة البرنامج عن طريق تحليل نموذج العناصر المتناهية باستخدام البرنامج التجاري (MSC/PATRAN 2004) مع (MSC/NASTRAN 2004) لأحد دراستي الحالة المختارتين.

يتكون نموذج جسم الطائرة من قشرة واضلاع طولانية من الألومنيوم تكون خاضعة لأحمال القص واللي وتم تمثيل القشرة باستخدام عنصر صفيحة رُبعية القص نوع (CQUAD4) وتمثيل الاضلاع باستخدام عنصر قضيبي (Bar). كما طبقت الشروط الحدية الكابولية على النموذج مع استعمال أسلوب التقييد متعدد النقاط وذلك بتطبيق عنصر صلب من نوع (REB2) عند موضع تأثير الحمل وذلك للتأكد من أن الأحمال على مقطع الجسم تؤثر بالتساوي دون التسبب في إضافة أي صلابة له. وجد ان نتائج البرنامج الحالي متوافقة بشكل جيد مع النتائج النظرية ونتائج المنشورة في البحوث الاخرى.

ABSTRACT

This paper deals with developing a computer program using Python language, to calculate the structural properties, bending and shear stresses of the aircraft fuselage section. The structural properties are in the form of the center of gravity and moment of inertias. The generalized formulation allows performing the analysis of the circular and non-circular fuselage sections with skins and multiple stringers. Two aircraft fuselage sections from the open literature are selected as case studies in the analysis and validation of the developed program. The developed program is validated also with the finite element model generated and analyzed by the commercial finite element software, MSC/PATRAN 2004 and MSC/NASTRAN 2004 respectively for one case study of the fuselage section. The fuselage model has consisted of skin and stringers made from aluminum materials and subjected to shear and torsional loads. The fuselage skin is modeled using CQAD4 shear panel elements and Bar elements for the stringers. Cantilever boundary condition is implemented to the fuselage model. The Multi-Point Constrained, MPC is used by the application of rigid element, REB2 at the location of the applied load. This is to make sure that the loads in the section are equally applied without adding any stiffness to the fuselage model. The results of the program are in good agreement with theoretical and fuselage model results available in open literature.

KEYWORDS: Fuselage Stresses; Centre of Gravity, Stiffness's; Bending; Shear Stresses; Finite Element.

INTRODUCTION

When designing an aircraft, it is all about finding the optimal proportion of the weight of the vehicle and payload. It needs to be strong and stiff enough to withstand the exceptional circumstances in which it has to operate. Durability is an important factor. In addition, if a part fails, it does not necessarily result in failure of the whole aircraft, [1-3].

The main sections of an aircraft, the fuselage, tail, and wing, determine its external shape. The load-bearing members of these main sections, those subjected to major forces, are called the airframe. In transport aircraft, the majority of the fuselage is cylindrical or near-cylindrical, with tapered nose and tail sections. The semi-monocoque construction, which is virtually standard in all modern aircraft, consists of a stressed skin with added stringers to prevent buckling, attached to hoop-shaped frames floor beams and pressure bulkheads as shown in Figure (1) [2,4-5].

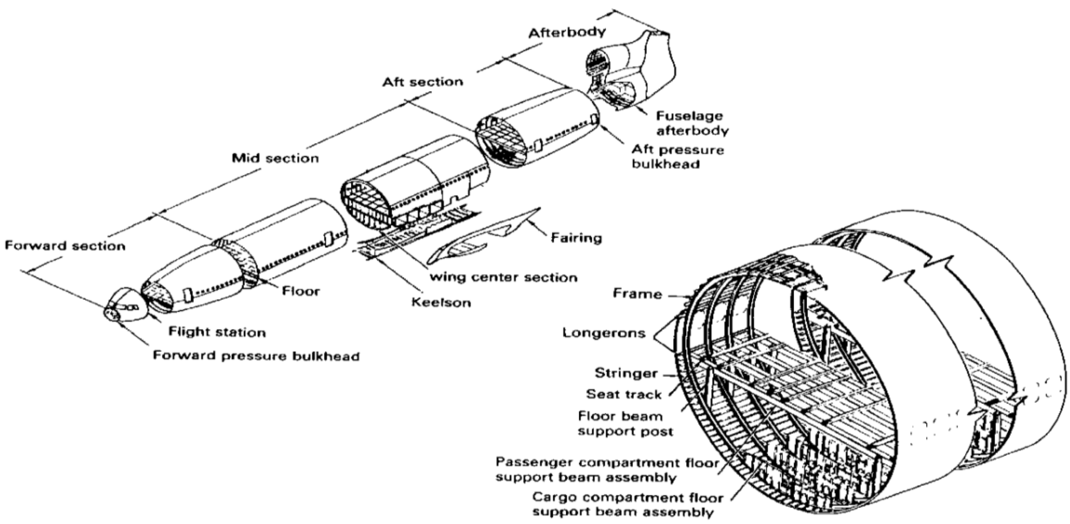


Figure 1: Typical Semi-monocoque Stiffened shell, [4-5].

In the most modern aircraft, the skin plays an important role in carrying loads. Sheet metals can usually only support tension, but if the sheet is folded, it suddenly has the ability to carry compressive loads and stiffeners are then used this purpose. Aircraft structure is usually subjected to different types of loadings, cabin air pressure, and inertia loading or ground reactions during landing, from which three types of applied loadings are developed on the three main aircraft structures, namely shear force, bending moment, and torque.

Aircraft fuselage structure is one of the main sections in the aircraft structure and usually carries some of the applied loads on the aircraft. As a result, the fuselage section should be designed to withstand the ultimate loads generated from the flight envelopes of the aircraft according to the airworthiness regulations. These loads will create different types of stresses, (normal and shear stresses) which may cause a structural failure of the aircraft.

In this work, two approaches are used to obtain the applied stresses along the fuselage structure. The first approach is the theoretical analysis, which is done by developing a computer program using Python Language to determine the structural properties in the form of fuselage stiffness, a center of gravity, and both bending and shear stresses distributions around the fuselage sections. The second approach is done by modeling and simulating the fuselage section presented in [3] using the commercial finite element program, MSC/PATRAN 2004 as pre-and post-processer and analyzed using MSC/NASTRAN 2004. The obtained results of the fuselage sections using the two approaches and [3] are compared and validated.

METHODOLOGY, FORMULATION, AND PROCEDURE

Assumptions

- The material is homogeneous.
- The material is isotropic.
- The material is elastic.
- The longitudinal stiffeners and spar flanges carry only axial stresses.
- The web, skin and spars webs carry only shear stresses.
- The axial stress is constant over the cross-section of each longitudinal stiffeners.
- The shearing stress is uniform through the thickness of the webs.
- Transverse frames and ribs are rigid within their own planes and have no rigidity normal to their plane.
- Stress concentration factor is neglected.

Bending Stresses

Normal at any point in the structure can be determined as a function of the applied moments at that point and the area properties of the cross-section. Equation 1 shows that the longitudinal stress is found by using the following equation, [3 and 5]:

$$\sigma_z = -\frac{M_y I_{xx} + M_x I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} x + \frac{M_x I_{yy} + M_y I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} y \qquad (1)$$

Where

M_X and My are the bending moments in the structure.

I_{xx}, I_{yy}, and I_{xy} are area moment of inertias.

σ_z Direct or normal stress due to bending.

It should be noted that when neutral axis is passing through the centroid on those points the normal bending stresses are zero. The position of the neutral axis compared to the reference axis are depends on the form of the applied loading and the geometrical properties of the cross-section.

The calculation of the center of gravity in x and y-axes; and area moment of inertia is carried out using the following equations, [1-3].

$$x_{cg} = \frac{\sum_{i=1}^{n} x_{i B_i}}{\sum_i^n B_i} \qquad (2)$$

$$y_{cg} = \frac{\sum_{i=1}^{n} y_{i}B_{i}}{\sum_{i}^{n} B_{i}} \tag{3}$$

Where

- B_i Boom area of the i^{th} skin-stringer.
- x_i distance between the reference axis and the i^{th} skin-Stringer in the x-direction.
- y_i distance between the reference axis and the i^{th} skin-Stringer in the y-direction.

The second moment of area about the centroid is given as

$$\begin{aligned} I_{xx} &= \sum_{i=1}^{n} B_i y_i^2 \\ I_{yy} &= \sum_{i=1}^{n} B_i x_i^2 \\ I_{xy} &= \sum_{i=1}^{n} B_i\, x_i y_i \end{aligned} \tag{4}$$

The area moments of inertia about the principal axis are calculated using the developed program in terms of I_{uu} and I_{vv} and I_{uv} as illustrated in [3,6]. The bending stress is calculated using equation (1) for an unsymmetrical fuselage sections, and then it is reduced for the symmetrical fuselage sections, $I_{xy}=0$. The bending stress is usually calculated at each spar boom or stringer with respect to the principal axis, u and v, [2-6].

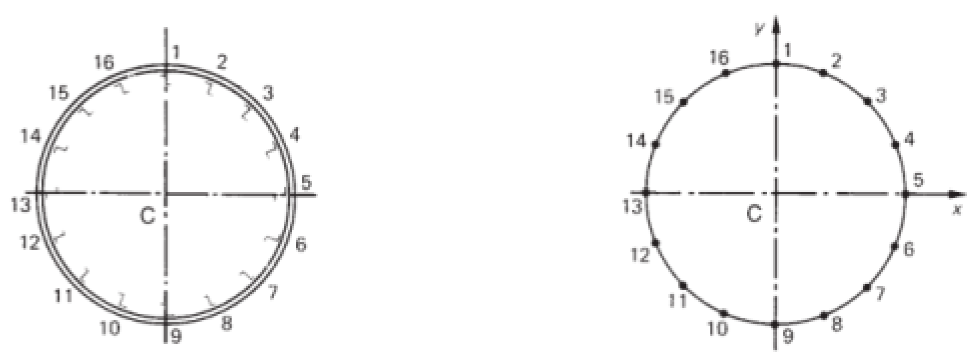
Structural Idealization

Usually, fuselage section is covered by a thin skin and the skin is reinforced by one of the stringer sections such as Z, C or T. The analysis of such a structure would be extremely complex and tedious unless some simplifying assumption for structural idealization should be carried out, Figure (2).

Shear Flow

For a fuselage having a cross-section of the type shown in Figure (2), the determination of the shear flow distribution in the skin produced by shear is basically the analysis of an idealized single cell closed section beam. The shear flow distribution is therefore given by equation (5), in which the direct stress carrying capacity of the skin is assumed zero, [3].

$$q_s = -\left(\frac{S_x I_{xx} - S_y I_{xy}}{I_{xx} I_{yy} - I_{xy}^2}\right)\sum_{r=1}^{n} b_r x_r - \left(\frac{S_y I_{yy} - S_x I_{xy}}{I_{xx} I_{yy} - I_{xy}^2}\right)\sum_{r=1}^{n} b_r y_r + q_{s,0} \tag{5}$$



(a): Actual Fuselage section. **(b): Idealized Fuselage section**

Figure 2: Actual (a) and (b) Idealized Fuselage section, [3].

Equation (5) is applicable to loading cases in which the shear loads, S_x and S_y are not applied through the section shear center so that the effects of shear and torsion are included simultaneously. Alternatively, if the position of the shear center is known, the loading system may be replaced by shear loads acting through the shear center together with a pure torque, and the corresponding shear flow distributions may be calculated separately and then superimposed to obtain the final distribution.

Open shear flow, (q_b) is obtained by supposing that the closed beam section is 'cut' at some convenient point there by producing an 'open' section (Figure 3). The balanced shear flow, ($q_{s;0}$) in the panel with a cut is found by taking moments about a convenient moment center of a cross-section.

Generally, shear flow is produced when the structure is subjected to the shear force and torsional loads. The calculation of the shear flow along the length of the fuselage is carried out using the method presented in [3-7], which based on the above equations.

Fuselage section

Since the fuselage section has been idealized as single cell closed section as shown in Figure 2, the final shear flow distribution is given by the following equation,

$$q_s = -\frac{S_y}{I_{xx}} \sum_{i=1}^{n} B_i y_i + q_{s,0} \tag{6}$$

Taking moments about the points where the shear forces are applied, then this equation becomes:

$$0 = \oint P q_b \, d_s + 2 q_{s,0} A \tag{7}$$

The open shear flow, (q_b) can be obtained by supposing that the closed beam section is 'cut' at some convenient point thereby producing an 'open' section, Figure (3b). The open shear flow distribution, (q_b) around this 'open' section is given by equation (8), [3].

$$q_b = -\frac{S_y}{I_{xx}} \sum_{i=1}^{n} B_i y_i \tag{8}$$

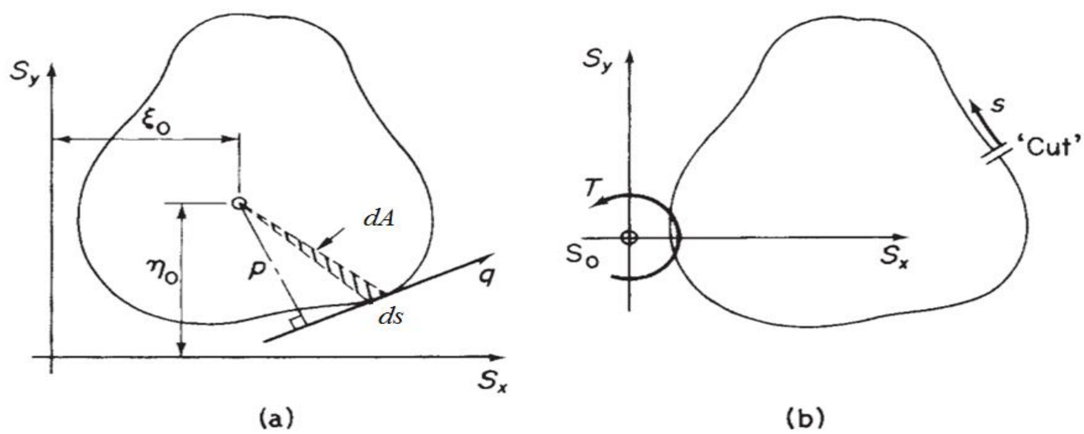


Figure 3: (a) Determination of $q_{s,o}$, (b) Equivalent loading on open section beam, [3].

The final shear flow of the closed fuselage single section is given by

$$q_s = q_b + q_{s,0} \tag{9}$$

Shear stress distributions, τ due to applied shear force and applied torsion for the fuselage sections can be obtained as:

$$\text{Shear Stress, } \tau = \frac{q_s}{t} \tag{10}$$

Where:

q_s is the final shear flow, (N/mm).

t is the thickness of the skin, (mm).

COMPUTER PROGRAM

Educational programs have contributed to a great level in improving the teaching-learning process and in the analysis at early or preliminary design stage; the otherwise difficult concepts for students to understand are made easy by these programs. Detailed homework assignments were usually given to the students in the aircraft component design course that require the students to use an available program and should be given to them starting from the load estimation until the final estimation of stresses on an aircraft fuselage structure.

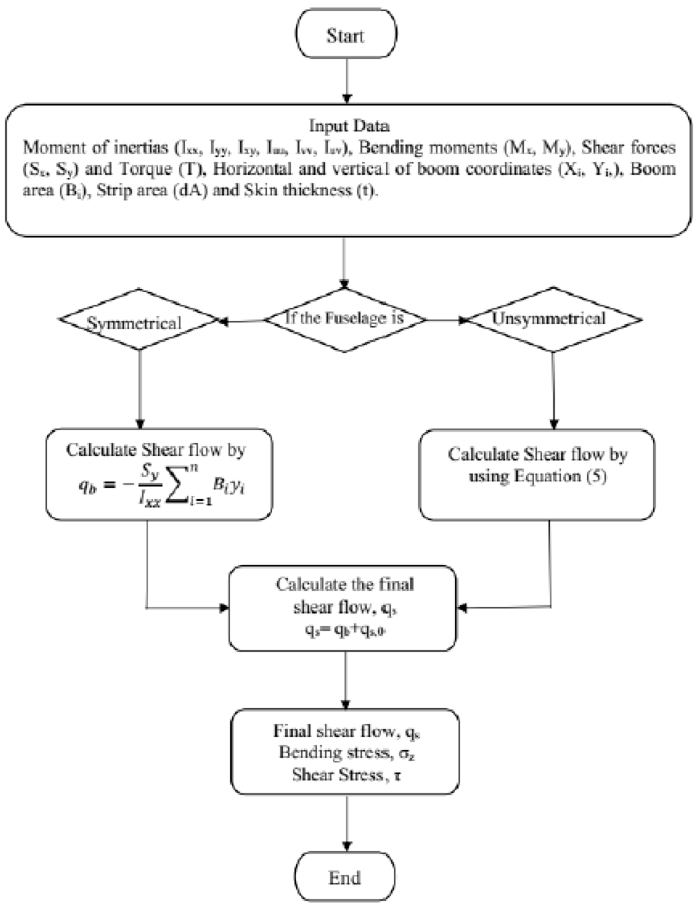


Figure 4: Flowchart of the developed program for the fuselage section.

This will make them clear about the application of these advanced topics in analysis and design of aircraft structures. To bring in all the important concepts taught in the aircraft structures course namely: symmetric and asymmetric bending, shear center and shear of open and closed symmetrical and unsymmetrical section of a semi-monocoque fuselage structure. Therefore, the first aim of the project work is to develop a computer program using the Python Computer Language, [8] that can conveniently compute all structure properties in the form of the center of gravity, stiffnesses, shear flows and bending and shear stresses at any section along the aircraft fuselage length. The detailed input data required and the overall process of the developed program are illustrated in a simple manner using flow chart shown in Figure (4).

Two case studies from the open literature, [3] are selected and used in the developed program for obtaining the structural properties, final shear flow, normal stresses due to bending moments and shear stresses due to applied shear force and torsional moment.

CASES ANALYSIS AND RESULTS

Case study 1: Fuselage circular section, [3]

This problem is presented in [3], page number 338 to calculate the following:

1. Centre of gravity. 2. Moments of inertias. 3. Shear flow.
4. Bending and shear stresses

For comprehensive details regarding the section properties, the reader should refer to [3]. The idealized fuselage section with the geometrical details and lumped areas of the stiffeners are presented in Figure (5) and Table (1). The fuselage is subjected to a vertical shear force of 100 KN at 150 mm form y-axis, and with skin thickness of 0.8 mm.

Using the equations presented above for the fuselage circular section and following the flowchart of the developed program shown in Figure (4), the final results of the shear flow, bending and shear stress distribution of the section are presented in Table (2) and [3]. The output results format of the developed program are shown in Figure (6).

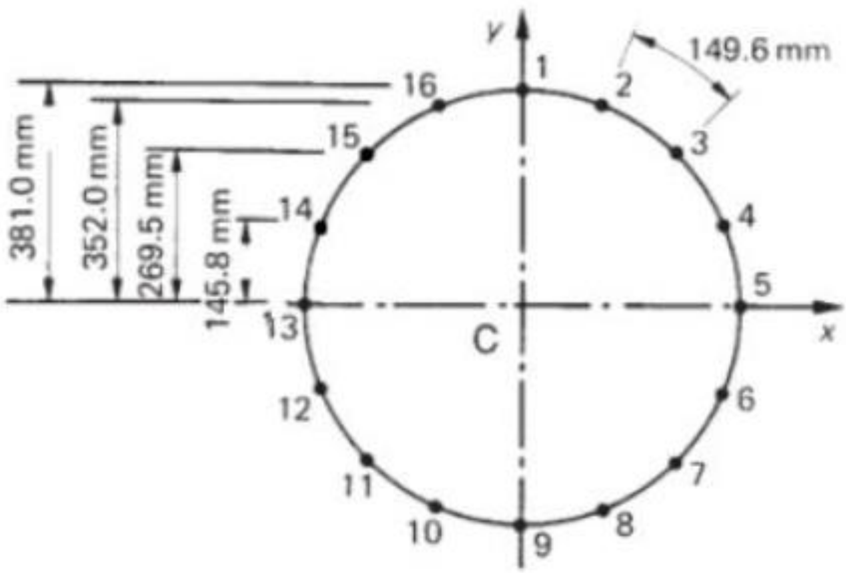


Figure 5: Geometrical details of the idealized fuselage case study 1, [3].

Table 1: Detailed geometry of fuselage case study 1.

Boom No.	Boom Area (mm^2)	X (mm)	Y (mm)
1	216.6	0	381
2	216.6	145.7	352
3	216.6	269.3	269.5
4	216.7	351.9	145.8
5	216.6	381	0
6	216.7	351.9	-145.8
7	216.6	269.3	-269.5
8	216.6	145.7	-352
9	216.6	0	-381
10	216.6	-145.7	-352
11	216.6	-269.3	-269.5
12	216.7	-351.9	-145.8
13	216.6	-381	0
14	216.7	-351.9	145.8
15	216.6	-269.3	269.5
16	216.6	-145.7	352

Table 2: Stress results of the developed program of case study 1.

Boom No.	Shear Flow, q_f (N/mm)	Bending Stress, σ_z (N/mm^2)	Skin Panel	Shear Stress, τ_s (N/mm^2)
1	-32.84	302.85	1-2	-41.05
2	-63.15	279.79	2-3	-78.94
3	-86.35	214.22	3-4	-107.94
4	-98.91	115.89	4-5	-123.63
5	-98.91	0.0	5-6	-123.63
6	-86.35	-115.89	6-7	-107.94
7	-63.15	-214.22	7-8	-78.94
8	-32.84	-279.79	8-9	-41.05
9	0.0	-302.85	9-10	0.0
10	30.26	-279.79	10-11	37.82
11	53.46	-214.22	11-12	66.82
12	66.01	-115.89	12-13	82.52
13	66.01	0.0	13-14	82.52
14	53.46	115.89	14-15	66.82
15	30.26	214.22	15-16	37.82
16	0.0	279.79	16-1	0.0

```
Command Prompt
Microsoft Windows [Version 10.0.10586]
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C:\Users\ammar.alzanati.com> python C:\Users\ammar.alzanati.com\PycharmProjects\untitled\h.py C:\Users\ammar.alz
['FUSELAGE']
cut at 1-2
Node No          bending stress        shear flow          final shear flow        shear stress
1-2              0.00000e+00           -3.28472e+01        3.02851e+02             -4.10590e+01
2-3              -3.03056e+01          -6.31528e+01        2.79800e+02             -7.89410e+01
3-4              -5.35083e+01          -8.63555e+01        2.14221e+02             -1.07944e+02
4-5              -6.60629e+01          -9.89101e+01        1.15894e+02             -1.23638e+02
5-6              -6.60629e+01          -9.89101e+01        0.00000e+00             -1.23638e+02
6-7              -5.35083e+01          -8.63555e+01        -1.15894e+02            -1.07944e+02
7-8              -3.03056e+01          -6.31528e+01        -2.14221e+02            -7.89410e+01
8-9              -7.10543e-15          -3.28472e+01        -2.79800e+02            -4.10590e+01
9-10             3.28023e+01           -4.48741e-02        -3.02851e+02            -5.60926e-02
10-11            6.31079e+01           3.02607e+01         -2.79800e+02            3.78259e+01
11-12            8.63106e+01           5.34634e+01         -2.14221e+02            6.68293e+01
12-13            9.88652e+01           6.60180e+01         -1.15894e+02            8.25225e+01
13-14            9.88652e+01           6.60180e+01         0.00000e+00             8.25225e+01
14-15            8.63106e+01           5.34634e+01         1.15894e+02             6.68293e+01
15-16            6.31079e+01           3.02607e+01         2.14221e+02             3.78259e+01
16-1             3.28023e+01           -4.48741e-02        2.79800e+02             -5.60926e-02

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```

Figure 6: Output results of the developed program of case study 1.

Case study 2: Fuselage section, non-circular, [3]

The second case study fuselage problem is presented in [3]. The fuselage section is subjected to a bending moment, M_x of 100 KN m about the x-axis of the fuselage section as shown in Figure 7.

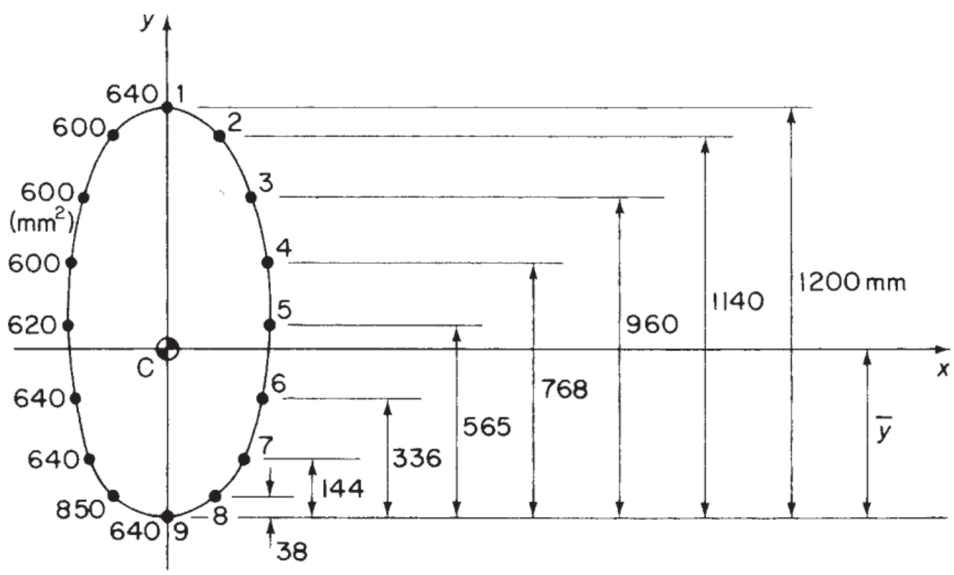

Figure 7: Idealized fuselage section of case study 2, [3].

The analysis of case study 2 is carried out using the developed program to obtain the normal stress distributions due to applied bending moment are presented in Figure 8 and Table (3). Only half of the fuselage section is considered due to the symmetry of the section.

Table 3: Stress results of the developed program of case study 2.

Boom No.	Y, (mm)	Boom Area, (mm^2)	σ_z (N/mm^2)
1	1200	640	35.66
2	1140	600	32.43
3	960	600	22.72
4	768	600	12.37
5	565	620	1.43
6	336	640	-10.91
7	144	640	-21.26
8	38	850	-26.95
9	0	640	-29.0

```
Command Prompt
Microsoft Windows [Version 10.0.10586]
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C:\Users\ammar.alzanati.com> python C:\Users\ammar.alzanati.com\PycharmProjects'
['FUSELAGE']
Node No          bending stress
1                3.56699e+01
2                3.24347e+01
3                2.27292e+01
4                1.23767e+01
5                1.43102e+00
6                -1.09160e+01
7                -2.12685e+01
8                -2.69597e+01
9                -2.90329e+01

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```

Figure 8: Bending stress of fuselage, case study 2.

FUSELAGE MODELLING AND SIMULATION

Finite element model of fuselage section, case study 1 is constructed using a commercial finite element program, MSC/PATRAN 2004. The fuselage model is constructed using the combinations of two and one-dimensional elements, namely four-node plate shear panel elements (CQUAD4) for the fuselage skins and beam (Bar) elements for the stiffeners (stringers). Multi-Point Constrains MPC is applied through the use of rigid elements, REB2 at the fuselage tip section as shown in Figure (9), [9-11].

A fixed free boundary condition is implemented and the fuselage section is subjected to 100 KN at 150 mm from the vertical axis of the fuselage. The fuselage is made from the aluminum material with E=72000 N/mm^2 and G=27000 N/mm^2, [12].

Analysis and results

Static analysis is carried out using solution number 101 in the solver MSC/NASTRAN 2004. The created model section is shown in Figure (9) and the output results of the fuselage model are shown in Figure (10) and presented in Table (4).

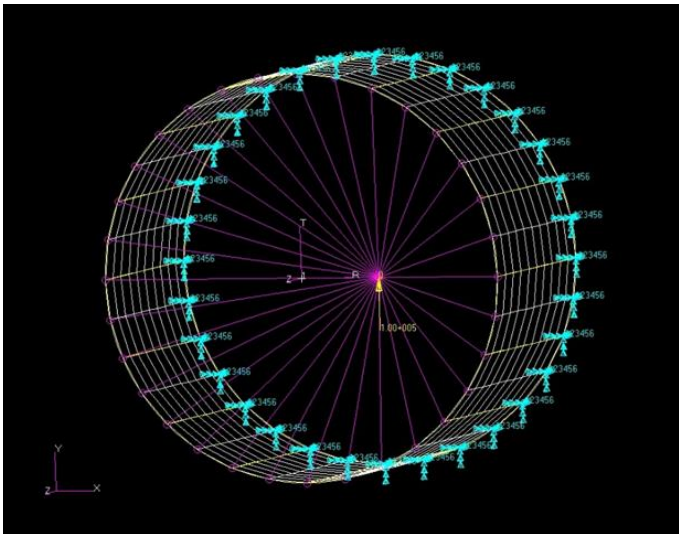

Figure 9: Finite element model of fuselage section, case study 1.

VALIDATION OF RESULTS

Fuselage circular section, case study 1, [3]

Table (5 and 6) show the comparison of results obtained by [3] and finite element model with the developed computer program. The computer program results are very close and acceptable compared with the results obtained by [3].

Table 4: Shear stress results of the fuselage finite element model, case study 1.

Skin Panel	Shear Stress τ, (N/mm^2) MSC/NASTRAN
1-2	40.9
2-3	81.9
3-4	106
4-5	123
5-6	123
6-7	106
7-8	81.9
8-9	40.9
9-10	0.0
10-11	35.8
11-12	65.5
12-13	81.9
13-14	81.9
14-15	65.5
15-16	35.8
16-1	0.0

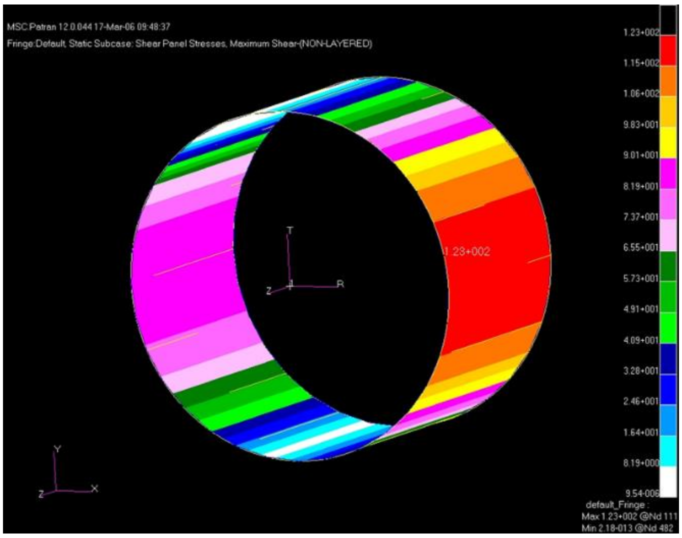

Figure 10: Shear stress results of the fuselage finite element model.

Table 5: Comparison of Stresses of the developed program with [3].

Boom No.	Bending stress σ_z, [3]	Developed Program	Error in %	Skin Panel	Shear Stress, τ, [3]	Developed Program	Error in %
1	302.4	302.85	0.14	1-2	-41	-41.05	0.14
2	279.4	279.79	0.14	2-3	-78.8	-78.94	0.17
3	213.9	214.22	0.15	3-4	-107.8	-107.94	0.13
4	115.7	115.89	0.16	4-5	-123.5	-123.63	0.11
5	0.0	0.0	0.0	5-6	-123.5	-123.63	0.11
6	-115.7	-115.89	0.16	6-7	-107.8	-107.94	0.13
7	-213.9	-214.22	0.15	7-8	-78.8	-78.94	0.17
8	-279.4	-279.79	0.14	8-9	-41	-41.05	0.14
9	-302.4	-302.85	0.14	9-10	0.0	0.0	0.0
10	-279.4	-279.79	0.14	10-11	37.8	37.82	0.06
11	-213.9	-214.22	0.15	11-12	66.8	66.82	0.04
12	-115.7	-115.89	0.16	12-13	82.5	82.52	0.02
13	0.0	0.0	0.0	13-14	82.5	82.52	0.02
14	115.7	115.89	0.16	14-15	66.8	66.82	0.04
15	213.9	214.22	0.16	15-16	37.8	37.82	0.06
16	279.4	279.79	0.14	16-1	0.0	0.0	0.0

Table 6: Comparison of shear stresses of the developed program with MSC/NASTRAN.

Skin Panel	Developed Program	Shear Stress τ, (N/mm^2) MSC/NASTRAN	Error in %
1-2	-41	40.9	0.24
2-3	-78.9	81.9	3.80
3-4	-107.9	106	1.76
4-5	-123.6	123	0.6
5-6	-123.6	123	0.6
6-7	-107.9	106	1.76

7-8	-78.9	81.9	3.80
8-9	-41	40.9	0.24
9-10	0.0	0.0	0.0
10-11	37.8	35.8	5.29
11-12	66.8	65.5	1.94
12-13	82.5	81.9	0.72
13-14	82.5	81.9	0.72
14-15	66.8	65.5	1.94
15-16	37.8	35.8	5.29
16-1	0.0	0.0	0.0

Fuselage non-circular section, case study 2, [3]

Table (7) shows the comparison of bending stress results obtained by [3] and the developed computer program for case study 2. The computer program results are very close and acceptable compared with the results obtained by [3].

Table 7: Bending stress validation for fuselage case study 2.

Boom No	**Bending stress σ_z, [3]**	**Developed program**	**Error in%**
1	35.6	35.66	0.16
2	32.3	32.43	0.4
3	22.6	22.27	1.46
4	12.3	12.37	0.56
5	1.3	1.43	10
6	-11	-10.91	0.81
7	-21.4	-21.26	0.68
8	-27	-26.98	0.07
9	-29	-29.03	0.10

CONCLUSIONS

Structural properties, bending and shear stresses are calculated successfully using the developed computer program written with Python Language for aircraft fuselage sections made from aluminum materials. The finite element commercial program, MSC/PATRAN 2004 and MSC/NASTRAN 2004 are used successfully for constructing and analyzing the fuselage model. The results of the developed program for the two case studies are further compared and validated to the open literature and finite element model of the fuselage sections. The error in percentage is found to be acceptable.

The program can be used for preliminary design and sizing of an aircraft fuselage sections. The present program is expected to be a useful tool to enhance the teaching and learning process of courses on aircraft structures and aircraft structural design.

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