Iranian Society of Acoustics and Vibration
5th ISAV2015
International Conference on Acoustics and Vibration
25-26 Nov 2015 Tehran - Iran
University of Tehran

Design and Sample Manufacturing of Micro Satellite Sandwich Panel Structure: Analytical-Experimental

Morteza Shahravi[a], Majid Mokhtari [b, c*]

[a] *Associate Professor, Aerospace Engineering Department, Malek-e-Ashtar University of Technology, Lavizan, Tehran, Iran.*

[b] *Ph. D student, Aerospace Engineering Department, Malek-e-Ashtar University of Technology, Lavizan, Tehran, Iran.*

[c] *Structure & Manufacturing Department, CRG, Tehran, Iran.*

** m_mokhtari@mut.ac.ir*

Abstract

The current investigation describes design, analysis and sample manufacturing process of a micro-satellite structural sandwich panel. Structure design and analysis is based on a simplified analytical model of satellite structure and modal analysis requirements. Stress analysis of satellite panel is based on simple analytical expressions which have been mentioned in literature and numerical modelling with commercial finite element (FE) code, ABAQUS 6.12.1. This study will help to guide the designers and manufacturing specialists to choosing the simple and effective methods of designing and manufacturing of satellite sandwich panel structures.

Keywords: satellite structure; carbon fiber reinforced polymers (CFRP); modal analysis; stress analysis.

1. Introduction

Satellites have gained considerable interest in the past decades as a result of applications in communication, remote sensing, and etc. The capabilities of micro satellites have continued to increase every year for better utilities to carryout different types of missions which previously required much large satellite platforms. The miniaturization of electronic components enabled to develop satellites of small size, with weight ranging from 10-100 kg. The benefits of developing a micro satellite leads to a shorter development time and much lower costs than that required for larger satellite, while achieving the same and high levels of performance and capability.

Complex environment and space mission constraints have pushed demands such as lighter weight structure, limitation of mass, low energy consumption, and reduced launch cost. However, due to the flexible property of large sandwich panels, the deflection of the flexible panels has a significant influence on the dynamics performance and structural durability of satellites. Mainly, larger space structures are made with composite materials. Space composites properties depend upon many factors such as fiber orientation, matrix and fiber types and so on. Environmental conditions and various corrosion properties/condition affect the composites performances [1-5].

Huang et al. [6] developed an efficient curved beam element compatible with the shell element for the modelling of stiffeners. They predicted buckling response of grid stiffened composite panels having different stiffening arrangements. Their parametric study showed the effects of skin thickness, stiffener breadth and stiffener depth on the buckling capacity of grid stiffened composite plates.

Detailed simulation of the coiling process of a C-section boom made from a two-ply laminate of carbon fiber reinforced plastic, studied with Stabile et al. [7]. They checked the feasibility of coiling an ultra-thin composite boom into a small deployment system. They verified their results using energy balance assessments, and they evaluate the structural strength of the boom via three specific failure criteria.

Deformation and stress analysis of a composite flapping wing according to research into unsteady, low Reynolds number aerodynamics, presented with Yu et al. [8]. They showed that the maximum deformation at the beginning stages of down stroke is 19% larger than the maximum deformation at the beginning stages of upstroke, and the maximum stress at the beginning stages of down stroke is 29.9 larger than the maximum stress at the beginning stages of upstroke.

Kumar et al. [9] extracted the properties of composite materials with a Laminate Theory Software. They modelled a spoiler using Solid Works software and analysed for the static and dynamic conditions using ANSYS software. Their software extracted results analysed for its compact ability and finally best prediction of fiber orientation.

Veeramuthuvel et al. [10] studied a dynamic – thermal analyses of a structurally reconfigured electronics package of a mini satellite. At first they determined the vibration responses at various critical locations, including on the Printed Circuit Board (PCB) for the vibration loads specified by launch vehicle using Finite Element Analysis (FEA). Their predicted temperature values of thermoelastic analysis have been compared with on-orbit observations.

Design, structural analysis, and qualification by analysis and experimental validation of the 1st Greek cube-satellite presented with Ampatzoglou et al. [11]. Their key innovative approach includes the replacement of the aluminium parts of the primary structure with structural composite components. Their results have shown that the redesign of the structure using CFRP can offer similar levels of performance in terms of stiffness while reducing mass by approximately 40%.

In the present work, composite laminate design, modal analysis and manufacturing process of a model micro-satellite structure have been explained. Design and analysis of the structure is based on a simplified analytical model of satellite structure and modal analysis. Stress analysis of satellite panel is based on simple analytical expressions which have been mentioned in literature and numerical modelling with commercial FE code, ABAQUS 6.12.1. This study will help to guide designers and manufacturing specialists to choosing the simple and effective methods of designing and manufacturing of primary structures of satellite sandwich panels.

2. Composite Laminate Design

In preliminary design phase, dynamic loadings subjected to base of the satellite have been extracted with respect to natural frequency of pre-designed satellite and power spectral density (PSD) diagrams of selected launcher LM (Miles' equation [12]), as shown in Table 1. Geometrical model of micro-satellite is shown in Figure 1.

Table 1 Load condition with respect to Miles' equation and Launcher User's Manual

	Longitudinal	Lateral
Steady-State Acceleration	8.5g	1g
Sine-Vibration	1.875g	1.5g
Quasi-Static	**9g**	2.85g
Random Vibration	5.81g	5.81g

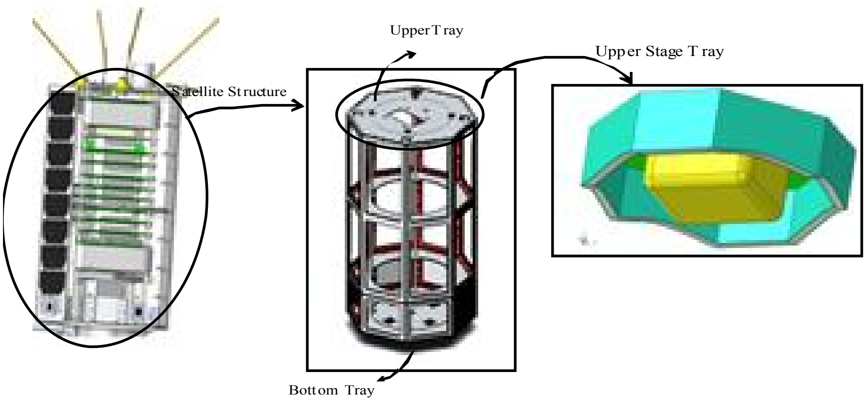


Figure 1 Schematic sketch of model satellite geometry and its upper tray with attached equipment (about 350×350×700 mm)

After extrication of exerted dynamic loads; the optimum ratio between the core thickness h_c (m) and face sheet thickness t_f (m), ($\frac{h_c}{t_f}$) has been optimized with respect to a minimum mass m_{total} (kg/m) of the sandwich construction against: maximum bending stiffness, strength, general buckling and faces dimpling.

The mass per unit area of sandwich panel (m_{total}) becomes total of the mass of the core $\rho_c h_c$ (kg/m), the mass of both face sheets (in this case equal face sheets) $2\rho_f h_f$ and the mass per unit of area of both adhesive layers (especially in the case of adhesive film usage and core bonding and splicing) $2\rho_a h_a$.

$$m_{total} = \rho_c h_c + 2\rho_f h_f + 2\rho_a h_a \tag{1}$$

The maximum optimum thickness of sandwich panel core is calculated with comparison of expressions results of minimum mass optimization against: bending stiffness, strength, general buckling and faces dimpling. At first in comparison reasons and primary calculation, two aerospace Aluminium grades, 7075 and 2618 (with average mechanical properties: $E_f = 71GPa, \nu_f = 0.28$, $\rho_f = 2850\frac{kg}{m^3}, \sigma_{ultimate} = 540MPa$) and T 300/Epoxy ($\rho_f = 1500\frac{kg}{m^3}$ [3]) were selected and compared to each others. Also, Alominum core HC-1/8-5056-0.001 ($\rho_c = 72\frac{kg}{m^3}$ [5], $E_c = 1275MPa$, $G_L = 483MPa$, $G_W = 193MPa$) is selected as sandwich panel core.

As shown in Table 2, the maximum optimum core thickness vs. 0.6 mm face-sheet thickness is 9.366 mm. In order to simplifying the manufacturing process, available raw materials and upper safety factor against less than 2% core additional mass, 10 mm core thickness has been selected to continue manufacturing and design process.

Table 2 Optimum h_c in two suggested t_f against these parameters: bending stiffness, general buckling, strength and face dimpling

	Materials	Al-7075		Al-2618		T300/914	
Opt. against [4]		h_c (mm)	t_f (mm)	h_c (mm)	t_f (mm)	h_c (mm)	t_f (mm)
Bending stiffness	$\frac{\rho_c h_c}{2\rho_f t_f} = 2$	3.122	0.2	3.066	0.2	16.6	0.2
		9.366	**0.6**	9.198	0.6	5	0.6
General buckling	$\frac{h_c}{t_f} = \frac{4\rho_f - 3\rho_c}{\rho_c}$	2.522	0.2	2.766	0.2	16.06	0.2
		7.56	0.6	7.39	0.6	4.8	0.6
Strength	$\frac{\rho_c h_c}{2\rho_f t_f} = 1$	1.560	0.2	1.532	0.2	8.3	0.2
		4.68	0.6	4.596	0.6	2.5	0.6
Face Dimpling	$\frac{\rho_c h_c}{2\rho_f t_f} = \frac{1}{3}$	1.560	0.2	1.532	0.2	2.76	0.2
		4.68	0.6	4.596	0.6	0.83	0.6

Primary thickness study of the satellite sandwich panel core shows that the optimum thickness of the face sheets, against the mentioned parameters is 10 mm. Closed form elasticity solution of a composite sandwich panel plate (Figure 2) [5] with distributed load case and simply supported boundary conditions (BCs) with design correction factors and related nomenclatures of Table 3 have been presented from [5]. T300 /Epoxy composite face-sheets ($E_1 = 95.98GPa$, $\sigma_u = 373MPa$, $\mu = 0.28$) with isotropic lamination [0/45/90/0] have been selected to design and manufacturing purpose.

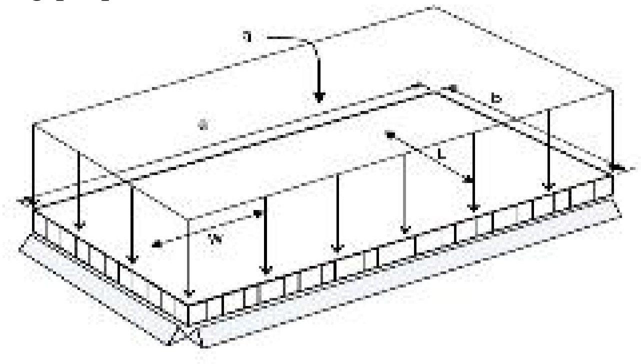


Figure 2 Schematic model of critical micro satellite sandwich panel structure (upper tray), fully simply supported with sidelong mechanical joints (a=30 cm, b=250 cm, q=40 kN/m^2)

Table 3 Nomenclature

a	Panel length	h	Distance between facing skin centres	t_c	Thickness of core

b	Panel width	K_1, K_2, K_3	Panel parameter (used for simply supported plate)	t_f	Thickness of facing skin
A	Area of applied load	P	Applied load	V	Panel parameter (used for simply supported plate)
E_c	Compression modulus of core	q	Uniformly distributed load	δ	Calculated deflection
E_f	Modulus of elasticity of facing skin	P	Applied load	σ_c	Core compressive stress
G_c	Core shear modulus - in direction of applied load	q	Uniformly distributed load	σ_f	Calculated facing skin stress
G_L	Core shear modulus - Ribbon direction	R	Ratio G_L / G_W	τ_c	Shear stress in core
G_W	Core shear modulus - Transverse direction	S	Panel shear stiffness	λ	Bending correction factor for Poisson's Ratio effect $1-\mu^2(Poisson's\ ratio)$

Plate deformation and facing stress of described problem in Figure 2, is presented in Eq. (5) and Eq. (6), respectively.

$$\text{plate coefficient } \left(\frac{b}{a}\right) \longrightarrow \frac{b}{a}=\frac{250}{300}=0.83 \quad (2)$$

$$R=\frac{G_L}{G_W} \longrightarrow R=2 \quad (3)$$

$$V=\frac{\pi^2 E_f t_f h}{2b^2 G_W \lambda} \longrightarrow V=1.66 \quad (4)$$

$$\delta=\frac{2k_1 q b^4 \lambda}{E_f t_f h^2} \longrightarrow \delta=7.1\times10^{-2}\ mm \quad (5)$$

$$\text{Facing stress } \sigma_f=\frac{K_2 q b^2}{h t_f} \longrightarrow \sigma_f=5.12\times10^5\ Pa \quad (6)$$

$$\text{Core shear: } \tau_c=\frac{K_3 q b}{h} \longrightarrow \tau_c=3.077\times10^5\ Pa \quad (7)$$

Preliminary design results have been used to verification purpose of numerical results.

3. Numerical Modelling

ABAQUS/CAE, a commercial nonlinear finite element code developed by SIMULIA Inc., was used to perform the numerical analysis. The complete structure has been meshed using two dimensional quad-dominated quadratic elements.

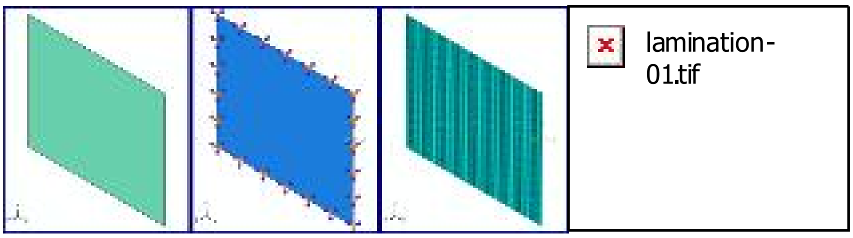


Figure 3 Model Geometry, BCs, mesh size and composite lay-up of FEM model (respectively from left to right)

The model geometry, BCs and mesh created are shown respectively in Figure 3. Upper surface of satellite panel (300×250×16 mm) is subjected to a uniform distributed pressure of 3000 kPa, which is selected to simulate the specimen under launch condition. To ensure there is no relative motion between the face sheets and honeycomb, both of them are constrained by merging with the adhesive layer. After mesh study process and find the best mesh size, modal analysis results of a single sandwich plate and the whole satellite structure performed and is shown in Figure 4 and Figure 5 respectively. Non-zero natural frequencies are reported in Table 4. According to critical loading condition highlighted in Table 1, maximum equivalent stress and deformation of sandwich panel plate has been studied and presented in Figure 6 and Table 5. As shown in Table 5 there is a good agreement between the maximum stress results of FEM and analytical result.

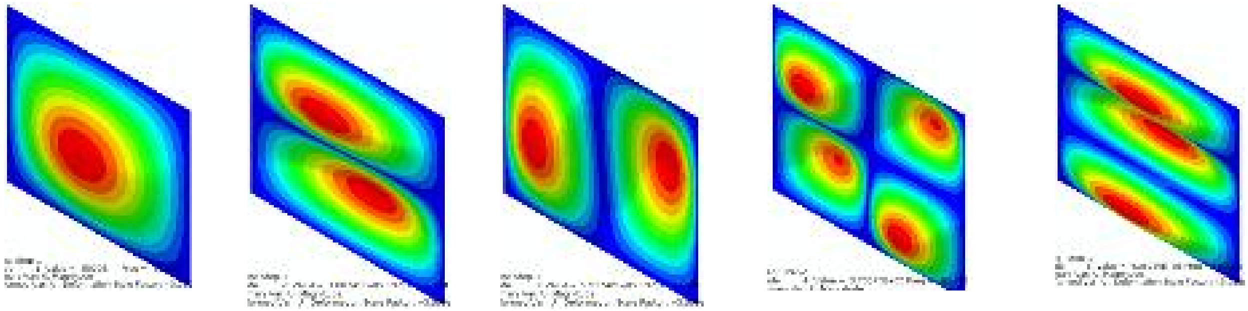
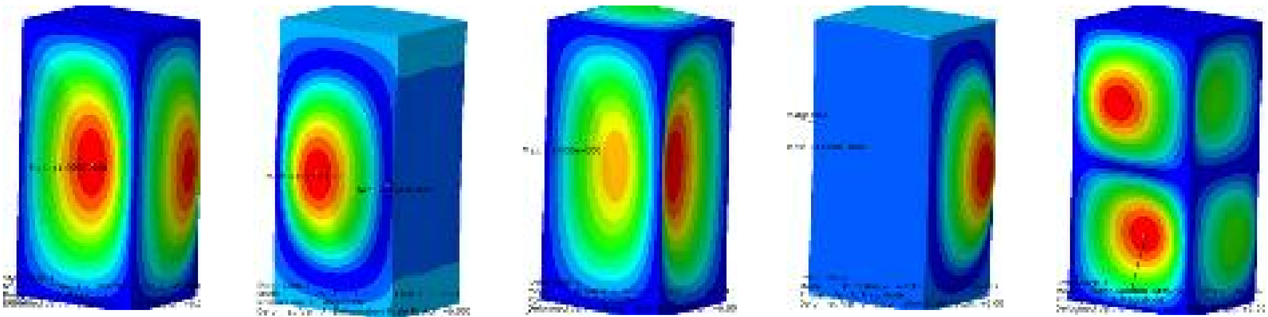


Figure 4 Shape modes and Natural frequencies (NFs) of simply supported sandwich panel of satellite structure (First NF: 48.52 Hz- second NF: 75.1 Hz, Third NF: 83.7 Hz)



1th shape mode and NF (25.5 Hz)	2th shape mode and NF (33.23 Hz)	3th shape mode and NF (34.01 Hz)	4th shape mode and NF (34.43 Hz)	5th shape mode and NF (34.71 Hz)

Figure 5 Non-zero NFs and shape modes of satellite Box in unconstrained condition with 8.36 kg structural mass

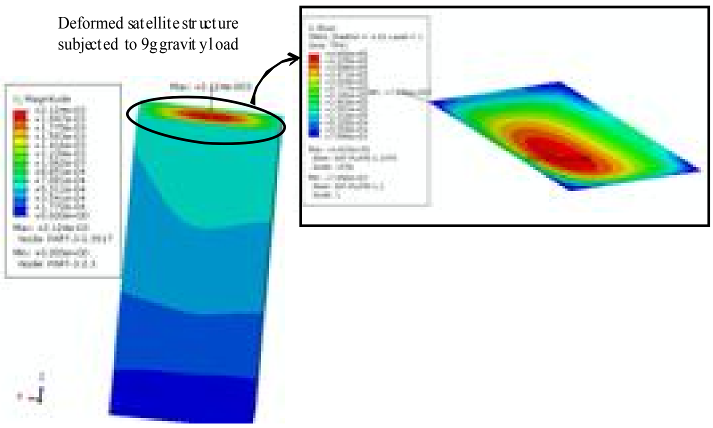


Figure 6 Deformation contours of whole satellite box and stress distribution in upper surface of satellite sandwich panel structure subjected to critical loading condition (Max. deformation: 2.12×10^{-3}mm, Max. equivalent stress: 4.6×10^{5}Pa)

Table 4 Non-zero natural frequencies of FEM models (Hz)

FEM Model	NF Mode#1	NF Mode#2	NF Mode#3	NF Mode#4	NF Mode#5
Critical sandwich panel of satellite structure	35.9	69.1	74.9	96.8	105
The whole satellite structure	25.5	33.23	34.01	34.43	34.71

Table 5 Comparison of maximum stress results of preliminary analytical results and FEM modelling

	Max. Stress	% Discrepancy based on AS
Analytical solution (AS) Eq. (6)	5.12×10^{5} Pa	
FEM model	4.6×10^{5} Pa	10

4. Manufacturing

The first step of the panel construction was to read the MSDS for materials and chemicals to be used. T300 carbon fiber and Araldite 5052 epoxy resin have been used to hand lay-up process and epoxy gel coat with Aerosile fillers and some other additives is used for core bonding purpose. The second step is to measure and cut a total of 8 sheet of the T300 carbon fiber plain weave of 0.15 mm thickness into 300 mm x 250 mm squares with proper wipe and weft directions.

Step three is to co-curing and bond the spliced core on the hand lay-upped T300/epoxy face sheet with 60% fiber fractions and place the half-prepared panel into the vacuum bag for curing process.

The fourth step is to hand lay-upping the second T300/epoxy face sheet on the half-prepared and cured panel and place the panel into the vacuum bag for preparing as shown in Figure 7. The average time for curing is approximately 10 to 12 hours.

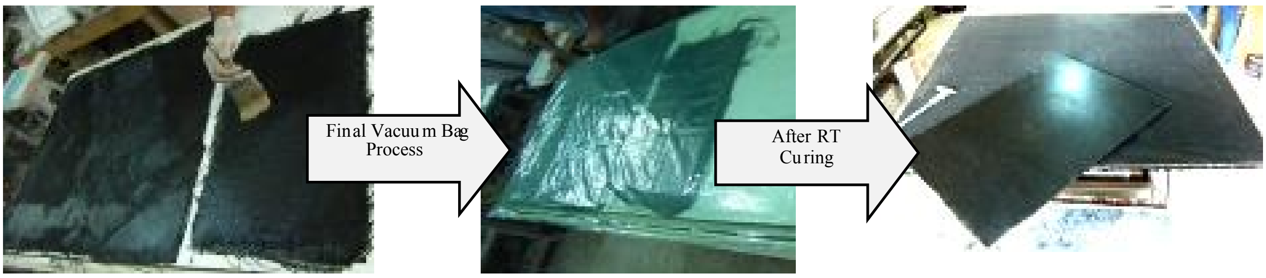


Figure 7 Lay-up process and preparing to sandwich panel vacuum bagging

5. Conclusion

This study will help to guide designers and manufacturing specialists to choosing the simple and effective methods of designing and manufacturing of primary structures of satellite sandwich panels .From the above analysis and manufacturing experiments following conclusions were made:

- Hence with the implementation of CFRP material the weight of structure can be minimized without affecting the performance.
- Because of difficult verification process in satellite structure design, preliminary design of satellite sandwich panel with presented analytical expressions and methods are acceptable in comparison with FEM results (about 10% discrepancy)
- Hand lay-up manufacturing process followed with vacuum, has the minimum visual defects in comparison with hand lay-up alone.
- Composite sandwich panels with CFRP face sheets are the best choice of aerospace composite materials in Iran, due to coasts benefits and feasible manufacturing.
- Galvanic corrosion is the main challenge in vicinity of CFRP face sheets and Aluminum honeycomb, so some parameters like service life time and isolator layer usage like fiber glass layer or appropriate scrim of adhesive film layer are very important to facing this problem. Because of the limited life time of model satellite and according to experimental sample tests on galvanic corrosion challenge, galvanic corrosion is

not the major problem and main end life failure parameter.

- According to coast manufacturing and experimental process, the presented simplified analytical expressions can be used as primary verification process in aerospace design process.

REFERENCES

1. F.C. Campbel, *ASM- composite materials*. ASM, 2010.
2. F.C. Campbell, *Manufacturing technology for Aerospace structural materials*. Elsevier, 2006.
3. Barbero EJ, *Introduction to Composite materials design*. USA, Edwards Brothers, 1998.
4. J.J. Wijker, *Spacecraft Structures*. Springer, 2008.
5. Hexcell, *Hexweb-Honeycomb Sandwich design technology*. hexcell, 2000.
6. Liang Huang, Abdul H. Sheikh, Ching-Tai Ng, Michael C. Griffith, "An efficient finite element model for buckling analysis of grid stiffened laminated composite plates," *Composite Structures*, (2014).
7. A. Stabile,S. Laurenzi, "Coiling dynamic analysis of thin-walled composite deployable boom," *Composite Structures*, 113, 429–436, (2014).
8. Chunjin Yu, Jin Zhang, Wei Song, Kehong Yi, "Deformation and stress Analysis of FlappingWing Aerial Vehicles Based on Composites Model," *Advanced Materials Research*, 1006-1007, 7-10, (2014).
9. G. Naveen Kumar, G. Bharath Reddy, K. ChandraShekar, "Dynamic Response of NACA 0018 for Car Spoiler using CFRP Material," *International Journal of Current Engineering and Technology* , 4,(2014).
10. P.Veeramuthuvel, S.Jayaraman, K.Shankar, M.Annadurai, A.K.Sharma, "Dynamic – Thermal Analyses of a structurally reconfigured Electronics Package onboard Mini Satellite," *Applied Mechanics and Materials*, 592-594, 2117-2121,(2014).
11. A. Ampatzoglou, A.Baltopoulos, A.Kotzakolios, V. Kostopoulos, "Qualification of Composite Structure for Cubesat Picosatellites as a Demonstration for Small Satellite Elements," *International Journal of Aeronautical Science & Aerospace Research*, 1, 1-10,(2014).