UDK: 624.539.623.629.3.
COSATI: 13-13, 01-03, 13-03

Design and Analysis of the Flat Honeycomb Sandwich Structures

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Structural sandwich is a unique form of the composite structure, and it finds a widespread use in the aerospace industry, where weight saving is a primary concern. The major difference between analysis procedures for sandwich construction and those for homogeneous structural elements is the inclusion of core shear effects on deflection, buckling and stress. The design procedure given in this article is intended to guide the designer in sizing the sandwich parts for primary loading properly. These procedures are usually iterative, and optimum design may require the design of several face–core combinations. Comparing the results obtained through the analytical procedure and Finite Element Analysis (FEA) one can conclude their good agreement. Differences for most of the results are from 10% to 15%, which is quite satisfactory, taking into account that the analytical models are formed on the basis of a number of assumptions and approximations.

Key words: **sandwich structure, honeycomb core, application in aviation, aircraft, aircraft structure, face, structural analysis, finite element analysis.**

Notation and symbols

Mx – Bending moment at a given section
D – Flexural stiffness
Vx – Shear at the section
N – Shear stiffness
a – Length of the longer side of a pressure loaded panel
b – Length of the shorter side of a pressure loaded panel
c – Thickness of the core
d – Distance between the centroids of the faces
h – Total thickness of the sandwich element
t_1, t_2 – Thicknesses of faces 1 and 2
L – Longitudinal direction, parallel to the core ribbon direction
T – Short transverse direction, trough the core thickness
W – Transverse direction, perpendicular to the core ribbon direction
F_{sL} – Allowable core shear stress in the LT plane
F_{sW} – Allowable core shear stress in the WT plane
G_L – Core shear modulus in the LT plane
G_W – Core shear modulus in the WT plane
G_{XZ} – Core shear modulus in the XZ plane
G_{YZ} – Core shear modulus in the YZ plane
F_{sL} – Allowable core shear stress in the LT plane
F_{sW} – Allowable core shear stress in the WT plane
K_{thkW} – Thickness correction factor
K_{thkL} – Thickness correction factor
V – Dimensionless shear parameter
E – Elastic modulus of the facing material
μ – Poisson's ratio for the facing material
R – Mean radius of curvature at the neutral axis of the panel, or G_{YZ}/G_{XZ}
K_{1b} – Face stress coefficient for the *"b"* direction
K_{1a} – Face stress coefficient for the *"a"* direction
K_{2b} – Core shear coefficient for side–b
K_{2a} – Core shear coefficient for side–a
K_3 – Panel deflection coefficient
$f_{1,2}$ – Maximum face stresses
f_{sb} – Core shear stress at the midlength of side–b
f_{sa} – Core shear stress at the midlength of side–a
$MS_{1,2}$ – Margin of safety for each facesheet
$MS_{sb,sa}$ – Margins of safety for core shear

Introduction

A sandwich panel consists of three discrete structural elements: two relatively thin facings, bonded to a thicker, lightweight core (Fig.1).

Depending on the condition that a plane or a missile is exposed to, the face material may be aluminum alloys, reinforced plastic, titanium alloys, heat resistant steel, etc. Materials and geometric forms of the core can be very diverse. A very popular type of the core is a "honeycomb core", which consists of a thin film formed in the hexagonal cell perpendicular to the faces [1].

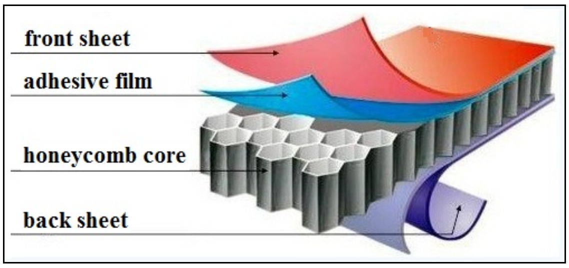

Figure 1. Elements of the honeycomb sandwich structures

Structural sandwich is a unique form of the composite structure, and it finds a widespread use in the aerospace industry, where weight saving is a primary concern. Most commercial airliners and helicopters (Figure 2), and almost all military air and space aircraft widely used the sandwich construction [3].

From the structural point of view, the main role of the core is separating and keeping external faces at a given distance in

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order to provide stability against buckling. Essentially, the existence and the thickness of the core create and maintain the required moment of inertia of the cross section.

The major difference between the analysis procedures for sandwich construction and those for homogeneous structural elements is the inclusion of core shear effects on deflection, buckling and stress [4]. The reasons for the inclusion of this effect are discussed below.

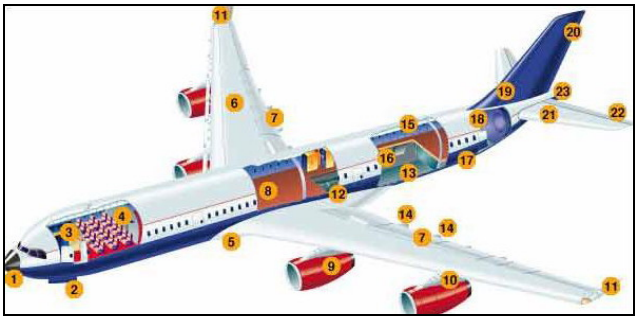


Figure 2. The application of structural sandwich in construction of civil aircraft: 1. Radome 2. Landing Gear Doors and Leg Fairings 3. Galley, Wardrobes, Toilets 4. Partitions 5. Wing to Body Fairing 6. Wing Assembly 7. Flying Control Surfaces 8. Passenger Flooring 9. Engine Nacelles and Thrust Reversers 10. Pylon Fairings 11. Winglets 12. Keel Beam 13. Cargo Flooring 14. Flaptrack Fairings 15. Overhead Storage Bins 16. Ceiling and Side Wall Panels 17. Airstairs 18. Pressure Bulkhead 20. Rudder 21. Horizontal Stabilizer 22. Elevator 23. Tail Cone [3]

The analysis procedures outlined in this paper are intended for use in the structural analysis of both preliminary and final designs of sandwich parts. In fact, the analytical procedure of analysis with an example, which follows, is primarily intended to guide the designer in sizing the sandwich parts for primary loading properly. These procedures are usually iterative, and optimum design may require the design of several face–core combinations [5].

Analysis of the flat honeycomb sandwich panels with isotropic faces

This chapter presents data and methods for the design and analysis of simply supported flat sandwich panels under uniform pressure loads.

Sandwich Stiffness

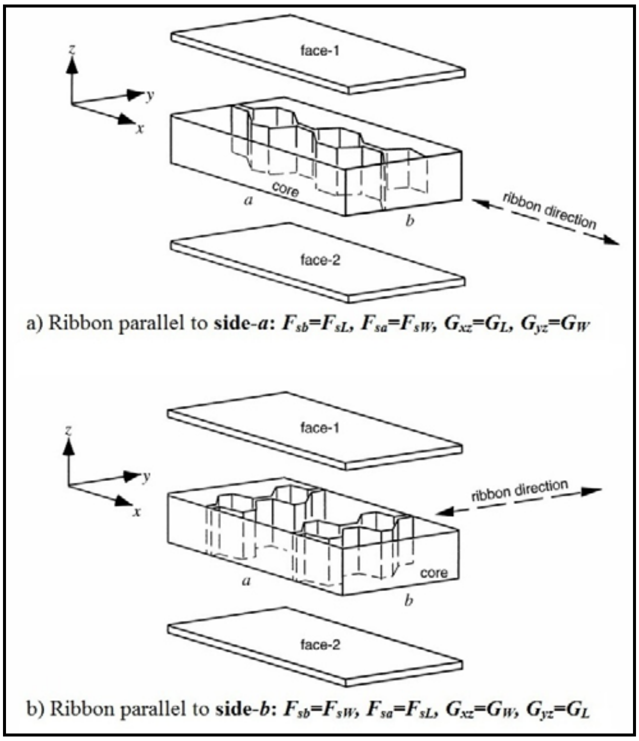
The stiffness of a structure is defined as its ability to resist deformation when subjected to an applied load. The deformations of the sandwich structures, unlike those of monolithic beams, are significantly affected by the contributions of shear deformation.

The deflection of a monolithic beam or plate, according to the elementary beam theory, is governed by the solution of the following differential equations [4]:

$$\frac{d^2y}{dx^2}=\frac{M_x}{D}+\frac{1}{N}\left(\frac{dV_x}{dx}\right)$$

Mx - the bending moment at a given section
D - flexural stiffness
Vx - shear at the section
N - shear stiffness

For most monolithic beams having constant cross sections and large (compared to the beam depth) spans, the second term, which accounts for the shear deformation, may be neglected. This simplifying assumption may normally be made because the shear stiffness, N, is relatively large. However, since sandwich materials have relatively low core shear module, the shear stiffness of most sandwich elements is not so large and this assumption does not hold. Therefore, the deflection calculations for sandwich elements must include the shear contribution [4].



a) Ribbon parallel to **side-*a*:** $F_{sb}=F_{sL}$, $F_{sa}=F_{sW}$, $G_{xz}=G_L$, $G_{yz}=G_W$

b) Ribbon parallel to **side-*b*:** $F_{sb}=F_{sW}$, $F_{sa}=F_{sL}$, $G_{xz}=G_W$, $G_{yz}=G_L$

Figure 3. Core orientation to determine F_{sa}, F_{sb}, G_{XZ} and G_{YZ} [4]

Shear module and strength allowable are based on the core ribbon direction, which may be oriented either parallel to side–a or side-b. Typical values should be used for the core shear module (G_{XZ}, G_{YZ}), while statistically derived allowable should be used for the core shear strengths (F_{SL}, F_{SW}). Both the shear modulus and shear strength must be corrected for core thickness and temperature. Core material properties, along with the appropriate correction curves can be found in the relevant literature (for example BDM-4231 through BDM–4240 [6]). Figures 3 and 4 more clearly define the terms associated with the core ribbon.

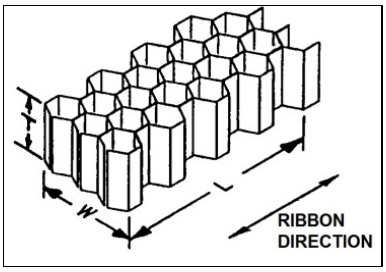


Figure 4. Definition of the core ribbon terms

Analysis Procedure

The analysis procedure for the flat honeycomb sandwich panels with similar isotropic faces subjected to uniform pressure is outlined below and is followed by an example analysis [5].

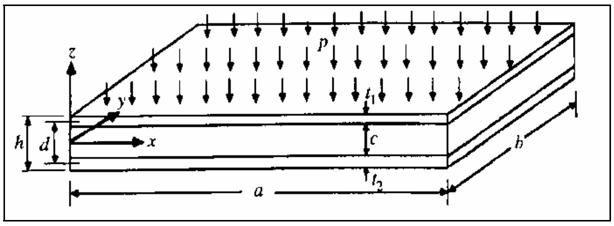


Figure 5. Geometry and loading conventions [5]

Using the data provided in Table 1, determine whether the given panel will withstand the design load. The configuration is shown in Fig.5.

Table 1. Example data for a uniform pressure loaded flat panel

Environmental Temperature		70°F (21°C)					
Face Material		Ti-6Al-4V (Annealed)					
Core Material		Aluminum Alloy Honeycomb 6.5-3/8-50, (5052)					
Core Orientation		Longitudinal direction parallel to x–axis $G_{xz}=G_L$					
Example Values	(AS)	p (psi)	c (in)	t_1 (in)	t_2 (in)	a (in)	b (in)
		65	1,25	0,015	0,015	20	16
	(SI)	p (N/mm^2)	c (mm)	t_1 (mm)	t_2 (mm)	a (mm)	b (mm)
		0,44816	31,75	0,381	0,381	508	406

Step 1: Determine the panel aspect ratio *b/a*.

$$\frac{b}{a} = \frac{16}{20} = 0,8$$

Step 2: Identify G_{XZ} as G_L or G_W using Fig.3 and obtain core properties, for example by using data from BDM–4231 through BDM–4240. Use typical values for the core shear modulus corrected for the applicable thickness and temperature [6].

$$F_{sa} = F_{sW} = (K_{thkW}) \cdot F_{sWnom} = \\ = 0,87 \cdot 306 = 266,33 psi (1,83\mathrm{N/mm}^2)$$

$$F_{sb} = F_{sL} = (K_{thkL}) \cdot F_{sLnom} = \\ 0,83 \cdot 500 = 415 psi (2,861\mathrm{N/mm}^2)$$

$$G_{yz} = G_W = 49700\, psi\ (342,67\ \mathrm{N/mm}^2)$$

$$G_{xz} = G_L = 103900\ \mathrm{psi}\ (716,36\ \mathrm{N/mm}^2)$$

Step 3: Determine the facesheet material properties E, μ, F_{cy} and F_{tu}, corrected for temperature (from BDM-4143 [7]). Determine the core thickness, c, and calculate V.

$$c = h - (t_1 + t_2)$$

$$V = \left[\frac{\pi^2 cE}{(1-\mu^2) b^2 G_{xz}}\right]\left(\frac{t_1 t_2}{t_1 + t_2}\right)$$

$$E = 16000000\, psi\ (110.316\ \mathrm{N/mm}^2),\ \mu = 0,31,$$

$$F_{tu} = 134000\, psi\ (923.89\ \mathrm{N/mm}^2),$$

$$F_{cy} = 133000\, psi\ (917\ \mathrm{N/mm}^2)$$

$$c = 1,25\ in\ (31,75\ \mathrm{mm})$$

$$V = 0,0615$$

Step 4: Determine R. Using R and V, determine K_{1b} and K_{1a} from Figures 6 and 7.

If

$$G_{xz} = G_L \text{ and } G_L > G_W;\ R = 0,4$$

If

$$G_{xz} = G_W \text{ and } G_L > G_W;\ R = 2,5$$

If

$$G_{xz} = G_L;\ R = 1,0$$

$$K_1 = \max(K_{1b}, K_{1a})$$

$$R = 0,4 \Rightarrow K_{1b} = 0,0642;\ K_{1a} = 0,0518$$

$$K_{1b} > K_{1a} \Rightarrow K_1 = K_{1b} = 0,0642$$

Step 5: Calculate the distance between facing centroids d and determine the maximum face stresses.

$$d = h - \frac{t_1 + t_2}{2} = 1,265\ in\ (32,13\,\mathrm{mm})$$

$$f_{1,2} = K_1 \cdot \frac{p \cdot b^2}{d \cdot t_{1,2}}$$

$$t_1 = t_2 \Rightarrow f_1 = f_2 = 56300\, psi\ (388,17\ \mathrm{N/mm}^2)$$

(Note that since p is positive, f_1 is compressive and f_2 is tensile. F_{cy} is used to calculate the MS for face–1 and F_{tu} is used for face–2).

Step 6: Determine the margin of safety for each facesheet.

$$MS_{1,2} = \frac{F_{allow}}{f_{1,2}} - 1$$

$F_{allow} = F_{tu}$, for facesheet in tension

$F_{allow} = F_{cy}$, for facesheet in compression

$$MS_1 = 1,362$$

$$MS_2 = 1,380$$

Step 7: Determine the coefficients K_{2b} and K_{2a} from Fig.8 and calculate the maximum core shear stresses f_{sb} and f_{sa}.

$$\textit{From figure 8} \Rightarrow\ K_{2b} = 0,3519;\ K_{2a} = 0,3819$$

$$f_{sb} = K_{2b} \cdot p \cdot \left(\frac{b}{d}\right)$$

$$f_{sa} = K_{2a} \cdot p \cdot \left(\frac{b}{d}\right)$$

$$f_{sb} = 289,3\, psi\ (1,994.\ \mathrm{N/mm}^2)$$

$$f_{sa} = 314,0\, psi\ (2.164\ \mathrm{N/mm}^2)$$

Step 8: Determine the margins of safety for the core shear; the orientation of the core will determine how F_{sa} and F_{sb} correspond to the core shear allowable F_{SL} and F_{SW}, see Fig.3.

$$M.S._{sb} = \frac{F_{sb}}{f_{sb}} - 1$$

$$M.S._{sa} = \frac{F_{sa}}{f_{sa}} - 1$$

$$MS_{sb} = 0,434$$

$$MS_{sa} = -\ 0,152$$

Step 9: Determine the deflection coefficient K_3 from Fig.9 and check the maximum deflection of the panel δ (at the center of the panel).

$$\delta = -\left[\frac{K_3 \cdot p \cdot b^4 \cdot (1-\mu^2)}{E \cdot d^2}\right] \cdot \left(\frac{t_1 + t_2}{t_1 \cdot t_2}\right)$$

$$\delta = -\ 0,1384\ in\ (3,5\ \mathrm{mm})$$

Step 10: Check the compressive face for local instability failure by consulting the appropriate BDM–6716 (Intracell buckling of the honeycomb sandwich structures), BDM–6718 (Face wrinkling of the flat honeycomb sandwich panels with isotropic faces) and BDM–6720 (Shear crimping of the honeycomb sandwich structures).

The obtained maximum stresses in the faces and the maximum core shear stresses on the side *b* are below the allowable limit stresses. However, the maximum shear stress on the side *a* is above the limit stress, due to which designed honeycomb sandwich beams do not provide the required level of security. In order to ensure the appropriate level of security the plates will increase the thickness of faces.

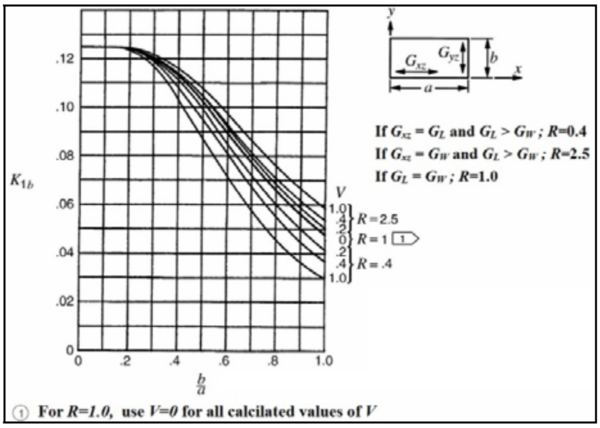


Figure 6. Face stress coefficient K_{1b} for the "b" direction [5]

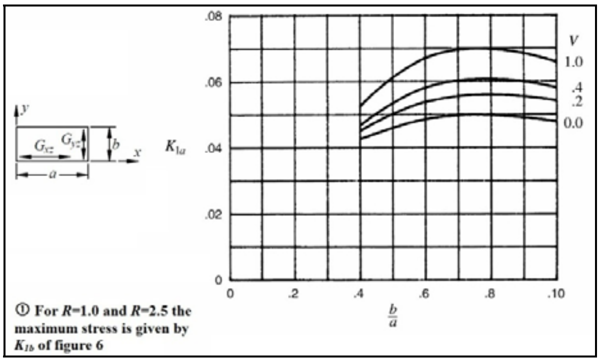


Figure 7. Face stress coefficient K_{1a} for the "a" direction (R=0.4) [5]

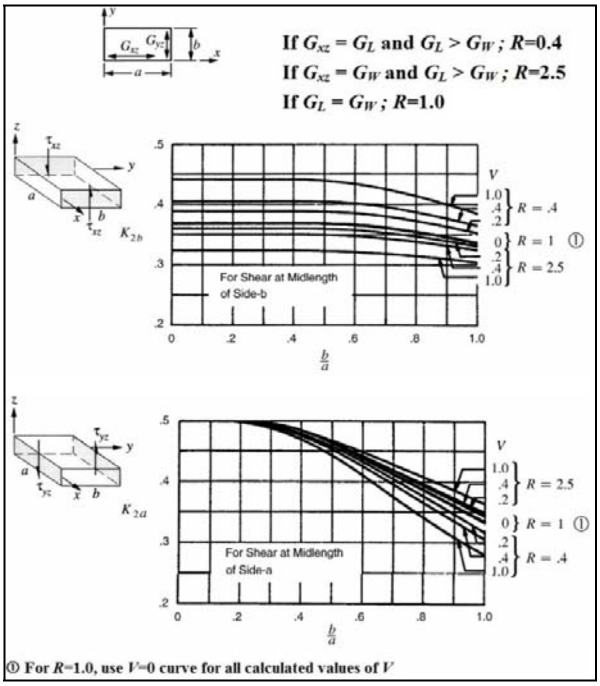


Figure 8. Core shear stress coefficient K_2 [5]

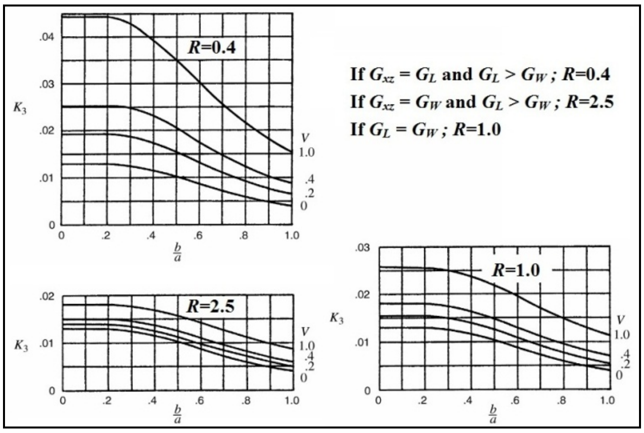
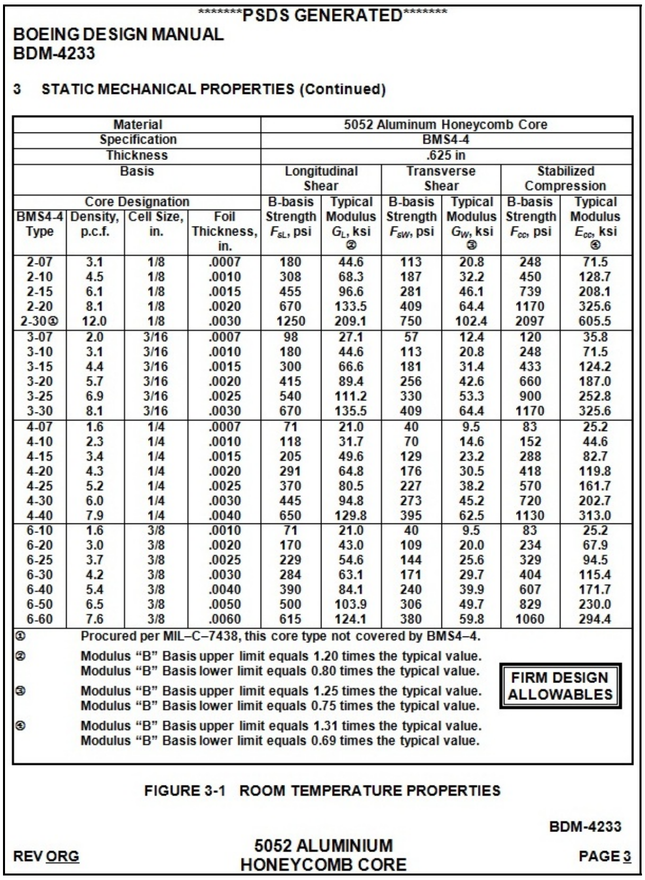


Figure 9. Panel deflection coefficient K_3 [6]



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BOEING DESIGN MANUAL
BDM-4233

3 STATIC MECHANICAL PROPERTIES (Continued)

Material				5052 Aluminum Honeycomb Core					
Specification				BMS4-4					
Thickness				.625 in					
Basis				Longitudinal Shear		Transverse Shear		Stabilized Compression	
Core Designation				B-basis Strength F_{sL}, psi	Typical Modulus G_L, ksi ②	B-basis Strength F_{sW}, psi	Typical Modulus G_W, ksi ③	B-basis Strength F_{cc}, psi	Typical Modulus E_{cc}, ksi ④
BMS4-4 Type	Density, p.c.f.	Cell Size, in.	Foil Thickness, in.						
2-07	3.1	1/8	.0007	180	44.6	113	20.8	248	71.5
2-10	4.5	1/8	.0010	308	68.3	187	32.2	450	128.7
2-15	6.1	1/8	.0015	455	96.6	281	46.1	739	208.1
2-20	8.1	1/8	.0020	670	133.5	409	64.4	1170	325.6
2-30①	12.0	1/8	.0030	1250	209.1	750	102.4	2097	605.5
3-07	2.0	3/16	.0007	98	27.1	57	12.4	120	35.8
3-10	3.1	3/16	.0010	180	44.6	113	20.8	248	71.5
3-15	4.4	3/16	.0015	300	66.6	181	31.4	433	124.2
3-20	5.7	3/16	.0020	415	89.4	256	42.6	660	187.0
3-25	6.9	3/16	.0025	540	111.2	330	53.3	900	252.8
3-30	8.1	3/16	.0030	670	135.5	409	64.4	1170	325.6
4-07	1.6	1/4	.0007	71	21.0	40	9.5	83	25.2
4-10	2.3	1/4	.0010	118	31.7	70	14.6	152	44.6
4-15	3.4	1/4	.0015	205	49.6	129	23.2	288	82.7
4-20	4.3	1/4	.0020	291	64.8	176	30.5	418	119.8
4-25	5.2	1/4	.0025	370	80.5	227	38.2	570	161.7
4-30	6.0	1/4	.0030	445	94.8	273	45.2	720	202.7
4-40	7.9	1/4	.0040	650	129.8	395	62.5	1130	313.0
6-10	1.6	3/8	.0010	71	21.0	40	9.5	83	25.2
6-20	3.0	3/8	.0020	170	43.0	109	20.0	234	67.9
6-25	3.7	3/8	.0025	229	54.6	144	25.6	329	94.5
6-30	4.2	3/8	.0030	284	63.1	171	29.7	404	115.4
6-40	5.4	3/8	.0040	390	84.1	240	39.9	607	171.7
6-50	6.5	3/8	.0050	500	103.9	306	49.7	829	230.0
6-60	7.6	3/8	.0060	615	124.1	380	59.8	1060	294.4

① Procured per MIL–C–7438, this core type not covered by BMS4–4.

② Modulus "B" Basis upper limit equals 1.20 times the typical value.
Modulus "B" Basis lower limit equals 0.80 times the typical value.

③ Modulus "B" Basis upper limit equals 1.25 times the typical value.
Modulus "B" Basis lower limit equals 0.75 times the typical value.

④ Modulus "B" Basis upper limit equals 1.31 times the typical value.
Modulus "B" Basis lower limit equals 0.69 times the typical value.

FIRM DESIGN ALLOWABLES

FIGURE 3-1 ROOM TEMPERATURE PROPERTIES

BDM-4233

REV ORG

5052 ALUMINIUM
HONEYCOMB CORE

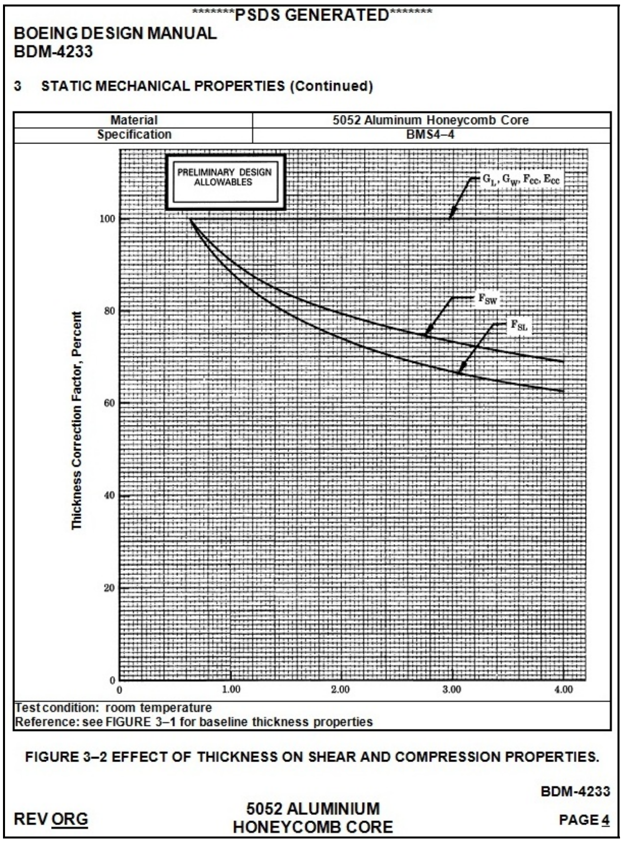
PAGE 3

Figure 10. Static mechanical properties of the aluminum honeycomb core – 5052 aluminum [6]

As already mentioned, these procedures are usually iterative and optimum design may require the design of several face–core combinations.

It is evident that increasing the thickness of faces resulting in increased margins of safety for faces and core. However, increasing the thickness of faces does not significantly affect the increase in the margins of safety for core shear. Therefore, we will increase the thickness of the core.

By increasing the thickness of the faces and thickness of the core we get an acceptable combination of face–core (thickness of the faces $t_{1,2}=0,035in$ and thickness of the core $c=1,50in$) which can support the design load. The maximum deflection of the panel δ is -0,0502 *in* (-1,28 mm).



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BOEING DESIGN MANUAL
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3 STATIC MECHANICAL PROPERTIES (Continued)

Material	5052 Aluminum Honeycomb Core
Specification	BMS4–4

Test condition: room temperature
Reference: see FIGURE 3–1 for baseline thickness properties

FIGURE 3–2 EFFECT OF THICKNESS ON SHEAR AND COMPRESSION PROPERTIES.

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5052 ALUMINIUM
HONEYCOMB CORE
BDM-4233
PAGE 4

Figure 11. Effect of thethickness on shear and compression properties – 5052 aluminum [6]

Table 2. Comparative analysis of the honeycomb sandwich panels

c [in]	1,25			1,50		
$t_{1,2}$ [in]	0,015	0,025	0,035	0,015	0,025	0,035
b/*a*	0,8	0,8	0,8	0,8	0,8	0,8
F_{sa}=F_{sW} [psi]	266,22	266,22	266,22	257,04	257,04	257,04
F_{sb}=F_{sL} [psi]	415	415	415	400	400	400
G_{XY}=G_W [psi]	49700	49700	49700	49700	49700	49700
G_{XZ}=G_L [psi]	103900	103900	103900	103900	103900	103900
V	0,0615	0,1025	0,1435	0,0738	0,1230	0,1722
R	0,4	0,4	0,4	0,4	0,4	0,4
K_{1b} (chart)	0,0642	0,0629	0,0617	0,0638	0,0623	0,0608
K_{1a} (chart)	0,0518	0,0531	0,0543	0,0522	0,0537	0,0552
K_1=max (K_{1b}, K_{1a})	0,0642	0,0629	0,0617	0,0638	0,0623	0,0608
h [in]	1,280	1,300	1,320	1,530	1,550	1,570
d [in]	1,265	1,275	1,285	1,515	1,525	1,535
$f_{1,2}$ [psi]	56300	32836	22828	46716	27191	18831
MS_1	1,362	3,050	4,826	1,847	3,891	6,063
MS_2	1,380	3,081	4,870	1,868	3,928	6,116
K_{2b} (chart)	0,3519	0,3499	0,3479	0,3513	0,3488	0,3464
K_{2a} (chart)	0,3819	0,3799	0,3778	0,3813	0,3788	0,3764
f_{sb} [psi]	289,3	285,4	281,6	241,2	237,9	234,7
f_{sa} [psi]	314,0	309,9	305,8	261,8	258,3	255,0
MS_{sb}	0,434	0,454	0,474	0,659	0,682	0,704
MS_{sa}	-0,152	-0,141	-0,129	-0,018	-0,005	0,008
K_3	0,0069	0,0075	0,0082	0,0071	0,0078	0,0086
δ [in]	-0,1384	-0,0888	-0,0683	-0,0993	-0,0646	-0,0502

Analysis of the flat honeycomb sandwich plates by the Finite Element Method

The modern design processes of the new, as well as monitoring the integrity of the existing structures in the real exploitation conditions, is unthinkable outside the environment of computer mechanics. Specific engineering problems are being solved by using numerical methods implemented on computers. One of the main advantages of the computer designing is short time and inexpensive simulation of behavior of the model of a real object observed.

At an early stage of design, these programs provide an opportunity to obtain reliable information about the validity of the assumed size and accuracy of the provided constructive solutions. The advantage of using these packages in the design is primarily reflected in the ease of making model and its correction.

Most software packages that have the ability of structural analysis are based on the Finite Element Method - FEM (Finite Element Analysis - FEA). The basic idea of this method is division of the structure into the finite number of small elements that constitute the basis for all considerations [8].

Analysis of the flat honeycomb sandwich panels under uniform load using the ALGOR software package

In this case, the subject of the analysis is a flat honeycomb sandwich plate under the uniform load, whose dimensions and margins of safety for provided constructive solutions are previously analyzed analytically.

It was found that the honeycomb sandwich panel with the faces of titanium, Ti-6Al-4V, the thickness of 0.035 in (0.889 mm) and aluminum honeycomb infill 6.5-3/8-50 (5052), the thickness of 1.5 in (38.1 mm) will withstand the design load of 65 psi (0.448159 N/mm^2).

Fig.13 shows the model of the flat honeycomb sandwich panel with a realistic uniform load. Considering that sandwich plate and load are axially symmetric, we can model and analyze one quarter of the plate.

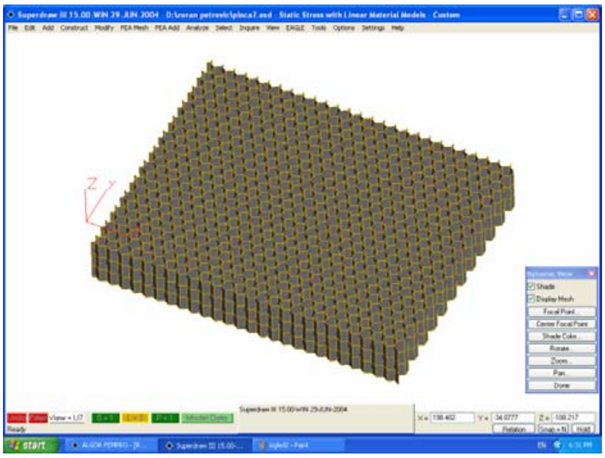


Figure 12. Honeycomb core model

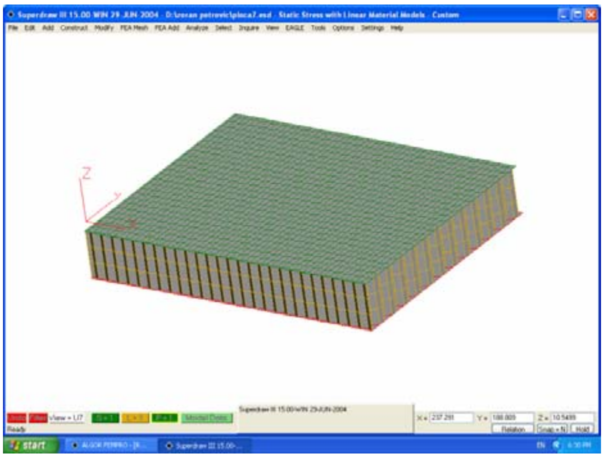


Figure 13. The final model of the honeycomb sandwich panel

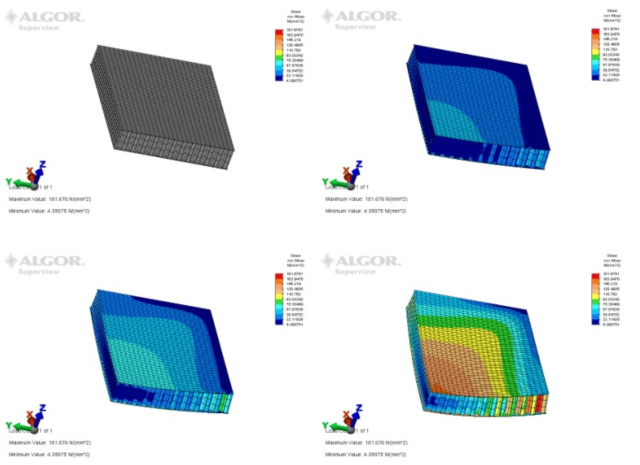


Figure 14. The resulting stresses at $t=t_0$, $t=t_1$, $t=t_2$ i $t=t_{4\,(final)}$.

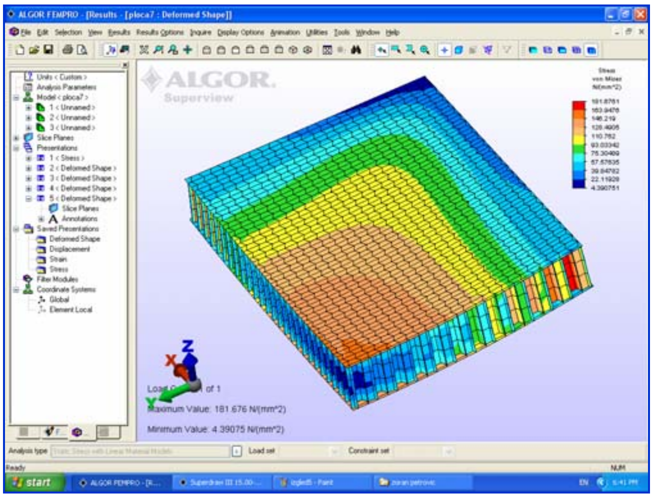


Figure 15. Stress state of the panel (view from above).

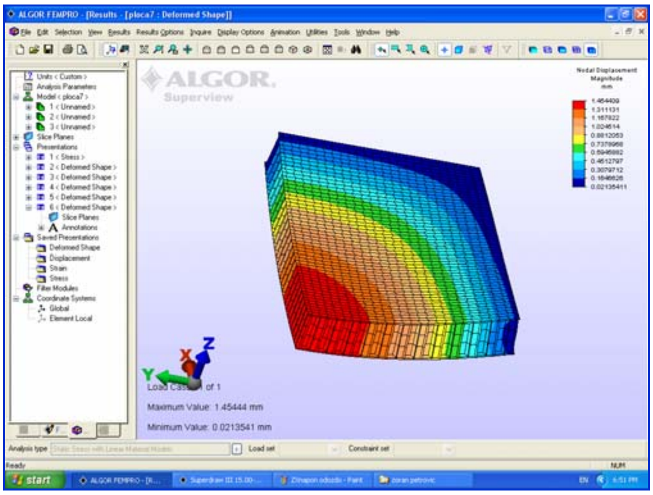
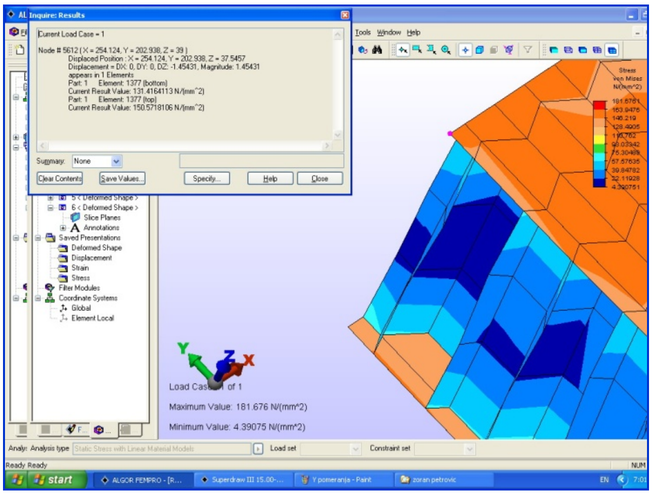


Figure 16. The resulting deflections of the honeycomb sandwich panel under uniform load.

By using the ALGOR software package, following results were obtained, $f = 150{,}57$ N/mm^2 and $\delta = -1{,}454$ mm (Fig.17).

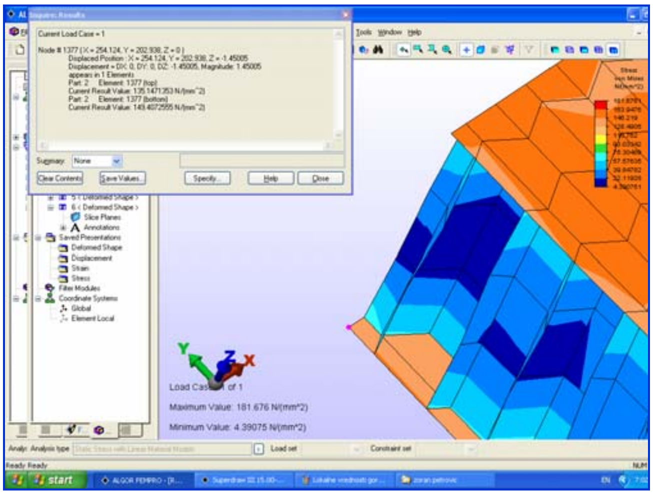
Comparing the results obtained through analytical and computer procedures, one can conclude their good agreement. Differences for most of the results are from 10% to 15%, which is quite satisfactory, taking into account that the analytical models are formed on the basis of a number of assumptions and approximations.



Current Load Case = 1

Node # 5612 (X = 254.124, Y = 202.938, Z = 39)
Displaced Position : X = 254.124, Y = 202.938, Z = 37.5457
Displacement = DX: 0, DY: 0, DZ: -1.45431, Magnitude: 1.45431
appears in 1 Elements
Part: 1 Element: 1377 (bottom)
Current Result Value: 131.4164113 N/(mm^2)
Part: 1 Element: 1377 (top)
Current Result Value: 150.5718106 N/(mm^2)

Figure 17. Local values of stress and deflection in the center panel on the upper face



Current Load Case = 1

Node # 1377 (X = 254.124, Y = 202.938, Z = 0)
Displaced Position : X = 254.124, Y = 202.938, Z = -1.45005
Displacement = DX: 0, DY: 0, DZ: -1.45005, Magnitude: 1.45005
appears in 1 Elements
Part: 2 Element: 1377 (top)
Current Result Value: 135.1471353 N/(mm^2)
Part: 2 Element: 1377 (bottom)
Current Result Value: 149.4072555 N/(mm^2)

Figure 18. Local values of stress and deflection in the center panel on the lower face

Conclusion

Structural sandwich is a unique form of the composite structure, and it finds a widespread use in the aerospace industry, where weight saving is a primary concern. Most commercial airliners and helicopters, and nearly all military air and space vehicles, make the extensive usage of the sandwich construction.

The major difference between the analysis procedures for sandwich construction and those for homogeneous structural elements is the inclusion of core shear effects on deflection, buckling and stress.

The design procedure given in this article is intended to guide the designer in sizing the sandwich parts for primary loading properly. These procedures are usually iterative, and optimum design may require the design of several face–core combinations

Comparing the results obtained through analytical and computer procedures, one can conclude their good agreement. Differences for most of the results are from 10% to 15%, which is quite satisfactory, taking into account that the analytical models are formed on the basis of a number of assumptions and approximations.

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Received: 08.01.2015.
Accepted: 24.02.2015.

Projektovanje i analiza ravnih saćastih sendvič struktura

Strukturalni sendvič kao jedinstvena forma kompozitne strukture je rasprostranjen u vazduhoplovno-kosmičkoj industriji, gde je smanjenje težine primarni interes. Osnovna razlika između postupaka analize za sendvič konstrukciju i onih za homogene strukturalne elemente je uključenje uticaja smicanja jezgra na ugibanje, izvijanje i naprezanje. Postupak analize, koji sledi, prvenstveno je namenjen za vođenje projektanta u pravilnom dimenzionisanju sendvič delova za primarno opterećenje. Ovakvi postupci su obično iterativni i optimalni projekat može zahtevati projektovanje nekoliko kombinacija oplata–jezgro. Poređenjem rezultata dobijenih analitičkim i računarskim putem, može se konstatovati njihovo dobro poklapanje. Razlike za većinu rezultata se kreću od 10% do 15% što je sasvim zadovoljavajuće, kada se ima u vidu da su navedeni analitički modeli formirani na bazi niza pretpostavki i aproksimacija.

Ključne reči: **sendvič elementi, saćasta struktura, primena u vazduhoplovstvu, letelica, struktura letelice, oplata, strukturna analiza, metoda konačnih elemenata.**

Проектирование и анализ плоских сотовых сэндвич - структур

Сэндвич-структура как уникальная форма композитной структуры распространена в авиационно-космической промышленности, где снижение веса является основным интересом. Основной разницей между методами анализа для сэндвич-конструкции и для однородных структурных элементов является включение воздействия отклонения ядра на прогиб, изгиб и напряжение. Процесс анализа, который следует, в первую очередь предназначен для направления дизайнера в правильном определении размеров сэндвич-компонентов для первичной нагрузки. Эти процедуры, как правило, итеративные и оптимальный проект может потребовать проектирование и сочетание нескольких комбинаций обшивка-ядро. Сравнивая результаты, полученные аналитическими и вычислительными способами, можно сделать вывод, что они хорошо согласуются. Разницы в отношении большинства результатов могут колебаться от 10% до 15%, что является вполне удовлетворительным, имея в виду, что приведённые аналитические модели формируются на основе ряда предположений и приближений.

Ключевые слова: **сэндвич-элементы, сотовая структура, использование в авиации, самолёт, конструкция самолёта, обшивка, структурный анализ, метод конечных элементов.**

Conception et analyse des structures sandwich plates alvéolées

Le sandwich structural comme une forme unique de structure composite est répandu dans l'industrie aérospatiale où la diminution du poids représente l'intérêt primordial. La différence principale entre le procédé d'analyse pour les construction sandwich et celles pour les éléments structuraux homogènes est l'inclusion de l'influence de déflection du noyau quant au fléchissement , torsion et tension. Le procédé d'analyse qui suit est destiné avant tout pour guider le dessinateur dans la conception adéquate du dimensionnement des parties sandwich pour la charge primaire. Ces procédés sont généralement itératifs et la conception optimale peut exiger plusieurs combinaisons revêtement - noyau. En faisant la comparaison entre les résultats obtenus par la voie analytique et numérique on peut constater bon accord entre eux. Les différences pour la plupart des résultats sont de 10% à 15% ce qui est satisfaisant si l'on tient compte du fait que les modèles analytiques cités sont formés à la base d'une série d'hypothèses ou d'approximations.

Mots clés: **éléments sandwich, structure alvéolée, emploi dans l'aviation, aéronef, structure d'aéronef, revêtement, analyse de structure, méthode des éléments finis.**