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Processing, Properties
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Edited by Sohel Rana and Raul Figueiro

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Woodhead Publishing Series in Composites
Science and Engineering: Number 70

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Sohel Rana

Raul Figueiro

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The Officers' Mess Business Centre, Royston Road, Duxford, CB22 4QH, UK
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The Boulevard, Langford Lane, Kidlington, OX5 1GB, UK

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British Library Cataloguing-in-Publication Data

A catalogue record for this book is available from the British Library

Library of Congress Cataloging-in-Publication Data

A catalog record for this book is available from the Library of Congress

ISBN: 978-0-08-100939-0 (print)

ISBN: 978-0-08-100054-0 (online)

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Publisher: Matthew Deans

Acquisition Editor: Gwen Jones

Editorial Project Manager: Charlotte Cockle

Production Project Manager: Omer Mukthar

Designer: Victoria Pearson Esser

Typeset by TNQ Books and Journals

*Dedicated to my mentor and dearest grandfather, **Md. Abdur Razzaque**
and my lovely wife, **Gargi** whose love and inspiration are always behind
every piece of my work, including this book.*

— Sohel Rana

*Dedicated to my wife **Ana** and my daughters **Carolina**, **Ana João**
and **Inês**...with you everything is possible!*

— Raul Figueiro

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Preface

Currently, the aerospace industry is looking for alternative materials to reduce fuel consumption and improve safety and performance. Fibre-reinforced composites offer a wide range of properties by combining fibres from different materials (eg, polymeric, inorganic or metallic), scales (macro, micro or nano) and forms (tubes, particles, short fibres, tows or fabrics) with various matrix materials. Advanced composite materials can address all the demands asked for in aerospace materials, and, consequently, their use in the aerospace industry is steadily increasing.

This book focusses predominantly on advanced composite materials and their use in aerospace applications. It discusses the basic and advanced requirements of these materials that are necessary for various applications in the aerospace sector. All the main types of commercial composites such as laminates, sandwich composites, braided composites, ceramic and metal matrix composites and carbon–carbon composites are discussed and compared to metals. Various aspects, including the type of fibre, matrix, structure, properties, modelling, testing and mechanical and structural behaviour are discussed. The introductory chapter (chapter: Advanced composites in aerospace engineering) provides brief discussion on the main requirements of the aerospace materials and how advanced composites can fulfil these demands. Also, this chapter introduces different types of composite materials which are already in use or have future prospects in the aerospace industry. Detailed discussions on these various composite materials are presented separately in subsequent chapters.

Besides materials, architecture of reinforcements also has strong influence on the performance of composites. The use of advanced fibrous architectures in composite materials is increasing to address the demands of advanced technical sectors. In consideration of this, chapter “Advanced fibrous architectures for composites in aerospace engineering” has been dedicated to advanced 2D and 3D fibrous structures manufactured by weaving and knitting technologies. Their production methods, properties and applications are discussed in detail.

There are several new types of composite materials that have huge potential for applications in the aerospace sector, such as nanocomposites, and multiscale and auxetic composites; these are also explored in detail. Like other industrial sectors, nanotechnology has also shown huge promise to develop outstanding high-performance and multifunctional materials. Nanocomposites and multiscale composites are such advanced materials with great prospects in the aerospace sector. The concepts, production, processing, challenges, modelling and applications of these composites are presented in chapters “Polymer nanocomposite: an advanced material for aerospace

applications and Multiscale composites for aerospace engineering”. Auxetic composites are also new and show excellent characteristics for considering them in aerospace applications. Details on these composites have been presented in chapter “Auxetic composites in aerospace engineering”.

Self-sensing of damage and its automatic healing are essential characteristics which can greatly improve the safety, reduce maintenance and improve performance of aerospace materials. Chapters “Self-sensing structural composites in aerospace engineering and Self-healing composites for aerospace applications” present detailed information on the concept, types, mechanisms and properties of various self-sensing and self-healing composites and their application in aerospace engineering. Looking at the environmental benefits and sustainability, natural fibre and bio-composites can also find potential applications in the aerospace industry, and these important materials are discussed in chapter “Natural fibre and polymer matrix composites and their applications in aerospace engineering”.

Quality control is an important task for materials used in aerospace engineering, which demands high safety. Quality control for aerospace materials begins with the inspection of the raw materials and continues through the monitoring of the production process and characterization of the final structural parts. All these essential steps of quality control for aerospace composites are discussed in detail in chapter “Quality control and testing methods for advanced composite materials in aerospace engineering”.

The book’s main strength is that it covers the design, processing, properties, modelling and application potential of these materials. What sets the book apart from other published titles is that it is completely dedicated to the use of advanced composite materials in aerospace applications. It discusses the overall requirements for these materials, the different types that are currently available, structures, properties, testing, modelling and product design examples. The book discusses all the existing types of advanced composite materials currently available as well as novel materials that are still in the initial stages of research. We strongly believe that this book will be essential reading for engineers and scientists currently working in this field of research as well as students taking selected courses on advanced composite materials.

We are highly indebted to all authors for their excellent efforts and valuable contributions. We also would like to thank the Fibrenamics Research Group and Centre for Textile Science and Technology, University of Minho for providing all the resources required for the preparation of the book. Sincere thanks are also due to our colleagues (especially Mr Subramani Pichandi) for their kind help in the preparation of the artwork and formatting of the book chapters. We also acknowledge the great help and cooperation from the publishing team (Gwen Jones, Steven Mathews, Kate Hardcastle and Charlotte Cockle) of Elsevier Publishing Ltd.

Sohel Rana and Raul Figueiro

University of Minho

Sandwiched composites in aerospace engineering

5

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5.1 Introduction

One of the biggest challenges to the design engineer in the aircraft and space industries is weight reduction. In these markets, each kilogram costs large amounts of money, and sandwiched composites became one of the most efficient solutions to conjugate the highest bending stiffness and strength-to-weight ratios in structural components (Noor, 2000; Marasco, 2005).

Typical sandwich composites consist of two thin, stiff, high-strength facing skins separated by a thick and light core (see Fig. 5.1). The skins are bonded to the core using structural adhesives strong enough to transfer loads between the two faces. The core acts like an I-beam's web, which means taking the shear loads as well as providing structural rigidity by keeping high-strength facing skins away from the neutral axis where the higher tensile and compressive stresses are developed. Sandwich structures differ from the I-beam ones mainly because they have the web spread over the entire cross-section, which provides much higher torsional rigidity than a concentrated middle-section web may offer in an I-beam (Zenkert, 1995; Mazumdar, 2001).

This chapter will review the main raw materials, manufacturing technologies and design methods applied in sandwich composite structures.

Figure 5.1 Typical sandwich composite panel.

Adapted from Sezgin, F.E., 2011. Mechanical Behavior and Modeling of Honeycomb Cored Laminated Fiber/Polymer Sandwich Structures. Master Thesis, İzmir Institute of Technology, Turkey.

5.2 Sandwiched composite structures

5.2.1 Raw materials

The relevant properties of major raw materials used as facing skins, cores and adhesives in sandwich composite structures for aircraft and aerospace applications are summarized in this section.

5.2.1.1 Facing skins

Facing skins usually consist of polymer matrix-based fibre-reinforced composite laminates. Typical, such laminates are manufactured by using hand lay-up, autoclave, vacuum-assisted resin infusion, compression moulding, resin transfer moulding (RTM) or automated tape-laying technologies to stack polymer-based prepregged materials, or prepregs (unidirectional tapes, woven fabrics etc.) (Bitzer, 1997).

The selection of the matrix must be carefully considered in order to achieve the required surface properties (corrosion, temperature behaviour etc.) and manufacturing type of the composite material. Both thermosetting and thermoplastic resins are commonly used as matrices (see Table 5.1).

Epoxy, unsaturated polyester and vinyl ester are the most widely used thermosetting resins. Epoxies are almost always preferred for advanced applications, such as the aerospace and aircraft markets, where lower weight, high strength and dimensional accuracy are much more critical. Epoxies are also tougher, and they present better behaviour to heat distortion and much less shrinkage than unsaturated polyester and vinyl ester resins. Being less expansive, easy to process and superior in corrosion resistance, unsaturated polyester resins are much better adapted to more commercial and less demanding markets. Vinyl esters usually present higher corrosion behaviour than unsaturated polyesters and may be considered an intermediary solution between the epoxies and unsaturated polyesters.

Currently, thermoplastics are applied as matrix in facing-skin laminates. Compared to brittle thermosets, they present higher toughness, ductility and failure strains; enhance the fatigue, corrosion and thermal behaviours (polyetherimide (PEI) and polyetheretherketone (PEEK)); and can be much more easily recycled and reprocessed. However, it is still very difficult and expensive to manufacture stacking laminates using thermoplastic matrices because their processing requires the use of much higher temperatures and pressures. Heated compression moulding and automated tape-laying processing are the major technologies used to produce thermoplastic matrices in facing skins of sandwich composite structures.

Table 5.1 exhibits the main properties of the polymer matrices mostly used in the facing skins of sandwich structural panels. In addition to the mechanical properties, the maximum service temperature is many times the most relevant criterion for selecting the polymer matrix material to be applied in advanced composite structures for aircraft and aerospace applications.

Carbon fibres, due to their extremely low weight, high stiffness and high tensile strength, are by far the most common to be chosen to reinforce facing skin laminates

Table 5.1 Main polymers used as matrices in sandwich-facing skins

Material	Density (Mg/m ³)	Tensile modulus (GPa)	Tensile strength (MPa)	Maximum service temperature (°C)
Thermosetting resins				
Epoxy	1.2–1.4	2.5–5	50–130	80–215
Unsaturated polyester	1.1–1.4	1.6–4.1	35–95	60–150
Vinyl ester	1.1–1.3	3.0–3.5	73–81	60–150
Phenolic	1.1–1.4	2.7–4.1	35–60	70–120
Cyanate esters	1.1–1.3	2.6–3.5	70–125	150–250
Bismaleimide	1.2–1.3	3.9–4.6	120–180	230–320
Thermoplastics				
Polypropylene (PP)	0.9–0.91	1.15–1.57	31.0–41.9	50–135
Polyamide/(nylon) (PA)	1.12–1.42	0.25–3.84	35.2–167.6	75–150
Polycarbonate (PC)	1.20	2.20–2.41	62.7–73.3	115
Polysulphone (PSU)	1.24–1.25	2.51–2.72	60.0–74.7	150
Polyethersulphone (PES)	1.37–1.46	2.44–2.86	68.3–100.6	180
Polyimide (PI)	1.33–1.43	2.10–4.02	72.4–119.4	170
Polyphenylene sulphide (PPS)	1.35	3.28–3.42	48.3–87.3	120–220
Polyetherimide (PEI)	1.26–1.27	2.72–4.02	62.1–150.2	170
Polyether ether ketone (PEEK)	1.3–1.32	3.63	70.3–104.8	120–250

Adapted from Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press; Muzzy, J.D., 2000. Thermoplastics – properties. In: Kelly, A., Zweben, C. Comprehensive Composite Materials. Vol. 2. Elsevier Ltd, pp. 57–76; Peters, S.K. (Ed.), 1998. Handbook of Composites, second ed. Chapman & Hall.

of sandwich structures used in aerospace and aircraft applications. Glass and aramid fibres are other reinforcements that may be used as alternatives. Glass fibres may be used in price-competitive parts that require less stiffness due to their much lower cost.

Aramid fibres, which are based on aromatic polyamide precursors and present the lowest density, may be used in facing skin laminates requiring superior wear and impact behaviour due to their excellent toughness and abrasion properties. However, the major limitations to the application of aramid fibres are their much higher price (in comparison to glass fibres), low service temperature, compressive properties and well-known lack of adhesion properties.

The typical properties of the most frequently used reinforcing fibres are shown in Table 5.2. As may be seen, carbon fibres exhibit the highest specific modulus ratio between the tensile modulus and density.

Table 5.2 Properties of major fibres used as reinforcements in sandwich skins

Type of fibre	Density (Mg/m ³)	Tensile modulus (<i>E</i>) (GPa)	Tensile strength (σ) (MPa)	Specific modulus (<i>E</i> / ρ) (GPa m ³ /Mg)	Specific strength (σ/ρ) (MPa m ³ /Mg)	Melting point (°C)	Relative cost
E-glass	2.54	70	3450	27.6	1385	≥1540	Low
S-glass	2.50	86	4500	34.4	1800	≥1540	Moderate
HM-carbon (high modulus)	1.90	400	1800	210.5	947	≥3500	High
HS-carbon (high strength)	1.70	240	2600	141.2	1529	≥3500	High
Boron	2.6	400	3500	153.8	1346	2300	High
Aramid (Kevlar [®] 29)	1.45	80	2800	55.2	1931	500	Moderate
Aramid (Kevlar [®] 49)	1.45	130	2800	89.7	1931	500	Moderate

Adapted from Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press; Peters, S.K. (Ed.), 1998. Handbook of Composites, second ed. Chapman & Hall.

The prepregs, which present the reinforcing fibres already impregnated with the polymer matrix, are commonly used to manufacture composites of sandwich-facing skins for aircraft and aerospace structural components by using the autoclave, compression-moulding or tape-laying technologies.

Thermosetting matrix prepregs must be kept in refrigerators at low temperature to avoid the resin cure completion. In these prepregs, after impregnating the reinforcing fibres with the thermosetting matrix resin, the cure reaction is suspended by keeping the material at below-freezing temperatures. Due to that and contrarily to thermoplastic matrix prepregs, the thermosetting ones always present a limited shelf life. The typical properties of several prepreg materials that can be applied in the facing-skin sandwich composite structures may be seen in [Table 5.3](#), which shows that prepregs can use both unidirectional fibres and woven fabrics.

5.2.1.2 Sandwich cores

While facing-skin laminates should withstand both in-plane and bending (primary loading) efforts in sandwich structures, the core must carry transverse shear loads and the compressive and crashing efforts placed on the panels. The core must also ensure an adequate transfer of loads between the two facing-skin laminates through the structural adhesives used.

Foam, solid and honeycomb cores are relatively inexpensive and may consist of balsa wood, plywood and an almost infinite selection of open- and closed-cell thermoplastic and thermosetting foams, presenting a wide variety of densities and shear moduli, and rigid, flexible or rubbery behaviour ([HexWeb](#); [Beckwith, 2008](#)) (see [Figs. 5.2 and 5.3](#)).

Honeycomb architectures are widely used in aerospace, aircraft, transportation and marine structural applications, and the two most common types are hexagonal and square-shaped cell structures. They consist of a series of cells, nested together to form panels similar in appearance to the cross-sectional slice of a beehive. In its expanded form, honeycomb is 90–99% open space. Honeycomb is fire retardant, flexible and lightweight, and it has good impact resistance. It offers the best strength-to-weight ratio of the core materials. Honeycomb cores use both metallic and nonmetallic materials.

Aluminium is almost always the only metal used in the manufacture of honeycomb cores. Nonmetallic honeycomb cores may be made from paper or card, for less load-demanding applications, and from aramid (Nomex[®]), thermoplastic (polycarbonate, polypropylene, polyetherimide etc.) and other materials, such as glass-reinforced polyimide and phenolic composites [Figs. 5.4 and 5.5](#).

Two main methods are used to manufacture honeycomb cores: expansion and corrugation. Expansion is more common, and it is used for making aluminium and aramid (Nomex[®]) honeycombs. The process consists of stacking together sheets of material in a block form. Before stacking, adhesive nodelines are printed on the sheets to obtain interrupted adhesive bonding. After curing the stack of sheets, slices of appropriate thickness are cut from the block and then expanded to obtain the desired cell size and shape.

Table 5.3 Properties of prepregs applied in sandwich skins

Type of prepreg	Fibre volume fraction (%)	Processing temperature (°C)	Tensile modulus (E) (GPa)	Tensile strength (σ) (MPa)	Maximum service temperature (°C, dry)	Shelf life (–17.8°C, months)	Out time at room temperature (days)
Thermosetting matrix prepregs							
Unidirectional carbon (AS4 T300)–epoxy	55–65	120	103–152	1241–2206	85–120	6–12	14–30
Unidirectional carbon (IM7)–epoxy	55–60	120	138–172	2206–3034	120	12	30
Unidirectional S2-glass–epoxy	55–63	120–177	41–55	827–1586	85–120	6	10–30
Unidirectional aramid (Kevlar®)–epoxy	55–60	120–141	69	965	85	6	10–30
Unidirectional carbon (AS4)–bismaleimide	55–62	177–246	103–152	1379–2206	232–316	6	25
Unidirectional carbon (IM7)–bismaleimide	60–66	177–227	138–172	2620–2758	232–316	6–12	25
Unidirectional carbon (IM7)–cyanate ester	55–63	120–177	138–172	690–2723	232	6	10
Unidirectional S2-glass–cyanate ester	55–60	120–177	48	1241	204	6	10
Plain-weave fabric carbon (AS4)–epoxy	57–63	120	55–70	517–855	–	6	10

Plain-weave woven fabric aramid (285K–4HS)– epoxy	60	120–140	30–31	500	85	6	10–30
Plain-weave woven fabric S-2 glass–epoxy	50–55	120	20–34.5	552–600	85	6	10
Plain-weave woven fabric glass (7781-8HS)– phenolic	55	—	20	400	—	—	—
Unidirectional carbon (IM7)–PEEK	57–63	288	179	2827	177	Unlimited	Unlimited
Unidirectional carbon (G34/700)–PA (nylon 6)	55–62	232–260	110	1489	93	Unlimited	Unlimited
Unidirectional aramid/PA (nylon 12)	52	204	47	1413	—	Unlimited	Unlimited
Unidirectional carbon (AS4)–PPS	64	232–271	121	1965	—	Unlimited	Unlimited
Unidirectional carbon (IM7)–polyimide (PI)	62	321–352	172	2620	204	Unlimited	Unlimited
Plain weave fabric carbon HM (T650-35)– polyimide (PI)	58–62	349–388	69–124	896–1069	260–316	12	Unlimited

Adapted from Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press; HexWeb™ Honeycomb Sandwich Design Technology. Hexcel® Technology Manuals. <<http://www.hexcel.com/resources/technology-manuals>> (visit on 19.03.15.).

Figure 5.2 Balsa wood core BALTEK[®] SBC from Airex AG Industries [Balsa Wood Core](#).

Figure 5.3 Polymer foams: (a) polyvinyl chloride (PVC) foams; (b) polyurethane foams.

Figure 5.4 Aluminium honeycombs from Alcore Inc. — A Gill Company ([Aluminium Honeycombs](#)).

In corrugation, the sheet of material is given a corrugation form by passing through corrugating rolls. The corrugated sheets are then stacked together, bonded and cured. Finally, the honeycomb panels are cut from the block into the desired shape and size without any expansion.

In contrast, the techniques mostly used to manufacture foam cores include gas injection, blowing agents, expandable bead processes and so on. In all these processes,

Figure 5.5 Aramid (Nomex[®]) honeycombs.

one gas or blowing agent is supplied to the thermoplastic or thermosetting resins to decrease its density by forming closed or open gaseous cells. The purpose of the foam is to increase the bending stiffness and thickness of the structural sandwich without increasing its weight.

The main properties of several core materials used in composite sandwich structures are summarized in [Table 5.4](#).

Structural adhesives

The most common structural adhesives used in composite sandwiches are epoxy, acrylic and urethane resins. Typically, they are classified in three categories: (1) two-component mixed adhesives, (2) two-component unmixed adhesives and (3) one-component unmixed adhesives ([Mazumdar, 2001](#)).

The majority of epoxy and polyurethane adhesives are two-component mixed adhesives, and the acrylic and anaerobic (urethane methacrylate ester) adhesives fall into the category of two-component unmixed adhesives. The two-component mixed adhesives require the prior mixing of two components before application to the substrate surface. Once the two components are mixed, there is a limited pot life.

In two-component unmixed adhesives, the adhesive is applied on one substrate surface, and an activator, usually in a very small amount, is applied on the other substrate surface. When the two surfaces are put together, the adhesive cures by the reaction between both components.

Finally, one-component unmixed adhesives do not require mixing. Almost all these adhesives consist of a premix of two or more components such as resin, curing agent, fillers and additives. Epoxies, polyurethanes, cyanoacrylates and hot-melt, light-curable and solvent-based adhesives are examples of this type of structural adhesive.

[Table 5.5](#) summarizes some of the main guidelines to allow selecting the appropriate adhesive to be used on a sandwich composite panel for a specific application.

Table 5.4 Properties of core materials used in sandwich composites

Honeycomb density and cell size		Compressive properties		Plate shear behaviour			
Density	Cell size ^a	Stabilized		Length direction		With direction	
Kg/m ³	mm	Strength MPa	Modulus MPa	Strength MPa	Modulus MPa	Strength MPa	Modulus MPa
3003 Aluminium							
29	19	0.9	165	0.65	110	0.4	55
37	9	1.4	240	0.8	190	0.45	90
42	13	1.5	275	0.9	220	0.5	100
54	6	2.5	540	1.4	260	0.85	130
59	9	2.6	630	1.45	280	0.9	140
83	6	4.6	1000	2.4	440	1.5	220
5052 Aluminium							
37	6	1.35	310	0.96	220	0.58	112
50	5	2.3	517	1.45	310	0.9	152
54	6	2.6	620	1.6	345	1.1	166
72	3	4.2	1034	2.3	483	1.5	214
83	6	5.2	1310	2.8	565	1.8	245
127	6	10.0	2345	4.8	896	2.9	364
130	3	11.0	2414	5.0	930	3.0	372

5056 Aluminium							
37	6	1.8	400	1.2	220	0.7	103
50	3	2.4	669	1.7	310	1.1	138
50	5	2.8	669	1.8	310	1.0	138
72	3	4.7	1275	3.0	483	1.7	193
Aramid HRH10 Nomex®							
29	3	0.9	60	0.5	25	0.35	17.0
32	5	1.2	75	0.7	29	0.4	19.0
32	13	1.0	75	0.75	30	0.35	19.0
48	3	2.4	138	1.25	40	0.73	25.0
48	5	2.4	140	1.2	40	0.7	25.0
64	3	3.9	190	2.0	63	1.0	35.0
64	6	5.0	190	1.55	55	0.86	33.0
80	3	5.3	250	2.25	72	1.2	40.0
96	3	7.7	400	2.6	85	1.5	50.0
123	3	11.5	500	3.0	100	1.9	60.0
144	3	15	600	3.5	115	1.9	60.0
29	5 OX	1.0	50	0.4	14	0.4	21.0
48	5 OX	2.9	120	0.8	20	0.85	35.0
Polycarbonate (PC)							
127	3	4.8	379	—	—	—	—
48	6	0.8	103	—	—	—	—

Continued

Table 5.4 Continued

Honeycomb density and cell size		Compressive properties		Plate shear behaviour			
Density	Cell size ^a	Stabilized		Length direction		With direction	
Kg/m ³	mm	Strength MPa	Modulus MPa	Strength MPa	Modulus MPa	Strength MPa	Modulus MPa
Polyetherimide (PEI)							
60–110	—	0.8–16	160–290	0.8–1.4	18–30	0.8–1.4	18–30
Glass-reinforced polyimide							
128	5	9.0	869	—	—	—	—
Glass-reinforced phenolic							
64.1	5	3.3	—	1.45	79.3	0.76	34.5
88	5	5.2–6.5	655	2.55	134.5	1.31	58.6
56.1	6	2.8	—	1.2	62	0.69	24
72	6	3.9	—	1.7	96.5	0.97	41.4
Paper honeycomb							
80	6	2.8	—	1.3	208	0.57	45
35	12.7	0.97	—	0.54	82	0.28	30.3
thead1Phenolic foam (Gillfoam[®])							
112	—	1.3	—	0.71	24.8	0.71	24.8
160	—	2.3	—	1.1	44.1	1.1	44.1
288	—	8.7	—	2.2	65.5	2.2	65.5

Polyurethane foam							
32	—	0.19	—	0.14	1.6	0.14	1.6
64	—	0.55	—	0.33	5.2	0.33	5.2
96	—	0.97	—	0.62	10.3	0.62	10.3
320	—	5.9	—	3.1	103	3.1	103
PVC foam, closed cell							
56	—	0.76	—	0.53	12.4	0.53	12.4
99	—	1.4	—	0.83	15.2	0.83	15.2
400	—	5.8	—	4.5	108	4.5	108
Balsa wood end grain							
96	—	5.2	—	0.97	110	0.97	110
152	—	10.3	—	1.5	193	1.5	193
250	—	26.6	—	4.9	312	4.9	312
Polystyrene (PS) foam							
30–60	—	0.3–0.9	—	0.25–0.60	4.5–20	0.25–0.60	4.5–20
Polymethacrylimide (PMI) foam							
30–300	—	0.8–16	—	0.8–7.5	19–290	0.8–7.5	19–290
Epoxy foam							
80–320	—	0.62–7.4	—	0.45–5.2	—	0.45–5.2	—
Carbon–graphite foam							
30–560	—	0.2–60	—	0.05–3.9	—	0.05–3.9	—

^aCell size has imperial units, and presented metric values must be considered approximated.

Adapted from Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press; HexWeb™ Honeycomb Sandwich Design Technology. Hexcel® Technology Manuals. <<http://www.hexcel.com/resources/technology-manuals>> (visit on 19.03.15.); Beckwith, S.W., July/August 2008. Sandwich core materials & technologies – Part I. SAMPE Journal 44 (4); Doorway, M.C.G., 1997. Simplified Sandwich Panel Design 34 (3), M.C. Gill Corp.

Table 5.5 Adhesive selection guidelines

Characteristics	Standard epoxies	Urethane	Acrylic	Silicones	Polyolefins (vinylics)
Adhesive type ^a	L1, L2, F	L, W, HM	L1, L2, W	L1, L2	F
Cure requirement	Heat, ambient	Heat, ambient	Heat, ambient	Heat, ambient	Hot melt
Curing speed	Poor	Very good	Best	Fair	Very good
Substrate flexibility	Very good	Very good	Good	Good	Fair
Shear strength	Best	Fair	Good	Poor	Poor
Peel strength	Poor to fair	Very good	Good	Very good	Fair
Impact resistance	Fair	Very good	Fair	Best	Fair
Humidity behaviour	Poor	Fair	Fair	Best	Fair
Chemical resistance	Very good	Fair	Fair	Fair	Good
Temperature resistance	Fair	Fair	Fair	Good	Poor
Gap filling	Fair	Very good	Very good	Best	Fair
Storage (months)	6	6	6	6	12

^aAdhesive type: L1 = Liquid one part; L2 = Liquid two part; F = Film; W = Waterborne; HM = Hot melt.

Adapted from Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press.

5.2.2 Production methods

As mentioned in [Section 5.2.1](#), expansion and corrugation as well as gas injection, blowing agents and expandable bead processes in the case of foamed materials are the most commonly used processes to manufacture sandwich cores.

Concerning the production of final structural sandwich composite parts, there are two main possibilities: the conventional manufacturing of a composite sandwich is achieved by adhesive joining separately prepared skins to cores. The joining process during sandwich fabrication can require strict quality control. The joining process can be eliminated when the sandwich structures are manufactured by a co-cure method inside a mould. With this method, the sandwich panel adhesive and composite materials are cured simultaneously.

The most widely used production technologies include: vacuum bagging and autoclave, liquid moulding technologies (RTM, structural reaction injection moulding (S-RIM) and infusion), compression moulding, filament winding, wet lay-up and adhesive bonding. From an economical point of view, the continuous lamination manufacturing process is naturally preferable.

5.2.2.1 Liquid moulding technologies

The liquid moulding technologies include RTM, S-RIM and vacuum-injection moulding technologies, also known as infusion technologies. All these technologies are based on placing the reinforcements in the mould without any resin that is infused into the reinforcement fabrics by using pressure or vacuum. Using closed moulds, the liquid moulding technologies avoid much of closed moulds' styrene emissions problems and are suitable for producing complex shapes. In RTM, the reinforcement fabrics or mats, core materials and eventual inserts are conveniently placed in the mould that is then closed. Afterwards, the resin system is pressurized, using an injection machine that pumps and mixes in the right proportions the resin and catalysts, until full impregnation is obtained (see Fig. 5.6). The impregnation time must be lower than the resin gel-time. RTM relies on the mould-clamping structure being stiff enough to withstand the pressure of the injected resin without opening or distorting the mould. This aspect of the process can become problematic if large-scale components are moulded, with tooling sometimes becoming uneconomic simply due to its handling and clamping requirements.

Vacuum infusion (Fig. 5.7) is a wet lay-up process that is typically used to manufacture composite laminates where reinforcements are impregnated by thermosetting liquid resins. The resin sucked by vacuum is introduced into the mould, being also the bag compacted down against the laminate.

First, the reinforcement fabric layers and core are dried, laid up and precisely positioned in the mould. This may be performed manually or by automation (using robots). Next, when the resin is sucked into the mould, as the laminate is already compacted there is no room for excess resin. As very high resin-to-fibre ratios are obtained in the vacuum infusion process, high mechanical properties are usually obtained in the final composite laminate. Vacuum infusion is suitable for moulding very large structures and is considered a low-volume moulding process.

The mould may be gel-coated in the traditional fashion. After the gel coat cures, the dry reinforcement is positioned in the mould. This includes all the plies of the laminate

Figure 5.6 Resin transfer moulding (RTM) (Advanced Composite Manufacturing Centre).

Figure 5.7 Vacuum infusion (van Paepegem).

and core material if required. A perforated release film is placed over the dry reinforcement. Next, a flow medium consisting of a coarse mesh or a ‘crinkle’ ply is positioned, and perforated tubing is positioned as a manifold to distribute resin across the laminate. The vacuum bag is then positioned and sealed at the mould perimeter. A tube is connected between the vacuum bag and the resin container. A vacuum is applied to consolidate the laminate, and the resin is pulled into the mould. In RTM and vacuum injection moulding, conventional resin systems also used in the wet lay-up process are used. In contrast, S-RIM employs highly reactive resins that are mixed right before injection. Also, RTM and S-RIM technologies use matched moulds, whereas vacuum infusion employs a one-side mould that can sometimes be seen as a modified version of the wet lay-up technology with the mould covered by a vacuum bag to reduce the composite voids content. The filling of resin time of RTM and vacuum infusion is in the range of a few minutes to a few hours, enabling the production of large components before increasing resin viscosity prohibits further impregnation. In the S-RIM technology, the resin filling time is usually less than a minute due to the much higher resin reactivity, meaning that only smaller components than those used with the other two processes can be produced. The component extraction time (demoulding) can vary from a few minutes to hours in the case of RTM, a few hours if vacuum infusion is considered and only a few minutes in the case of S-RIM. In S-RIM, the two-component resin is mixed right before injection in an impingement-mixing nozzle using dedicated pumps. In all processes, multiple inlet points may be necessary for large-component production.

5.2.2.2 Bagging and autoclaving

High-performance composite materials are conventionally processed in autoclaves which require high capital investment for the equipment. The equipment cost increases

Figure 5.8 Vacuum bag lay-up example (Hexply).

exponentially with the size of the autoclave, which is one of the main constraints on the ability to fabricate large composite structures. Both monolithic laminates and sandwich structures are usually laid up using prepregs. The use of prepregs ensures good impregnation, and resins used in prepregs also tend to have better properties than the ones available for wet lay-up. Sandwich constructions can be manufactured by autoclave or vacuum bag moulding. For autoclave processing, sandwich constructions can usually be laid up and cured as a single-shot process. However, for the vacuum bag curing of large components, it may be necessary to lay up and cure in two or more stages. When using autoclave, the use of excessive pressures can lead to eventual core crush.

The vacuum bag technique involves placing and sealing a flexible bag over a composite lay-up (see Fig. 5.8), evacuating all the air from under the bag. The removal of air forces the bag down onto the lay-up with a consolidation pressure of up to 1 atm. The completed assembly, with vacuum still applied, is placed inside an oven or on a heated mould, and the composite is produced after applying an appropriate temperature curing cycle.

Autoclave processing is used for the manufacture of high-quality structural components containing high fibre volume and low void contents. The autoclave technique requires a similar vacuum bag (Fig. 5.8), but the oven is replaced by an autoclave (Fig. 5.9). It allows the application of atmospheric pressure and provides the curing conditions for the composite where the application of vacuum, pressure, heat-up rate and cure temperature are controlled. Typically, high processing pressures are used for thick sections of complex shapes and lower pressures are applied for the production of sandwich structures. The curing cycle can be very long if slow heat-up rates are required to guarantee even temperature distribution on the tooling and composite components.

Manufacturing of sandwich structures made from prepregs may be accomplished in two manners. In some cases, the laminate may be laid directly onto the core in a way similar to that used in wet hand lay-up. Alternatively, previously manufactured skin laminates may be adhesively bonded to the core in a separate process. Sandwich composites manufactured using prepreg lay-up are characterized by good to excellent mechanical properties, low void contents in the laminate skins and consistent laminate quality.

Figure 5.9 Autoclave process (Hexply).

5.2.2.3 Compression moulding

The compression-moulding technology used to manufacture sandwich structures is similar to the one used in the production of monolithic composites. If thermoplastic skins are to be used, the process starts by preheating the skins in an oven and, immediately after, placing them in a cooled mould with the core sandwiched in between. The mould must be closed fast (this requirement is essential) and pressure applied, forcing the skins to bond to the core and allowing a good surface finishing. The core material must withstand the moulding pressure and, if it is of thermoplastic nature, may further enhance bonding since heating of the face sheets will cause the core surface to melt.

This technology can be used to promote the bonding of thermosetting skins to a core panel, obtaining sandwich panels of good quality.

5.2.2.4 Wet lay-up

This is the oldest and one of the most flexible technologies for processing composites and sandwich components with composite. The wet lay-up may be performed either by hand lay-up or spray-up using labour-intensive work. This technology uses a single-side mould, which is first covered by a mould release agent. Usually, a gel coat is deposited directly onto the mould, allowing the laminate to have good environmental resistance and producing a smooth, cosmetically appealing surface. Next, the reinforcements in the form of mats (woven, knitted, stitched or bonded fabrics) are impregnated by hand with resin, with the help of brushed or nip-roller-type impregnators, for forcing the resin into the reinforcements. In the case of spray-up, fibre is chopped in a handheld gun and fed into a spray of catalysed resin directed at the mould. Laminates can cure at atmospheric conditions or with the help of vacuum bagging. The main differences between hand laid-up and sprayed-up composites are due to the differences in labour costs and mechanical properties. The lower labour

costs of spray-up implies that longer series are economically feasible, and the inferior mechanical properties achieved mean that this approach is used more commonly for commodity-type products.

5.2.2.5 Filament winding

Filament winding is a fabrication technique involving wrapping pretensioned, resin-saturated continuous reinforcements around a rotating symmetrical mould (Fig. 5.10). By changing the rotational velocity of the mandrel with the linear velocity of the guiding system, different fibre patterns can be obtained allowing the control of specific mechanical properties of the obtained composite structure. Recent filament winding equipment always uses computerized numerical control systems that can control up to six axes. Existing software can adjust all settings such as wind angle, fibre bandwidth and desired laminate thickness.

The filament winding of a sandwich structure, usually a circular pipe, begins by producing the filament-wound inner shell face and then stopping the equipment until a flexible or prefoamed core is applied to the produced skin. Then, the machine is restarted and the outer skin is produced.

5.2.2.6 Adhesive bonding

Using the bonding technique, adhesive layers are interleaved between the faces and the core, and the whole stack is subjected to a temperature and pressure cycle, depending on the adhesive resin used. For demanding applications, pressure is applied using vacuum bagging and autoclave, whereas for current applications only vacuum bagging will be enough. Hydraulic presses can also be used for pressure application in adhesive bonding. Laminate skins for bonding are normally produced using 'peel ply' fabric that is removed before bonding, ensuring a proper rugged surface. Also, it is normally necessary to prepare the surfaces to be bonded. Depending on the substrate, surfaces are prepared by one of the following procedures: (1) degrease; (2) degrease, abrade and remove loose particles; or (3) degrease and chemically pretreat. The processing sequence normally involves removing peel ply from the composite skins, preparing surfaces, applying adhesive film and placing the core material between the skins.

Figure 5.10 Filament winding (Nuplex).

Then, using vacuum bagging and an autoclave or an oven, apply the specified vacuum, pressure and temperature cycles. The adhesives can be used in film or liquid form, depending on the application, and are usually epoxy or polyurethane. Sandwich components manufacturing through adhesive bonding is characterized by good to excellent mechanical properties.

5.2.2.7 Continuous lamination

Continuous lamination is generally used for the large-scale manufacture of long production runs with a standard design. Sandwich panels made by continuous lamination are manufactured by bonding steel, aluminium or composite foil skins to polyurethane or polystyrene foam, mineral wool or other insulating cores. Recent continuous panel laminators or double-belt laminators have been designed to increase production speed, allowing heating and cooling of the material while at the same time subjecting it to a specified pressure. Two rolls of face sheets are first uncoiled and guided in between the belts of the press. The core is then inserted between the face sheets together with the adhesive layer. The bonding of the skins to the core material is achieved by applying convenient heating and cooling temperatures and pressure values. At the final stage, the panels will be cut at the desired lengths. The core materials can be fed in discrete blocks or, alternatively, in a continuous way by placing in situ core foam in between the skins, through injection and subsequent expansion. Another possibility is to sandwich a thermoplastic polymer film containing a foaming agent between the skins. As the polymer film melts, the foaming agent is free to expand, filling the gap between the face skins with a core foam.

5.2.3 Composite sandwich properties

Tables 5.6–5.9 present main basic, physical and mechanical properties of selected composite sandwich panels used by some major aircraft industries. Some of the characteristics presented were determined by those industries' own testing methods.

5.2.4 Major applications

Table 5.10 presents typical applications of sandwich composites in major markets.

Figs. 5.11–5.13 also show examples of applications of sandwich composites in satellites and the aerospace, aeronautical and naval industries. In satellites, Fig. 5.11 presents: (1) solar panels: carbon–epoxy preregs as skins, aluminium honeycomb and film adhesive; (2) antenna reflectors: aramid–epoxy and carbon–cyanate preregs in the skins and aramid–aluminium honeycomb; and (3) satellite structure: carbon–epoxy prepreg, aluminium honeycomb and film adhesive.

The following applications in the aerospace industry may be seen in Fig. 5.12(a): (1) carbon–epoxy preregs, aluminium honeycomb and adhesives in fairings; (2) the external payload carrier assembly (Speltra); (3) carbon–epoxy prepreg in the EPS ring; (4) front skirt; (5) glass–epoxy and nonmetallic honeycomb in the booster capotage;

Table 5.6 Basic properties of sandwich panels used by Boeing and Douglas aircraft

Product	Core type and density	Adhesive	Facing skins	Panel thickness	Panel weight	Flexural strength supported in 2 points (508 mm span)		Climbing drum peel strength
				(mm)	(N/m ²)	Test method: military standard 401B		
						Ultimate load (N)	Deflection at 4362.2 N (mm)	(mm N/76.2 mm width)
4409 Ty II	3 mm cell; 144.2 kg/m ³ aramid honeycomb	Modified phenolic	0.25 mm unidirectional carbon—phenolic	10.2	26.3	15,093.3	10.3	24,043.6
4409 Ty III	3 mm cell; 80.1 kg/m ³ aramid honeycomb	Modified phenolic	0.25 mm unidirectional carbon—phenolic	10.2	21.1	11,734.4	10.6	24,486.7
4417 Ty I	3 mm cell; 80.1 kg/m ³ aramid honeycomb	Modified epoxy	0.38 mm unidirectional FRP—epoxy	10.1	24.7	11,908.9	20.2	35,456.1
4417 Ty II	3 mm cell; 144.2 kg/m ³ aramid honeycomb	Modified epoxy	0.38 mm unidirectional FRP—epoxy	10.1	30.1	12,694.1	20.1	38,780.2
4417 Ty III	3 mm cell; 144.2 kg/m ³ aramid honeycomb	Modified epoxy	0.56 mm unidirectional FRP—epoxy	10.2	36.3	16,663.7	13.7	34,348.2
4417 Ty IV	3 mm cell; 80.1 kg/m ³ aramid honeycomb	Modified epoxy	0.38 mm unidirectional FRP—epoxy	16.7	30.5	22,596.3	6.6	35,456.1
4509 Ty 1	3 mm cell; 128.2 kg/m ³ aramid honeycomb	Modified phenolic	0.30 mm unidirectional carbon—phenolic	10.0	25.0	21,811.1	6.7	28,253.9

Continued

Table 5.6 Continued

Product	Core type and density	Adhesive	Facing skins	Panel thickness	Panel weight	Flexural strength supported in 2 points (508 mm span)		Climbing drum peel strength
				(mm)	(N/m ²)	Test method: military standard 401B		
						Ultimate load (N)	Deflection at 4362.2 N (mm)	(mm N/76.2 mm width)
4509 Ty 2 aramid	3 mm cell; 64.1 kg/m ³ aramid honeycomb	Modified phenolic	0.30 mm unidirectional carbon—phenolic	9.9	20.0	18,932.0	7.4	26,148.8
5424 Ty I	3 mm cell; 97.7 kg/m ³ , 5052 alloy, aluminium honeycomb	Modified epoxy	0.38 mm unidirectional FRP—epoxy	10.2	26.0	12,345.1	18.8	44,320.2
5424 Ty II	3 mm cell; 136.2 kg/m ³ 5052 alloy, aluminium honeycomb	Modified epoxy	0.38 mm unidirectional FRP—epoxy	10.2	30.3	13,697.4	17.2	46,536.1
5433C	Fire-retardant epoxy woven FRP	Epoxy	Top face: 0.41 mm aluminium alloy 2024T3 Bottom face: 0.51 mm aluminium alloy 2024T3	1.5	35.9	NR	NR	44,320.2

FRP, fibreglass-reinforced plastic; *NR*, not required by customer specification.
Adapted from Doorway, M.C.G., 1997. Simplified Sandwich Panel Design. 34 (3), M.C. Gill Corp.

Table 5.7 Basic properties of sandwich panels used by Airbus Industry aircraft

Product	Core type and density	Adhesive	Facing skins	Panel thickness	Panel weight	Distributed surface load (N)			Concentrated load without permanent deformation	Impact strength
				(mm)	(N/m ²)	Airbus Industrie TL 63/5000/79 (Gillfab 4105) and DAA/MBB/A1 5360 M16 000100 (Gillfab 4205, 4322 and 4323)				
						Load at 10.9 mm deflection	Load at 17.0 mm deflection	Ultimate load	(N)	(N m)
4105	4.8 mm cell; 96.1 kg/m ³ aramid honeycomb	Modified epoxy	0.64 mm woven FRP epoxy	9.5	32.4	NA	NA	218,111	>8724.4	NA
4205	4.8 mm cell; 96.1 kg/m ³ aramid honeycomb	Modified epoxy	0.64 mm fibreglass fabric—carbon fibre	9.5	33.8	43,622	76,339	196,300	>8375.5	NA
4322	4.8 mm cell; 96.1 kg/m ³ aramid honeycomb	Modified epoxy	Top face: 0.61 mm fibreglass—phenolic Bottom face: 0.56 mm fibreglass—phenolic	9.5	33.4	NA	>61682	165,764	>8724.4	93.1
4323	4.8 mm cell; 96.1 kg/m ³ aramid honeycomb	Modified epoxy	Top face: 0.76 mm fibreglass—phenolic Bottom face: 0.51 mm fibreglass—phenolic	12.6	36.2	NA	>106176	242,845	>8724.4	239.3

FRP, fibreglass-reinforced plastic; NA, not applicable.

Adapted from Doorway, M.C.G., 1997. Simplified Sandwich Panel Design. 34 (3), M.C. Gill Corp.

Table 5.8 Physical and mechanical properties of sandwich panels used by Boeing and Douglas aircraft

Product	Stabilized compressive strength (MPa)	In-plane shear strength (MPa)	2 lb Gardner impact (N mm)	Insert shear strength (N)	Roller cart (number of cycles to failure)	30-day 97% humidity soak		Specifications
						508 mm flexure strength (N)	Climbing drum peel (N mm)	
	Testing methods							
	Military standard 401B	BMS 4-17D	Model 11K3	Shur-Lok 5107-A3	Military standard and DAC Dwg 7954400	Military standard 401B		
4409 Ty II	13.40	2.47	21,827.7	62,423	120076/36781	13,173.9	19,994.1	Boeing BMS 4-20
4409 Ty III	5.61	2.34	22,160.0	57,669	82,300	12,170.6	21,052.0	Boeing BMS 4-20
4417 Ty I	5.83	2.65	145,148.3	80,221	83,964	10,643.8	43,212.0	Boeing BMS 4-17
4417 Ty II	14.00	3.06	119,664.2	85,368	121020/38427	10,992.8	57,616.1	Boeing BMS 4-17
4417 Ty III	15.42	2.63	183,928.5	84,234	120001/35083	12,999.4	54292.2	Boeing BMS 4-17
4417 Ty IV	5.32	2.81	140,716.3	71,986	83,804	20,066.2	55,400.2	Boeing BMS 4-17

4509 Ty 1	10.88	2.84	28,808.2	57974 ^a	111,002	21,025.9	28,586.4	Douglas DAC Dwg 7954400, ty 1
4509 Ty 2	3.96	2.79	26,592.0	49380 ^a	NR	17,885.1	24,597.6	Douglas DAC Dwg 7954400, ty 1
5424 Ty I	7.49	3.14	122,988.3	85,150	83570 (no failure)	10,731.1	55,400.2	Boeing BMS 4-23
5424 Ty II	12.25	3.11	190,576.5	88,248	121652 (no failure)	11,647.1	592,824.6	Boeing BMS 4-23
5433C	NR	NR	34,348	NR	NR	NR	NR	Boeing BMS 7–326 ty VII, CI 2/1

FRP, fibreglass-reinforced plastic; *NR*, not required by customer specification.

^aCalled insert membrane.

Adapted from Doorway, M.C.G., 1997. Simplified Sandwich Panel Design. 34 (3), M.C. Gill Corp.

Table 5.9 Physical and mechanical properties of sandwich panels used by Airbus Industry aircraft

Product	Compressive fatigue 2×10^6 cycles	In-plane panel shear (N)	Building fatigue lower limit 1483.2 N and upper limit 14700.7 N 2×10^6 cycles	Roller cart strength (test cycles to failure)	Insert pull-out (N)	Flammability	Smoke D, flaming 240 s nonflaming 240 s	Toxic gas emission	Heat release	Oil burner
	Testing methods									
	Airbus Industrie TL 53/5000/79 (Gillfab 4105) and DAA/MBB/A1 5360 M1B 000100 (Gillfab 4205, 4322 and 4323)					DAA/ATS 1000.001			Far 25.853/FAR 26.855	
4105	NA	29,000	NA	5583.6 N–120000 6892.3 N–35000	74,157.7	PASS	NA	NA	NA	NA
4205	PASS	24,100	PASS	5583.6 N–120000 6892.3 N–35000	100,331.0	PASS	34 2	PASS	44.4/44.3	NA
4322	NA	NA	NA	NA	74,157.7	PASS	60 3	PASS	45/45	PASS
4323	NA	NA	NA	NA	73,241.6	PASS	83 7	PASS	43.6/37	PASS

FRP, fibreglass-reinforced plastic; NA, not applicable.

Adapted from Doorway, M.C.G., 1997. Simplified Sandwich Panel Design. 34 (3), M.C. Gill Corp.

Table 5.10 Typical sandwich composite applications by industry

Industry				
Aircraft	Aerospace	Electronics	Transportation	Construction
Floor panels Interior walls Food-handling galley assemblies Wing control surfaces Passenger storage racks Thrust deflector assemblies	Capsule panels Ablative shields for nose cones Instrumentation enclosures and shelves Bulkhead panels Space satellites	Electronic radome construction Large antenna or disk reflectors Military electronic instrumentation shelters Shipboard electronic deck shelters	Cargo pallets Shipping containers Refrigeration panels Rapid transit floor panels Special automobile bodies	Architectural curtain walls Floor panels Partitions and divider panels Expandable hospital shelters

Figure 5.11 Sandwich composites applied in satellites.

Figure 5.12 Aerospace and aeronautical applications: (a) aerospace; (b) aeronautical.

(6) carbon—epoxy filament winding in the yoke; and (7) carbon prepreg and high-temperature-resistant glass fabric in the heat shield.

Fig. 5.12(b) shows the following for the aeronautical industry: (1) epoxy or BMI prepregs and woven preforms (socks) applied in the radar transparent radomes; (2) carbon—epoxy prepregs in the foreplane canard wings; (3) carbon—epoxy prepregs, nonmetallic honeycomb cores and adhesives in the fuselage panel sections; (4) carbon

Figure 5.13 Application of sandwich composites in naval construction.

and glass—epoxy prepregs applied in the leading-edge devices; (5) fin fairings; (6) wings and ribs; (7) epoxy—quartz prepregs in fin tip; (8) rudder; (9) fin; and (10) carbon and glass—epoxy prepregs, honeycomb core and adhesives applied in the flying control surfaces.

Finally, some applications where sandwich composites are typically applied in naval construction are shown in [Fig. 5.13](#), such as: (1) decking; (2, 6) hull skin structures; (3) lightweight floor structures; (4) suspended ceilings; (5) interior furnishings; (7) accommodation cabin units; (8) bridge deck consoles; (9) weather shields; (10) communications equipment; (11) companionway stairs; (12) partitions; (13) lightweight superstructures; (14) products for fire, smoke, and toxicity (FST) and sound attenuation in the engine room; and (15) drive shafts and couplings.

5.3 Design of sandwich structures

The suitable design of sandwich structures is one of the most important steps to enhance and ensure their best performance in service. The major analytical equations and methods that should be used to design composite sandwich structures and ensure they withstand the most relevant predictable service loadings will be addressed, presented and discussed in this chapter. As in many other cases, the design of composite sandwiches with complex geometry or submitted to combined and/or much more complex loading situations may require the application of computer-aided engineering software programmes (eg, Abaqus[®] or Ansys[®]). These software programmes allow predicting and simulating the mechanical behaviour of the structures in conditions of service by using advanced numerical methods, such as finite element analysis (FEA) or finite difference method (FDM).

This section introduces briefly some of the more relevant and simple loading situations to be considered in the mechanical design.

5.3.1 Modes of failure under flexure

Sandwich composites are primarily used as structural beams submitted to bending and must ensure they get necessary stiffness to avoid excessive deformations in load conditions. Thus, first of all, designers must verify that maximum admissible deformations (deflections) are never reached in the sandwich structure in the worst loading service conditions (Fig. 5.14). Typically, such a situation depends only on the load conditions and the sandwich panel bending and shear stiffness (HexWeb; Doorway, 1997).

Another basic design confirmation concerns the verification that both composite facing-skin laminates (see the first case in Table 5.11) and sandwich core are able to withstand the tensile, compressive and shear stresses induced in the worst service loading conditions. In the same conditions, the skin-to-core adhesive also must be capable of transferring the developed shear stresses between skin and core.

Table 5.11 summarizes the major mechanical failure modes to be verified by the designer.

In the design sandwich composite structures, two different cases also must be considered: sandwich beams or sandwich plates. In the case of sandwich beams, the width–length ratio must be lower than one-third ($b/L < 1/3$; see Fig. 5.15), and, in contrast, if this ratio is greater than one-third ($b/L \geq 1/3$), the structure must be considered a plate and the more complex and involved plate theory must be used in the calculations.

5.3.1.1 Sandwich composite beams

Consider a sandwich composite beam of span L and width b , submitted to three-point bending with a central load W per unit width in the conditions defined in Fig. 5.15.

Assuming also the same thickness, t , for both composite facing skins and a thickness, c , for the core, that the beam bends in a cylindrical manner with no curvature in the yz plane, the cross-sections remain plane and perpendicular to the longitudinal axis

Figure 5.14 Excessive deflection in bending (HexWeb).

Table 5.11 Major failure modes of sandwich structures

<p>Facing failure</p>	<p>Insufficient panel thickness, facing thickness or facing strength may result in failure occurring in either the compression or tensile face.</p>
<p>Transverse shear failure</p>	<p>Transverse shear failure is caused when either the shear strength of the core or panel thickness is insufficient.</p>
<p>Local crushing of core</p>	<p>Local crushing of core is caused when compressive strength of the core material is too low.</p>
<p>Panel buckling</p>	<p>Panel (or general) buckling is caused when either the panel thickness or core shear stiffness is too low.</p>
<p>Shear crimping</p>	<p>Shear crimping, which can occur as a consequence of general buckling, is caused when either the shear modulus of the core material or the shear strength of the adhesive is low.</p>
<p>Face wrinkling</p>	<p>Core compression failure (buckling inwards) or adhesive bond failure (buckling outwards) may occur depending on the relative strength of the core in compression and adhesive in flatwise tension.</p>
<p>Intracell buckling (dimpling)</p>	<p>Intracell buckling (applicable to cellular cores only) occurs where the skins or faces are very thin and the cell size is large. This effect may cause failure by propagating across adjacent cells, thereby inducing face wrinkling.</p>

Figure 5.15 Sandwich composite beam submitted to three-point bending: (a) simple supported sandwich beam; (b) sandwich beam cross-section.

and sandwich skins remain firmly bonded to core the flexural stiffness of the sandwich beam, D , may be calculated as:

$$D = \frac{E_s \cdot b \cdot t^3}{6} + \frac{E_s \cdot b \cdot t \cdot d^2}{2} + \frac{E_c \cdot b \cdot c^3}{12} \quad [5.1]$$

where (see also Fig. 5.15):

E_s and E_c are the in-plane Young's moduli of the laminate facing skin and core in the x direction, respectively;

d is the distance between the midplanes of the upper and bottom skins; and

c is the core thickness.

In almost of cases as $\frac{d}{t} > 5.77$ and $\frac{E_s}{E_c} \cdot \frac{t}{c} \cdot \left(\frac{d}{c}\right)^2 > 16.7$, the first and third terms of Eq. [5.1] may be negligible because they amount to less than 1% of the flexural stiffness of the beam, which means it could be calculated by using the following simplified equation:

$$D = \frac{E_s \cdot b \cdot t \cdot d^2}{2} \quad [5.2]$$

The shear stiffness of the sandwich beam, S , is determined by:

$$S = b \cdot d \cdot G_c \quad [5.3]$$

where G_c is the core shear modulus that must be considered equal, as appropriate, to the core shear modulus in the ribbon, G_L , or transverse, G_W , direction:

$$G_c = G_L \quad \text{or} \quad G_c = G_W \quad [5.4]$$

Table 5.12 shows the most common bending conditions that sandwich composite beams may withstand.

Table 5.12 Values to be used on beams loaded in different bending conditions (HexWeb)

Beam type	Maximum shear force F	Maximum bending moment M	Bending deflection coefficient k_b	Shear deflection coefficient k_s
$P = q l b$ Simple support Uniform load distribution	$P/2$	$Pl/8$	$5/384$	$1/8$
$P = q l b$ Both ends fixed Uniform load distribution	$P/2$	$Pl/2$	$1/384$	$1/8$
Simple support Central load	$P/2$	$Pl/4$	$1/48$	$1/4$
Both ends fixed Central load	$P/2$	$Pl/8$	$1/192$	$1/4$
$P = q l b$ One end fixed (cantilever) Uniform load distribution	P	$Pl/2$	$1/8$	$1/2$
One end fixed (cantilever) Load one end	P	Pl	$1/3$	1
$P = (q l b)/2$ One end fixed (cantilever) Triangular load distribution	P	$Pl/3$	$1/15$	$1/3$

q , the uniformly distributed load.

The maximum deflection in the sandwiched composite beam (Fig. 5.14) may be calculated as:

$$\delta = \frac{k_b \cdot P \cdot L^3}{D} + \frac{k_s \cdot P \cdot L}{S} \quad [5.5]$$

where:

k_b and k_s are the deflection coefficients given in Table 5.12 for each bending case;
 L is the total sandwich beam length; and
 D and S are the sandwich beam bending and shear stiffness, respectively.

The values of the deflection calculated from Eq. [5.5] must be compared with the maximum allowable deflection required in the beam design.

5.3.1.2 Sandwich plates

As was stated in Section 5.3.1, when the ratio between the width, b , and length, a , is equal to or exceeds one-third (see Fig. 5.16), the beam theory must be replaced by the much more complex and involved plate theory.

The present text only presents solutions for the case, shown in Fig. 5.16, of a sandwich composite plate submitted to a uniformly distributed load and supported on all its four sides. In these conditions, some new coefficients and charts will be introduced and provided in the design calculations, such as:

$$\lambda = 1 - \mu^2 \quad [5.6]$$

where:

μ is the Poisson's ratio of the facing skin material (in the previous calculations of the beam submitted to the end loading condition, it was assumed that $\lambda = 1$).

Figure 5.16 Sandwich composite plate simply supported on all four sides.

$$R = \frac{G_L}{G_W} \quad [5.7]$$

where:

G_L and G_W are the core shear moduli in the ribbon and transverse directions, respectively; and

$$V = \frac{\pi^2 \cdot E_S \cdot t \cdot d}{2 \cdot b^2 \cdot G_W \cdot \lambda} \quad [5.8]$$

where (see also Fig. 5.15):

E_S is the laminate facing skin's Young's modulus;

t is the facing skin's thickness;

d is the distance between the midplanes of the upper and bottom skins;

b is the plate width;

G_W is the core shear modulus in the transverse direction; and

λ is the parameter calculated by Eq. [5.13].

After determining the value of coefficient K_1 from the chart depicted in Fig. 5.17, the total deflection, δ , of the plate submitted to the uniformly distributed load, q , may be calculated by:

$$\delta = \frac{2 \cdot K_1 \cdot q \cdot b^4 \cdot \lambda}{E_S \cdot t \cdot d^2} \quad [5.9]$$

5.3.2 Skin failure

5.3.2.1 Sandwich beams

The maxima tensile and compressive stresses, σ_S , in the facing-skin laminates of the beam may be determined by Eq. [5.10]:

$$\sigma_S = \pm \frac{M}{d \cdot t \cdot b} \quad [5.10]$$

where:

M is the maximum bending moment the sandwich beam has to withstand. Table 5.12 presents the maximum bending moment to be considered in the most common flexion cases.

The stress calculated from Eq. [5.10] must be compared with the maximum allowable stress that the laminate facing skin can support.

End loading conditions (see Fig. 5.18) may also cause skin failure due to excessive stress, sandwich panel buckling failure (the fourth line of Table 5.11), intracell dimpling or face wrinkling (the last two cases given in Table 5.11).

Figure 5.17 Charts for determining the coefficient K_1 .

Figure 5.18 Sandwich composite beam submitted to end loading conditions.

By considering that the end applied load, P , is supported by both skins with the same thickness, t , the stress developed in the facing skins may be calculated as:

$$\sigma_s = \frac{P}{2 \cdot t \cdot b} \quad [5.11]$$

where b is the panel width.

If the end load is applied with uniform distribution along length, q , the total load to be considered in Eq. [5.11], P , is calculated as:

$$P = q \times b \quad [5.12]$$

As happens in bending, the stress calculated by Eq. [5.11] must be lower than the maximum allowable one that the laminate facing skins support to ensure they do not suffer failure.

It is also necessary to ensure that the compressive end load, P , does not overpass the sandwich composite beam buckling load, P_b (Table 5.11, fourth line), which may be calculated as:

$$P_b = \frac{\pi^2 \cdot D}{l^2 + \frac{\pi^2 \cdot D}{G_c \cdot d \cdot b}} \quad [5.13]$$

and

$$|P| \leq |P_b| \quad [5.14]$$

where:

D is the panel flexural stiffness calculated from Eqs [5.1] or [5.2];

G_c is the lower value of the core shear modulus see Eq. [5.4]

l is the beam length; and

d is the distance between the midplanes of the upper and bottom skins (see Fig. 5.15).

Furthermore, to avoid buckling of the facing skins where they are unsupported by the honeycomb walls, the following intracell buckling (dimpling) stress, σ_{CR}^d , must not be exceeded in the skins:

$$\sigma_{CR}^d = 2 \times E_s \times \left[\frac{t}{s} \right]^2 \quad [5.15]$$

where (see also Fig. 5.15):

E_s is the laminate facing skin's Young's modulus;

t is the facing skin's thickness; and

s is the size of the core cell.

Finally, wrinkling and buckling may occur in facing skins with a wavelength greater than the core cell width at higher stresses than the critical one, σ_{CR}^W , which is given by Eq. [5.16]:

$$\sigma_{CR}^W = 0.5 \times [G_c \cdot E_c \cdot E_s]^{1/3} \quad [5.16]$$

where (see also Fig. 5.15) E_c and G_c are the core Young's and shear moduli, respectively.

5.3.2.2 Sandwich plates

In the case of sandwich composite plates, after determining the coefficient K_2 from the chart depicted in Fig. 5.19, the maximum facing-skin stress should be calculated as:

$$\sigma_s = \frac{K_2 \cdot q \cdot b^2}{d \cdot t} \quad [5.17]$$

where K_2 is a coefficient determined from the chart depicted in Fig. 5.19.

5.3.3 Core failure

5.3.3.1 Sandwich beams

In sandwich beams under flexure, the core may fail due to excessive shear stress. Thus, the maximum value of such shear stress developed in the core caused by the transverse

Figure 5.19 Chart for determining the coefficient K_2 for sandwich plates.

load on the beam (see the second line of [Table 5.11](#)), τ_c , calculated from [Eq. \[5.18\]](#), must be lower than the allowable transverse shear stress the core withstands:

$$\tau_c = \frac{F}{c \cdot b} \quad [5.18]$$

where:

F is the maximum shear force shown in [Table 5.12](#) for each bending case; and c and b the core thickness and beam width, respectively.

Furthermore, the core must also resist the local compressive stress cause by the application of the transverse load, P (see the case shown in the third line of [Table 5.11](#)). Such compressive stress may be calculated by [Eq. \[5.19\]](#) and must be smaller than the maximum allowable compression stress the core can withstand:

$$\sigma_c = \frac{P}{A} \quad [5.19]$$

where A is the area where the load, P , is applied.

Finally, in the case of end loading conditions (see [Fig. 5.18](#)), shear crimping failure may also occur in the core as a consequence of buckling that may be caused by a low

core shear modulus or weak adhesive shear strength. To avoid shear crimping failure, the total end load, P , applied to the sandwich beam must be smaller than the shear crimping and buckling, P_b^{SC} , determined as:

$$P_b^{SC} = c \times G_c \times b \quad [5.20]$$

where:

c is the core total thickness (see Fig. 5.15); and

G_c is the lower value of the core shear modulus (see Eq. [5.4])

5.3.3.2 Sandwich plates

In the case of a sandwich composite plate submitted to a uniformly distributed load and supported on all four sides, previously presented in Fig. 5.16, the maximum shear stress developed in the core, τ_c , may be calculated from:

$$\tau_c = \frac{K_3 \cdot q \cdot b}{d} \quad [5.21]$$

where:

K_3 is a coefficient to be determined from the chart depicted in Fig. 5.20;

q is the uniformly applied load;

b is the plate width; and

d is the distance between the midplanes of the upper and bottom beam skins.

The core of the plate must also withstand the same local compressive stress determined for sandwich composite beams by Eq. [5.19], which results from the application of the transverse load, P (see the case shown in the third line of Table 5.11).

5.3.4 Skin and core interfacial design

Skin—core bonding must withstand, of course, at least the above-mentioned shear stress developed for the skin and core. Loads that may cause skin—core debonding only can be compared with those associated to other modes of failure (eg, face yielding, face wrinkling and core shearing) if relatively large cracks preexist at the interface. However, debonding may occur as a result of poor manufacturing, localized impact, fatigue or exposure to extremely high temperature.

It is common to consider that skin—core debonding is avoided by ensuring that the preexistent cracks at the skin—core interface do not reach a critical length related to the adhesive critical strain energy release rates in modes I and II (G_{Ic} and G_{IIc}), normally determined by using double-cantilever beam (DCB), single-cantilever beam (SCB) and end-notched flexural (ENF) test methods, respectively. Mixed-mode fracture testing methods, which consist of a modified DCB testing setup, may allow determination of the interface fracture behaviour under simultaneous mode I and II conditions.

Figure 5.20 Chart for determining the coefficient K_3 .

In the case of the sandwich beam under flexural concentrated load, w , depicted in Fig. 5.15, by considering that the critical strain energy release rate in mode II, G_{IIc} , was already determined as the most relevant fracture mode in the considered situation, the maximum allowable critical load that the beam may sustain, w_c , may be determined as:

$$w_c = \sqrt{X \cdot L \cdot b \cdot G_{IIc}} \quad [5.22]$$

where:

X is a ratio between the applied load and deflection see Eq. [5.5], $\frac{w}{\delta}$; and G_{IIc} is the critical strain energy release rate in mode II.

Experimental evidence shows that debonding is not likely to occur unless a large crack exists in the interface. Thus, a maximum allowable crack length should be defined for more liable structures in order to be adequately monitored through

nondestructive quality control tests. In complex real structures and load conditions, studies of skin–core debonding usually require the use of advanced numerical methods, such as FEA or FDM.

5.4 Quality control, maintenance, testing, inspection and repairing

This section will discuss the major aspects related to quality control during sandwiched composites manufacturing and their maintenance, testing and repairing in service. Visual, ultrasonic, X-ray, back-light and moisture detectors are some of the techniques reviewed as the main methods commonly used to detect damages in sandwich structures that are applied in aerospace and aircraft parts ([Aviation Maintenance Technician Handbook, 2012](#)).

Major manufacturing damage and defects usually include delaminations, resin-starved and resin-rich areas, cracks, blisters and air bubbles, wrinkles, voids and thermal decomposition. This section summarizes the most common nondestructive testing techniques (NDTs) used to detect such defects in quality control inspections made during manufacturing and in-service conditions of composite sandwich structures.

5.4.1 Nondestructing testing

Sandwich structures can be subjected to different impact loads that possibly create delamination and debonding damage that needs to be assessed by reliable, efficient, easy-to-use NDT methods. The damage will vary depending on the nature of the composite part, its composition and its density. The most commonly used nondestructive methods to detect damages in sandwich structures applied in aerospace are as follows ([Aviation Maintenance Technician Handbook, 2012](#)):

- Visual inspection: This is the primary method for the in-service inspection of sandwich structures. It is fast and has a large field of view, being suitable for detecting impact damages larger than 0.5 mm. It cannot detect delaminations or debondings, and it is not suited for defect sizing and defect depth estimation.
- Audible sonic testing (coin tapping) and automated tap test: This is the most common technique for detecting delamination or debondings. It is performed by tapping with a round disk or a light-weight hammer, and listening to the sound response of the sandwich structure to the hammer. A clear, sharp, ringing sound is indicative of a well-bonded solid structure, whereas a dull or thud-like sound indicates a discrepant area. This inspection must be accomplished by experienced personnel. The automated tap test is similar to the manual tap test, but instead of a hammer a solenoid is used. A transducer and recording system allows saving the force-time signal of the impactor. As the duration of the impact force can be correlated with the stiffness of the structure, a variation of such duration allows one to predict delamination and debonding damages.
- Ultrasonic inspection: This technique uses sound waves with frequencies above the audible range. Ultrasonic waves are introduced into a material where they travel in a straight line and at a constant speed until they encounter an interface (eg, a delamination). At surface interfaces, some of the wave energy is reflected and some is transmitted. The amount of reflected or transmitted energy can be detected and provides information about the damage area. Pulse-echo and through transmission (relates to whether reflected or transmitted energy is

used), normal beam and angle beam (relates to the angle that the sound energy enters the test sandwich) and contact and immersion (relates to the method of coupling the transducer to the test sandwich) are the most frequently used ultrasonic techniques. Information from ultrasonic testing can be presented in three formats: A-scan (presentation displays the amount of received ultrasonic energy as a function of time) B-scan (presentation displays a cross-section of a test specimen) and C-scan (presentation displays a plain-type view of the test specimen and discontinuities). Using this technique, porosities, inclusions, delaminations and debondings can be detected.

- Radiography: Radiography involves passing X-ray radiation through a composite part and capturing an image on the opposite side through the phenomenon of differential absorption. Areas with varying degrees of density and absorption exhibit different behaviour in terms of absorbed radiation. It can be used to identify voids, water, impact delaminations, crack-damaged cores, insufficient adhesive and entrapment of foreign material during the manufacturing process.
- Thermography: The principle of thermography is to send a heat flow on the surface of an inspected sandwich part during a certain time and then to capture the resulting thermal response. All techniques make use of the different behaviour in terms of thermal conductivity between normal and defected areas. Usually, a heat source is used to elevate the temperature of the composite under examination while observing the heating effects caused by the decrease in thermal conductivity that damaged areas exhibit. Defects such as debondings, delaminations, cracks and the presence of water can be identified.
- Moisture detector: Uses radiofrequency power loss to detect water in sandwich structures.

5.4.2 *Repairing sandwich structures*

Sandwich structures are bonded constructions with thin-face skins that usually can be repaired by bonding. The repair of such kinds of structures uses similar techniques for the most common types of face sheet materials, such as glass or carbon fibre composite shells. The repairs must meet the strength requirements and can be classified as temporary (limited by time; must be removed and replaced), interim (not restoring the required durability to the component and, therefore, having a different inspection interval and/or method) and permanent repairs (restoring both strength and durability). Sandwich structures with minor core damage can be repaired using a potted repair. The core material could be left in place or could be removed and filled up with a potting compound (usually filled epoxy resins) to restore some strength. This technique is also used sometimes for cosmetic repair of skin panels. Structural repair normally consists of using the following steps on the damaged component ([Aviation Maintenance Technician Handbook, 2012](#)):

- Inspect the damaged area: Use an NDT method to assess the damaged area.
- Remove the water from the damaged area: This task can be performed using breather cloths and the vacuum bag method.
- Remove the damage: The damaged area must be trimmed out, usually in circular or oval shapes. If the damaged area includes the core, this material must also be removed.
- Prepare the damaged area: The damaged and surrounding areas should be cleaned by using sandpaper, compressed air and approved solvents.
- Installation of the honeycomb core (wet lay-up): The damaged core must be replaced by a plug of the same type and shape, using an adhesive layer or resin to bond it. The vacuum bag technique can be used to cure the core replacement.

Figure 5.21 Repair ply installation (*Aviation Maintenance Technician Handbook*, 2012).

Figure 5.22 Vacuum-processing technology (*Aviation Maintenance Technician Handbook*, 2012).

- Prepare and install the repair plies: Impregnate or remove the backing material of the repair plies that should have the correct size and ply orientation (*Fig. 5.21*).
- Vacuum-bag the repair: The vacuum bag technology is usually used to remove air and to pressurize the repair for curing (*Fig. 5.22*).

- Cure and inspect the repair: The repair should be cured using the appropriate curing cycle. The repair should be free from pits, blisters and resin-rich and resin-starved areas. Usually, a top finishing coating is applied. Finally, an NDT technique is used to inspect the repaired area and confirm if the repair is free of defects.

5.5 Conclusions

Sandwiched composites are of particular interest and widely used because they are very suitable and amenable to the development of lightweight structures with high in-plane and flexural stiffness. In recent years, their application has increased greatly in the automotive and marine industries, and even in primary structures of commercial aircrafts (fuselage). Thus, large growth in the structural applications of those materials in the aerospace industry is forecasted for the coming years.

The main raw materials, manufacturing technologies and design methods to be considered for applying sandwich composites in aerospace structures were briefly presented and discussed in this chapter.

The chapter may be considered as an introduction that does not disregard further fundamental readings on this matter and more specific knowledge on each particular application case.

References

- Advanced Composite Manufacturing Centre. <<http://www.tech.plym.ac.uk> <http://www.tech.plym.ac.uk/sme/acmc/cpd/rtm.htm>> (visited 03.06.15.).
- Aluminium Honeycombs, Alcore Inc. – A Gill Company, <http://www.thegillcorp.com> (visit on 28.04.28.).
- Aviation Maintenance Technician Handbook—Airframe. Chapter 7: Advanced Composite Materials, vol. 1, 2012. Federal Aviation Administration US Department of Transportation – Flight Standards Service.
- Balsa Wood Core, Airex AG Industrie, <http://www.3accorematerials.com> (visit on 28.04.15.).
- Beckwith, S.W., July/August 2008. Sandwich core materials & technologies – Part I. SAMPE Journal 44 (4).
- Bitzer, T., 1997. Honeycomb Technology – Materials, Design, Manufacturing Applications and Testing. Chapman & Hall.
- Broughton, W.R., Crocker, L.E., Gower, M.R.L., January 2002. Design Requirements for Bonded and Bolted Composite Structures. NPL Report MATC(A) 65. NPL Materials Centre – National Physical Laboratory, Middlesex, UK.
- Doorway, M.C.G., 1997. Simplified Sandwich Panel Design, vol. 34 (3). M.C. Gill Corp. Hexply-prepreg Technology, Hexcel.
- HexWeb™ Honeycomb Sandwich Design Technology. Hexcel® Technology Manuals. <<http://www.hexcel.com/resources/technology-manuals>> (visit on 19.03.15.).
- <<http://www.nuplex.com/composites/processes/filament-winding>> (visited on 03.06.15.).
- Marasco, A.I., 2005. Analysis and Evaluation of Mechanical Performance of Reinforced Sandwich Structures: X-cor™ and K-cor™. PhD thesis. Cranfield University.
- Mazumdar, S.K., 2001. Composites Manufacturing: Materials, Product and Process Engineering. CRC Press.

- Muzzy, J.D., 2000. Thermoplastics — properties. In: Kelly, A., Zweben, C. (Eds.), *Comprehensive Composite Materials*, vol. 2. Elsevier Ltd, pp. 57–76 (Chapter 2).01.
- Noor, A.K. (Ed.), 2000. *Structures Technology for Future Aerospace Systems*. Progress in Astronautics and Aeronautics, vol. 188. American Institute of Aeronautics and Astronautics.
- Peters, S.T. (Ed.), 1998. *Handbook of Composites*, second ed. Chapman & Hall.
- Sezgin, F.E., 2011. *Mechanical Behavior and Modeling of Honeycomb Cored Laminated Fiber/Polymer Sandwich Structures*. Master Thesis. İzmir Institute of Technology, Turkey.
- van Paepegem, W. Home Made Composites. <http://www.composites.ugent.be/home_made_composites/organizing_your_composite_workshop.html> (visited on 27.05.15.).
- Zenkert, D., 1995. *An Introduction to Sandwich Construction*. EMAS Publishing.