

Modelling of a Composite Propeller Blade with Complex Geometry and Layered Structure

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Reliable design of composite propeller blades requires integration of geometric modelling, laminate definition, structural assessment, and manufacturing preparation. Conventional CAD procedures employ equivalent solid representations or simplified laminate descriptions, which cannot capture ply continuity, local fibre orientation, or thickness variations over twisted aerodynamic surfaces. This paper presents a manufacturing oriented methodology for three-dimensional modelling and structural analysis of a composite propeller blade with complex geometry and an explicitly defined laminate architecture. Individual plies are modelled while preserving the aerodynamic surface and accounting for stacking sequence, ply thickness along the blade span. The CAD model provides a geometric basis for finite element analysis under prescribed loading conditions, mass property evaluation, centre of mass determination, and generation of ply geometries for manufacturing. Application to a blade composed of fibre reinforced plies and a lightweight core demonstrates that geometric and material fidelity can be maintained across design, analysis, and manufacturing preparation without additional geometric reconstruction.

Keywords: Propeller blade, composite materials, geometric modelling, layered structure

1. INTRODUCTION

The increasing complexity of modern engineering structures requires the integration of advanced computational tools and modelling methodologies capable of representing both geometric features and material behaviour. This requirement is particularly relevant to composite materials, which are widely used in the aerospace and marine industries because of their high specific strength and stiffness, as well as the possibility of tailoring their mechanical properties through the selection of suitable constituent materials and laminate stacking sequences [1,2]. The structural response of composite laminates is strongly influenced by fibre orientation, ply thickness, stacking sequence, and material distribution. Consequently, a detailed definition of the laminate architecture is essential for reliable structural assessment and safe application of composite components [3–6].

Propeller blades are geometrically demanding engineering components characterized by varying cross-sections, continuously changing curvature, and twist along the blade span. Their design must simultaneously satisfy aerodynamic, structural, mass, and manufacturing requirements. The use of composite materials introduces further challenges because the laminate architecture must be adapted to a curved and twisted

aerodynamic surface while preserving the prescribed stacking sequence, fibre orientation, and continuity of individual plies. Previous studies have shown that laminate configuration and geometric variations significantly affect the stiffness, strength, deformation, and dynamic response of composite propeller blades [6–8]. Structural optimization and fluid–structure interaction analyses have also been applied to evaluate blade behaviour under operational loading conditions and support the development of efficient composite blade structures [9,10].

In many existing design procedures, aerodynamic geometry, laminate definition, structural analysis, and manufacturing preparation are treated as separate stages. The transfer of data between independently developed models may require additional geometric reconstruction and manual adaptation of the laminate definition. Such procedures can introduce inconsistencies in ply geometry, fibre orientation, thickness distribution, and material representation. Recent studies have therefore emphasized the need for integrated design methodologies that establish a consistent connection between geometric modelling, numerical analysis, and manufacturing preparation [11–14]. Within such methodologies, the finite element method provides an effective means of assessing the structural response of anisotropic and heterogeneous composite components while accounting for the actual laminate configuration [15,16]. Despite the progress achieved in composite modelling, the development of a manufacturing oriented three-dimensional model of a propeller blade remains challenging. In particular, the representation of individual plies over a twisted aerodynamic surface must be sufficiently detailed to support both structural analysis and the

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generation of manufacturing data. Simplified equivalent material models can be suitable for preliminary structural calculations, but they do not provide the individual ply geometries required for material cutting and layup preparation. A unified model that preserves the relationship between the aerodynamic surface, laminate architecture, structural representation, and manufacturing geometry is therefore required.

This paper presents a methodology for the design, three-dimensional modelling, structural analysis, and manufacturing preparation of a composite propeller blade with complex geometry and an explicitly defined laminate architecture. The main objective is to establish a consistent procedure for adapting individual composite plies to the aerodynamic blade geometry while preserving their thickness, stacking sequence, continuity, and fibre orientation. The methodology connects the initial blade geometry with the laminate definition, finite element model, mass property evaluation, and generation of manufacturing data. In contrast to procedures in which these activities are performed independently, the proposed approach uses a common geometric basis throughout the development process. The methodology is demonstrated through the development and manufacture of a composite propeller blade comprising fibre reinforced plies and a lightweight core. The resulting CAD model provides the individual ply geometries required for cutting and layup operations and enables the determination of laminate mass distribution and centre of mass location. The same laminate configuration is used in the finite element model to evaluate the static structural response under the prescribed loading conditions. The scope of the present research is limited to static structural analysis, manufacturing preparation, and demonstration of the proposed design to manufacturing procedure.

2. MOTIVATION

The structural response of a composite propeller blade is governed by the combined influence of its aerodynamic geometry and internal laminate architecture. Fibre orientation, ply thickness, stacking sequence, material distribution, and the mechanical properties of the constituent materials determine the stiffness, strength, and failure behaviour of the resulting structure [1–5]. In a propeller blade, the relationship between the local material axes and the applied loading directions changes continuously along the span because of variations in cross-sectional geometry, surface curvature, and twist. Consequently, a laminate definition that does not adequately follow the blade geometry may introduce inconsistencies in the predicted structural response. Several studies have investigated the influence of laminate configuration and blade geometry on the behaviour of composite marine propellers. Lin et al. [6] evaluated the strength of a composite propeller blade using nonlinear hydroelastic analysis and examined the response of balanced and unbalanced stacking sequences. Ashok et al. [7] investigated the effect of stacking sequence on propeller performance, whereas Kumar and Rajagopalan [8] considered the combined influence of geometric variations and laminate configuration.

Hussain et al. [9] employed coupled fluid–structure interaction analysis and stacking sequence optimization to assess the strength and deformation of thin composite propeller blades. Although marine and aeronautical propellers operate under different loading environments, these studies demonstrate the importance of geometric–material coupling and provide relevant evidence of the sensitivity of blade response to laminate architecture.

Related investigations in aeronautical applications have considered structural vibration, fluid–structure interaction, bend–twist coupling, and torsional deformation. Oliveira et al. [10] incorporated structural vibration constraints into an aircraft propeller optimization procedure. Park [17] combined aerodynamic design, structural analysis, and fluid–structure interaction to develop a composite turboprop blade with a skin spar core configuration. Rokvam et al. [18] investigated the elastic response and bend–twist deformation of a composite propeller blade, while Möhren et al. [19] examined the structural mechanisms contributing to torsional deformation in different propeller configurations. These studies indicate that the structural behaviour of a propeller blade cannot be evaluated independently of its geometric characteristics and operational loading conditions. The finite element method remains the principal numerical tool for evaluating composite blade structures. Filippi et al. [15] developed an advanced finite element representation for the static and dynamic analysis of a double-swept composite rotor blade. Garinis et al. [16] applied modal analysis to a modified composite helicopter blade comprising a composite spar, carbon-fibre face sheets, and a honeycomb core. Detailed modelling of the core material was addressed by Dinulović et al. [20], who developed a meso-scale finite element model of Nomex honeycomb composite plates. Structural optimization based on finite element analysis was also employed by Svorcan et al. [21] to determine suitable laminate configurations for composite wind-turbine blades. Although these studies address different blade configurations, they demonstrate the importance of preserving material anisotropy and laminate definition in numerical models of lightweight composite structures. An increasing number of studies have also considered the relationship between numerical modelling and manufacturing. Kovačević et al. [11] presented a complete propeller development process comprising aerodynamic design, structural analysis, manufacturing, and experimental testing. Radulović and Milić [12] applied numerical analysis and experimental verification during the development of a composite main rotor blade. Torpan and Zaharia [13] examined the manufacture of composite tail rotor blades using 3D printed moulds, demonstrating the influence of tooling and production requirements on the design procedure. Integrated numerical and experimental methodologies have also been applied to other composite aerospace components, including an optimized VTOL UAV engine mount [14] and a composite multicopter arm [22]. Manufacturing oriented modelling has received further attention in studies dealing with non-conventional blade structures and thermoplastic composites. Malim et al. [23] developed and experimentally validated a structural methodology for a large scale 3D printed propeller. Ho-

Nguyen-Tan et al. [24] presented a procedure for the design and manufacture of a thermoplastic composite fan blade that incorporated ply shape generation, fibre draping analysis, and simulation of process induced deformation. This study is particularly relevant because it demonstrates that changes in fibre orientation caused by draping over a curved and twisted surface can influence both the manufactured geometry and the resulting structural response. The reviewed literature demonstrates substantial progress in composite blade analysis, optimization, fluid–structure interaction, experimental assessment, and manufacturing technology. However, the principal objectives of these studies are generally directed toward aerodynamic or hydroelastic performance, structural optimization, experimental validation, or the simulation of individual manufacturing phenomena. Comparatively less attention has been devoted to establishing a unified, manufacturing oriented CAD model in which each composite ply is represented as an individual three-dimensional entity and adapted directly to the curved and twisted aerodynamic surface. In many existing workflows, the aerodynamic geometry, laminate definition, finite element model, and manufacturing data are developed in separate software environments or through successive geometric reconstructions. Such procedures may introduce discrepancies in ply boundaries, local fibre orientation, thickness distribution, and material continuity. These discrepancies are particularly relevant to propeller blades because even moderate changes in ply geometry or orientation can alter the local stiffness and deformation characteristics of the laminate. The principal research problem is therefore not limited to the application of finite element analysis, but concerns the preservation of geometric and material consistency throughout the entire design to manufacturing process. To address this issue, the present study introduces an integrated CAD based methodology for the three-dimensional modelling, structural analysis, and manufacturing preparation of a composite propeller blade. The procedure explicitly represents individual composite plies and preserves their thickness, stacking sequence, spatial extent, and fibre orientation over the blade surface. A common geometric model is used to define the laminate architecture, evaluate mass properties and structural response, and generate the individual ply geometries required for cutting and layup operations. The methodology is demonstrated through the development and manufacture of a composite propeller blade with a fibre reinforced laminate and lightweight core. The scope of the study comprises geometric and laminate modelling, static finite element analysis, manufacturing preparation, and blade manufacture, while experimental validation of the structural response is outside the scope of the present research.

3. BLADE GEOMETRY

The external geometry of the propeller blade was defined according to the prescribed aerodynamic requirements, including the airfoil distribution, chord variation, local design angle, and thickness distribution along the blade span. The selected configuration was not

subjected to aerodynamic optimization within the present study. Instead, it was adopted as a representative geometry for demonstrating the proposed methodology for three-dimensional laminate modelling, structural analysis, and manufacturing preparation. The total blade length is 1000 mm. Three reference airfoil sections were selected at characteristic spanwise locations to represent the principal changes in chord, relative thickness, and local design angle. Additional sections were generated between these reference locations through aerodynamic interpolation. This procedure preserved surface continuity and provided a gradual transition between adjacent profiles without abrupt changes in curvature or section geometry. The resulting blade surface, together with the reference and interpolated cross-sections, is shown in Figure 1d. The spanwise coordinate z was measured from the blade root reference plane. The local design angle $\theta(z)$, defined relative to the propeller rotational plane, varies along the blade span according to Equation (1). Its distribution is presented in Figure 1a. The variation in cross-sectional geometry was additionally characterized using the maximum and minimum principal second moments of area, $I_{\max}(z)$ and $I_{\min}(z)$, calculated about the centroidal principal axes of each section. The fitted functions describing these distributions are given by Equations (2) and (3), while the corresponding results are presented in Figures 1b and 1c.

$$\theta(z) = 1.23 - 2.61z + 2.15z^2 - 0.65z^3 \quad (1)$$

$$I_{\max}(z) = 2118.69 - 66416.68z + 922580.40z^2 - 7542445.69z^3 + 4.07 \times 10^7 z^4 - 1.54 \times 10^8 z^5 + 4.25 \times 10^8 z^6 - 8.68 \times 10^8 z^7 + 1.32 \times 10^9 z^8 - 1.50 \times 10^9 z^9 + 1.24 \times 10^9 z^{10} - 7.41 \times 10^8 z^{11} + 2.97 \times 10^8 z^{12} - 7.19 \times 10^7 z^{13} + 7.96 \times 10^6 z^{14} \quad (2)$$

$$I_{\min}(z) = 963.09 - 32122.32z + 481456.01z^2 - 4279724.69z^3 + 2.52 \times 10^7 z^4 - 1.04 \times 10^8 z^5 + 3.15 \times 10^8 z^6 - 7.04 \times 10^8 z^7 + 1.17 \times 10^9 z^8 - 1.45 \times 10^9 z^9 + 1.32 \times 10^9 z^{10} - 8.57 \times 10^8 z^{11} + 3.74 \times 10^8 z^{12} - 9.88 \times 10^7 z^{13} + 1.19 \times 10^7 z^{14} \quad (3)$$

The adopted geometry contains continuous variations in twist, curvature, chord, and local thickness, making it suitable for evaluating the ability of the proposed modelling procedure to represent individual composite plies over a non planar aerodynamic surface. These geometric variations directly affect ply boundaries, local fibre directions, and the available thickness for the face laminates and lightweight core. Consequently, the geometric model provides the common reference for laminate definition, finite element analysis, mass property evaluation, and manufacturing preparation.

The structural design requirements derived from the prescribed operational loading conditions were used to define the material system and laminate architecture. The selection procedure considered structural stiffness and strength, mass reduction, geometric compatibility, and manufacturing feasibility. Unidirectional carbon fibre and glass fibre prepreps were selected as the face laminate materials, while Rohacell structural foam was used as the lightweight core.

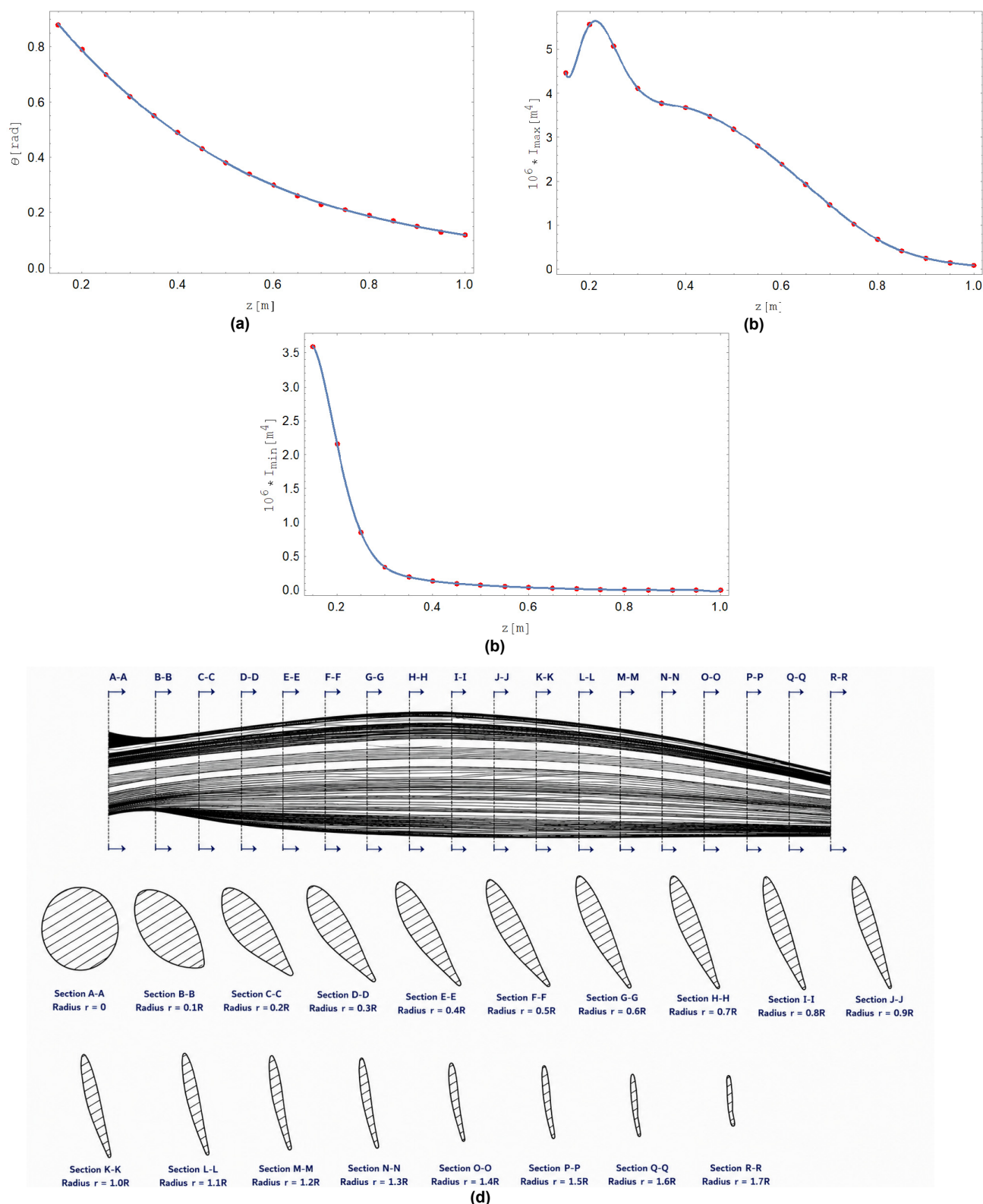


Figure 1. Geometric and design characteristics of the composite propeller blade: (a) spanwise variation in the local design angle $\theta(z)$, described by Equation (1); (b) distribution of the maximum principal second moment of area $I_{\max}(z)$, described by Equation (2); (c) distribution of the minimum principal second moment of area $I_{\min}(z)$, described by Equation (3); and (d) complete blade geometry with the reference and interpolated airfoil sections.

The material data used in the model are summarized in Table 1. The values reported for the prepreps represent areal mass rather than volumetric density. The nominal cured ply thickness was used to define the geometry of each composite layer within the CAD model. The Rohacell core thickness varies along the blade span according to the available internal section geometry.

Table 1. Characterization of composite materials.

Material	Density	Thickness
UD glass-fibre prepreg	210 g/m ²	0.25 mm
UD carbon-fibre prepreg	300 g/m ²	0.30 mm
Rohacell	80kg/m ³	

Each composite ply was represented as an individual geometric entity and assigned its prescribed material, nominal thickness, and fibre orientation. The orientation angle was defined relative to a local reference direction associated with the blade geometry. This local definition is required because the spanwise and chordwise directions vary continuously over the twisted blade surface. It also preserves the relationship between the intended stacking sequence and the material coordinate system subsequently employed in the finite element model. The spanwise distributions of the blade thickness, core thickness, and laminate thickness are presented in Table 2. The table contains sections A'–A' to R'–R', positioned between 150 mm and 1000 mm. These sections should not be interpreted as additional reference airfoils. They represent intermediate geometric stations generated from the three original reference profiles and used to define the local thickness distribution. The laminate thickness decreases toward the blade tip by terminating selected plies at prescribed spanwise locations. Each ply termination was introduced while maintaining geometric continuity and avoiding abrupt variations in the external blade surface.

The elastic constants and strength parameters used in the finite element analysis were obtained from the experimental material characterisation conducted as part of the author's doctoral dissertation [25]. The mechanical properties were determined by testing representative specimens manufactured from the corresponding composite material system. These experimentally determined values were adopted in the numerical model, while the manufacturers' technical data sheets and relevant literature were used as supplementary sources where required. Since all unidirectional plies within each laminate were oriented at 0° relative to the local spanwise direction of the blade, the total laminate thickness is reported for each cross-section.

Table 2. Spanwise distribution of blade, core, and laminate thicknesses.

Cross-section	Specified spanwise position [mm]	Total local blade thickness [mm]	Rohacell core thickness [mm]	Laminate thickness [mm] ¹
Section A'–A'	150	50.50	40.30	5.10
Section B'–B'	200	46.17	35.97	5.10
Section C'–C'	250	41.05	30.85	5.10
Section D'–D'	300	36.26	26.06	5.10
Section E'–E'	350	32.22	22.02	5.10
Section F'–F'	400	28.92	21.72	3.60
Section G'–G'	450	26.17	20.17	3.00
Section H'–H'	500	23.76	20.16	1.80
Section I'–I'	550	21.48	17.88	1.80
Section J'–J'	600	19.20	16.20	1.50
Section K'–K'	650	16.89	13.89	1.50
Section L'–L'	700	14.60	11.60	1.50
Section M'–M'	750	12.39	10.59	0.90
Section N'–N'	800	10.35	8.55	0.90
Section O'–O'	850	8.61	7.41	0.60
Section P'–P'	900	7.21	6.01	0.60
Section Q'–Q'	950	6.17	5.57	0.30
Section R'–R'	1000	5.45	4.85	0.30

¹ For each cross-section, the total laminate thickness per side, comprising multiple composite-fibre plies, is shown. The cross-sections are marked in Figure 1d.

Following the definition of the external blade surface and spanwise thickness distribution, the laminate architecture was implemented in a CAD environment intended for composite structure modelling. The individual plies were adapted to the curved blade surface while preserving their nominal thickness, prescribed boundaries, stacking order, and local fibre orientation. The core geometry was defined from the internal volume remaining after the face laminates and other structural constituents had been introduced.

The resulting model enables visualization of the complete laminate architecture and inspection of the blade structure at any selected cross-section. Because the individual plies are explicitly represented, the same model can be used for structural model preparation, evaluation of material distribution, calculation of blade mass, and determination of the centre of mass location. This common geometric basis reduces the need for separate reconstruction of the blade geometry during the transition from design to numerical analysis and manufacturing.

Flattened ply geometries were generated from the three-dimensional CAD model and used as nominal cutting patterns for manual layup. Their correspondence with the curved blade surface supports accurate material placement and helps control manufacturing deviations such as ply gaps, overlaps, wrinkles, and unintended local thickness variations. The generated patterns also preserve the identification of each ply and its intended position within the stacking sequence.

The complete three-dimensional CAD model is presented in Figure 2, while Figure 3 illustrates the laminate and core arrangement at a representative blade cross-section.

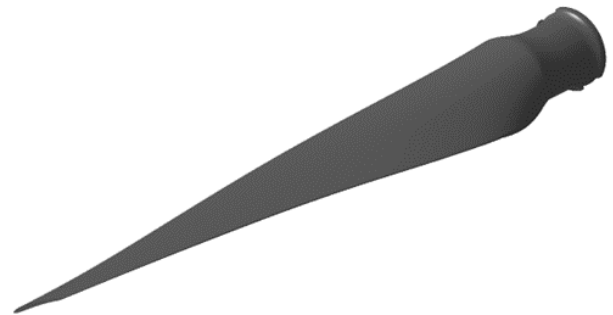


Figure 2. Developed CAD model of the blade.

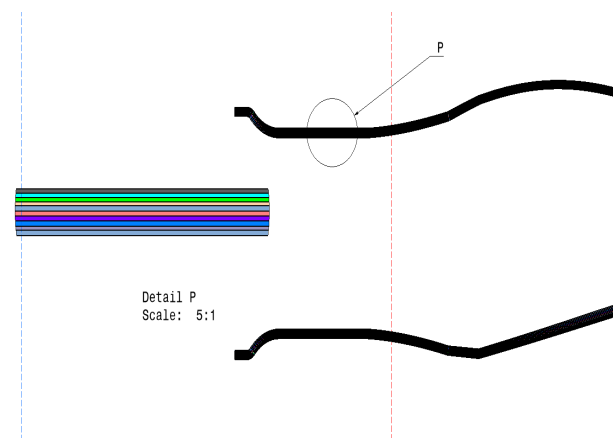


Figure 3. The blade structure in cross-section.

4. METHODOLOGY

The proposed methodology was developed to create a manufacturing-oriented three-dimensional CAD model of a composite propeller blade with a defined laminate architecture. The aerodynamic geometry and internal laminate structure are represented within the same model, which is then used for structural analysis, mass-property calculations, and manufacturing preparation. Unlike simplified models in which the laminate is represented as an equivalent homogeneous solid or shell, the present model includes the individual plies, their nominal thicknesses, stacking sequence, and fibre orientations. The modelling process starts with the aerodynamic definition of the blade. The external surface, airfoil sections, chord distribution, and spanwise twist provide the reference geometry for the laminate model. The materials, ply thicknesses, stacking sequence, and core dimensions are specified according to the structural and manufacturing requirements. In this study, unidirectional carbon-fibre and glass-fibre prepreps were used as the main load carrying materials, while Rohacell foam was used for the light-weight core. The basic material data are given in Table 1, and the variation of the laminate and core thicknesses along the blade span is presented in Table 2. Each ply is modelled as a separate parametric entity based on the corresponding blade surface. Its boundaries, nominal thickness, material, and fibre direction are defined individually. Local coordinate systems are used to control the fibre orientation over the curved and twisted blade geometry. The plies and the foam core are then assembled to form the complete blade model. This makes it possible to inspect the stacking sequence, material distribution, and local laminate thickness at any position along the blade.

The model was developed using the Shape Design and Composite Design workbenches. Since all laminate constituents are included in the CAD definition, the model can be used directly to calculate the blade mass, centre of mass, material volumes, and thickness distribution. It also provides the geometry required for the finite element model, allowing the structural response of the prescribed laminate to be assessed under representative operating loads.

Manufacturing preparation is based on the generation of flattened patterns for the individual plies. These patterns retain the prescribed fibre direction and provide the cutting geometry required for ply preparation and manual layup. Their use facilitates ply positioning and reduces the risk of material gaps, overlaps, and local accumulation during laminate assembly. For plies applied to doubly curved regions, the flattened geometry represents the nominal cutting pattern because material draping and in plane deformation are not simulated in the present model. The methodology assumes that all laminate constituents conform ideally to the aerodynamic surface. Manufacturing effects such as fibre wrinkling, ply bridging, resin rich regions, voids, cure induced thickness variations, and spring back are not considered. The model therefore represents the nominal blade configuration and is used for structural assessment, mass estimation, and manufacturing planning. The complete sequence of modelling and manufacturing preparation activities is shown in Figure 4.

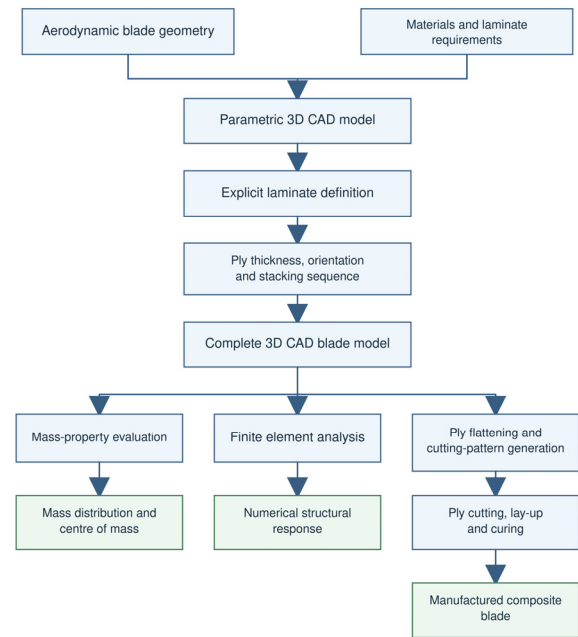


Figure 4. Block diagram of the proposed methodology for the design, CAD modelling, laminate definition, structural analysis, tooling development, and manufacturing preparation of the composite propeller blade.

4.1 Structural Analysis

A static finite element analysis was performed to assess the structural response of the composite propeller blade under centrifugal loading, which represents the dominant load acting on the blade during rotation. The analysis was based on a two-dimensional layered shell formulation that retained the stacking sequence, ply orientations, nominal ply thicknesses, and foam core defined in the CAD model. This allowed the response of the individual plies to be evaluated without replacing the laminate with an equivalent homogeneous material. The finite element model was generated from the developed blade geometry, maintaining consistency between the geometric definition, laminate architecture, and structural model. Orthotropic material properties were assigned to the carbon and glass fibre plies, while the Rohacell core was represented as an isotropic material. The laminate configuration corresponded to the spanwise thickness distribution summarized in Table 2. Layered composite shell elements were used to calculate the global deformation and evaluate the stresses and failure indices in the individual plies. A mesh convergence study was performed using maximum X-direction deformation as the convergence parameter. Element sizes of 2, 3, 4, 6, and 8 mm were considered, as shown in Figure 5. Based on the small variation in the calculated deformation and the balance between numerical accuracy and computational cost, a nominal element size of 4 mm was selected for the final model. The mesh was refined near the blade root and in regions with pronounced geometric transitions. The final finite element mesh is shown in Figure 6.

The blade root was constrained in all translational and rotational degrees of freedom to represent its attachment to the propeller hub. The centrifugal action was represented by an equivalent load with a resultant

force of 54 kN, corresponding to a rotational speed of 2500 rpm. This load case was selected because centrifugal force represents the dominant structural load acting on the blade under the considered operating condition. The calculated deformation increased gradually from the constrained root towards the blade tip.

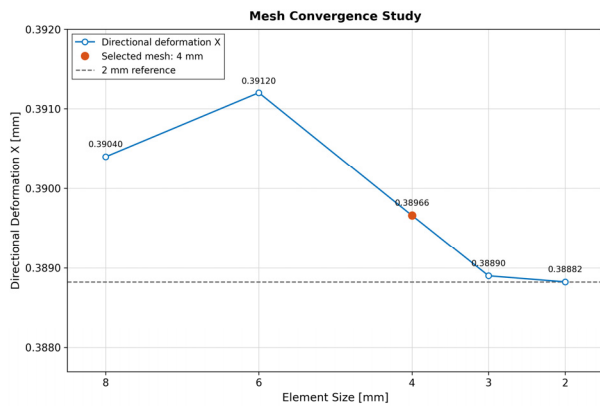


Figure 5. Mesh convergence study.

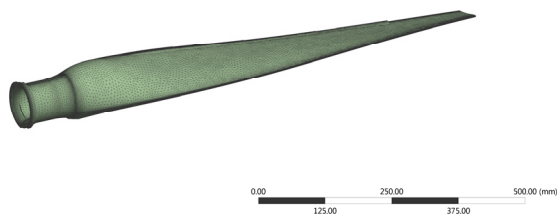


Figure 6. FEM mesh.

Although the absolute maximum total deformation of 0.39 mm was recorded locally at the contact surface, this value was attributed to the idealized representation of the attachment and the imposed boundary condition and was therefore not considered representative of the global blade response. Outside the boundary affected contact region, the deformation increased along the blade span and reached a maximum value of 0.304 mm at the blade tip, as shown in Figure 7. This distribution corresponds to the axial extension of the blade produced by centrifugal loading. The smooth variation of deformation along the span indicates continuous load transfer through the laminate without abrupt changes associated with local stiffness variations.

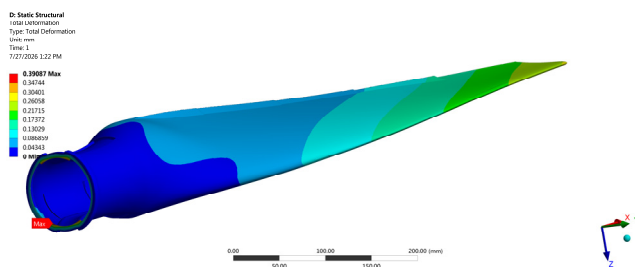


Figure 7. Total deformation of the composite propeller blade under centrifugal loading.

The laminate strength was evaluated using the Tsai–Hill failure criterion, with the results expressed in terms of the inverse reserve factor. A local maximum value of 1.6 was obtained in a small number of surface elements located directly at the fully constrained blade attachment. These isolated values were associated with the stress concentration produced by the idealized fixed

support boundary condition and were therefore not considered representative of the laminate response outside the immediate support region. Outside the elements directly affected by the constraint, the maximum inverse reserve factor was 0.8 and occurred in the blade root region and near the sleeve exit. Since this value remained below 1, no ply failure was predicted outside the constrained region under the centrifugal load corresponding to 2500 rpm. A more detailed assessment of the attachment region would require explicit modelling of the contact between the blade root, sleeve, and propeller hub. The inverse reserve factor distribution is shown in Figure 8.

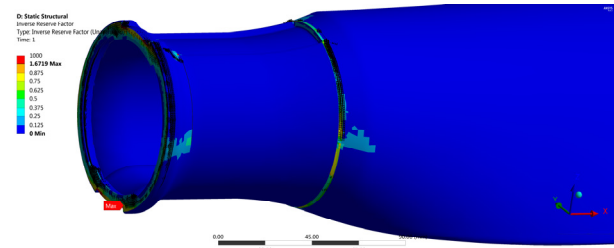


Figure 8. Inverse reserve factor distribution based on the Tsai–Hill failure criterion.

The laminate configuration was defined from the structural and manufacturing requirements and was subsequently assessed under centrifugal loading, laminate optimization was not performed in the present study.

4.2 Manufacturing

After completion of the laminate model, the blade mass was calculated using the material densities assigned to the individual plies and the foam core. The resulting mass was 1985 g. The centre of mass was located at $x=397.18$ mm, $y=0.66$ mm, and $z=97.05$ mm relative to the global coordinate system Oxyz. Its position is marked in green in Figure 9. These values represent the nominal mass properties of the CAD model and depend on the prescribed geometry and material data.

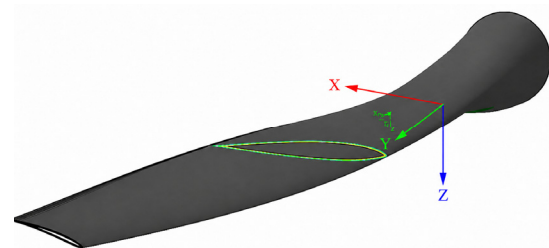
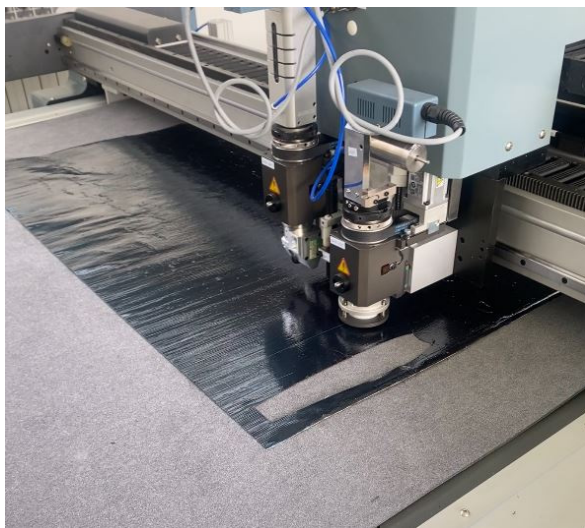


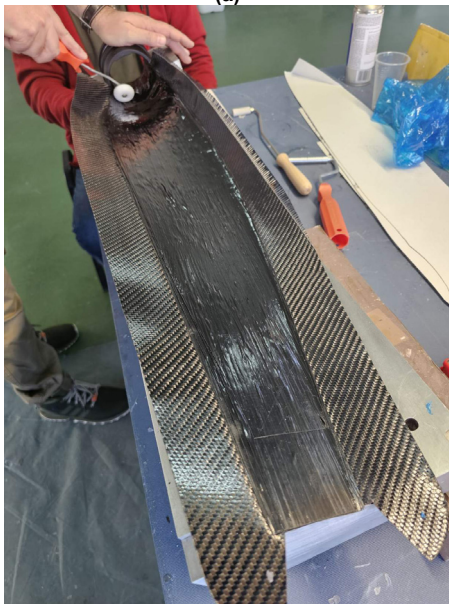
Figure 9. Blade geometry showing composite layers and the center of gravity marked in green ($397.18 \times 0.66 \times 97.05$ mm relative to O_x , O_y , O_z).

A flattened cutting pattern was generated for each composite ply. The patterns corresponding to the upper and lower blade surfaces were prepared and cut separately using an automated cutting machine, as shown in Figure 10a. Their geometry was derived directly from the laminate model, preserving the prescribed ply boundaries and fibre directions. Accurate preparation of the cutting patterns facilitates ply positioning during layup and reduces the occurrence of gaps, overlaps, wrinkles, and local material accumulation. After cutting, the plies were placed in the mould in the prescribed

order. The layup started from the lower blade surface and continued towards the upper surface, following the defined stacking sequence and fibre orientations. The layup procedure is shown in Figure 10b, while Figure 10c presents the closed mould containing the complete laminate before curing.



(a)



(b)



(c)

Figure 10. Manufacturing stages of the composite propeller blade: (a) cutting of the flattened ply patterns; (b) placement of the individual plies in the mould according to the prescribed stacking sequence and fibre orientations; (c) closed mould containing the complete laminate before curing.

The laminate was cured in the closed mould using a prescribed time–temperature–pressure cycle adapted to the prepreg materials. The curing cycle is shown in Figure 11. The controlled temperature increase reduced the resin viscosity and initiated the curing reaction, while the applied pressure promoted laminate consolidation and reduced the amount of trapped air. The dwell period was selected to achieve the required degree of cure. After controlled cooling and depressurization, the mould was opened and the blade was removed for subsequent trimming and finishing.

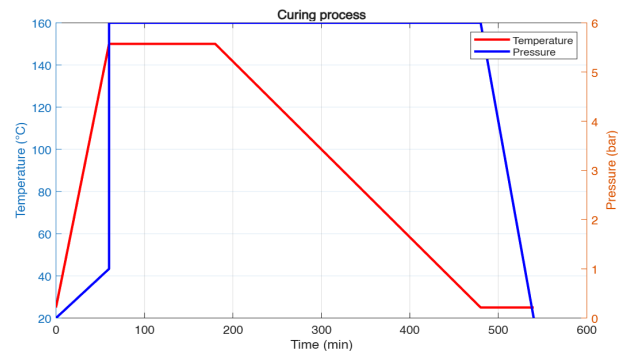


Figure 11. Time, temperature, and pressure gradients used during the composite curing process, ensuring complete polymerization and optimal fibre impregnation.

The manufactured blade is shown in Figure 12. Its external surface followed the intended aerodynamic form, while the upper and lower blade surfaces were formed within the corresponding mould sections. The completed component demonstrates that the laminate definition, flattened ply patterns, and prescribed stacking sequence can be transferred from the CAD model to the manufacturing process. Quantitative dimensional inspection and mechanical testing of the manufactured blade were not included in the present study.



Figure 12. Final composite blade showing the completed laminated structure and smooth aerodynamic surfaces.

5. DISCUSSION

The central outcome of this research is a ply resolved geometric definition of a twisted composite propeller

blade that can be used for structural analysis and manufacturing preparation without redefining the laminate architecture. Ply boundaries, nominal thicknesses, material assignments, fibre orientations, and core geometry originate from the same CAD model and remain consistent during their transfer to the layered finite element model and the generation of cutting patterns. The scientific contribution therefore lies in establishing geometric continuity between laminate design, numerical assessment, and manufacturing data for a composite blade with continuously varying curvature and twist. Such continuity is important because the local material directions of a composite blade cannot be described adequately by a single global coordinate system. Changes in blade curvature and twist modify the relationship between the prescribed fibre direction and the structural load path. Defining each ply relative to the local blade surface preserves the nominal orientation field and thickness distribution throughout the span. Equivalent homogeneous models may reproduce global stiffness after suitable calibration, but they do not retain the geometric information required to examine individual plies or prepare manufacturing patterns. In the adopted procedure, the surfaces used to define the numerical plies also determine their flattened cutting geometries, reducing inconsistencies between the laminate assumed in the analysis and the laminate prepared for lay up.

The structural response obtained at 2500 rpm corresponds to the expected load transfer in a blade dominated by centrifugal force. The resultant centrifugal load of 54 kN produced a gradual increase in displacement from the constrained root towards the tip. Ply stresses reached their highest values near the root and decreased along the span because each section carries the centrifugal action generated by the material located farther from the axis of rotation. This response is mechanically consistent with the radial tensile loading of a rotating blade and indicates that the layered model preserves the expected relationship between blade geometry, mass distribution, and laminate stiffness. Assessment according to the Tsai–Hill failure criterion requires particular attention near the constrained boundary. An inverse reserve factor of 1.6 occurred in a small number of surface elements directly adjacent to the fully fixed attachment. The abrupt suppression of all translational and rotational degrees of freedom produces a localized stress concentration whose magnitude is sensitive to the boundary representation and mesh density. This peak should therefore be interpreted as an indication of boundary condition sensitivity rather than evidence of general laminate failure. Outside the elements directly influenced by the constraint, the highest inverse reserve factor was 0.8 and occurred in the blade root region and near the sleeve exit. The value remained below the failure threshold, indicating that first ply failure was not predicted in the structurally representative part of the blade under the analysed centrifugal load. The local result at the attachment cannot, however, be dismissed without further examination. Reliable assessment of this region would require explicit representation of the sleeve, hub, contact interfaces, and load transfer through the connection. The mesh convergence study estab-

lished the stability of the global directional deformation and supported the selection of a 4 mm element size, but deformation convergence does not necessarily imply convergence of local ply stresses at a fully constrained boundary. In addition, the Tsai–Hill criterion identifies the onset of failure through the interaction of ply stress components but does not distinguish between failure modes or describe damage evolution after first ply failure. The calculated mass of 1985 g and centre of mass coordinates of $x=397.18$ mm, $y=0.66$ mm, and $z=97.05$ mm were obtained from the spatial distribution of the plies and foam core. These quantities are particularly relevant to centrifugal loading, propeller balancing, and subsequent dynamic assessment. They remain nominal values because their accuracy depends on the assigned densities and cured ply thicknesses. Resin content variation, laminate compaction, and local material accumulation during layup may alter both the mass and its spatial distribution. Comparison with measured mass properties would be required to quantify the influence of these manufacturing deviations. The practical applicability of the geometric model was examined through the preparation and manufacture of the blade. Flattened patterns generated from the ply surfaces were used for automated cutting and placement in the mould according to the prescribed stacking sequence. The manufactured component demonstrates that the ply definition can be transferred from the three-dimensional model to the layup process without preparing an independent set of cutting templates. This represents an important engineering consequence of the methodology because the manufacturing data remain traceable to the laminate evaluated numerically.

The correspondence between the nominal and manufactured laminate nevertheless cannot be established from visual appearance alone. Flattening a ply located on a doubly curved surface does not fully account for prepreg deformation during placement. Draping may produce local changes in fibre angle, ply boundaries, and thickness, particularly in regions of pronounced curvature. Ho-Nguyen-Tan et al. [24] demonstrated that these effects can influence both the geometry and structural response of composite blades. Dimensional inspection and non-destructive examination would therefore be necessary to determine whether local wrinkling, bridging, gaps, or thickness variations occurred during production. Previous investigations have combined different stages of propeller development, but their principal research objectives differ from those of the present work. Kovačević et al. [11] considered propeller design, computation, manufacturing, and testing as a broader development procedure. Park [17] addressed the aeroelastic analysis of a turboprop blade, while Rokvam et al. [18] focused on experimental verification of bend-twist deformation. Malim et al. [23] examined structural modelling and validation of an additively manufactured propeller, whereas Ho-Nguyen-Tan et al. [24] investigated fibre draping and process induced deformation in a thermoplastic fan blade. In comparison, the specific contribution of this research is the preservation of detailed ply geometry as a common definition for CAD modelling, layered finite element analysis, mass property calculation, ply flattening, and manufacture of

a conventionally laminated prepreg blade. The conclusions are restricted to the dominant centrifugal load at 2500 rpm and should not be interpreted as a complete assessment of operational behaviour. A rotating propeller is also affected by bending, torsion, vibration, and coupling associated with its initially twisted geometry. Further development should therefore include a more representative attachment model and combined operational load cases, followed by progressive damage and fatigue analyses. Experimental measurement of blade mass, centre of mass, strain, and deformation would provide the evidence required to determine the predictive accuracy of the numerical model and support subsequent operational qualification. No quantitative reductions in production time, material consumption, or cost were measured in this study. Therefore, such benefits cannot yet be claimed. The procedure requires composite oriented CAD expertise and appropriate software and manufacturing equipment, which may increase the initial implementation effort. The same workflow can be transferred to other curved layered aerospace components, provided that their geometry, laminate architecture, loading, and manufacturing constraints are redefined.

6. CONCLUSION

This study presented a method for developing a composite propeller blade with complex curvature and twist from a single three-dimensional CAD definition. Starting from the aerodynamic surface, each ply was created individually, with its material, thickness and fibre direction retained throughout the formation of the laminate. The resulting model provided the basis for finite element analysis and the determination of blade mass properties. The ply surfaces were also flattened to obtain the cutting patterns used during layup and manufacture. Connecting these stages through a common laminate definition reduced the need for repeated geometric reconstruction and the possibility of discrepancies between the blade configuration used in the analysis and that prepared for production. The structural analysis showed the response of the prescribed laminate under the selected centrifugal load, while the manufacture of the blade demonstrated that the generated ply geometry could be transferred to the chosen production procedure. The methodology established a clear connection between geometric modelling, structural assessment and manufacturing preparation, while preserving the origin and continuity of the ply data throughout the process. Although it was demonstrated on a propeller blade, the procedure can also be applied to other curved composite structures whose geometry and laminate architecture must be transferred consistently from design to manufacture. Future research will focus on the experimental validation of the developed model by comparing the numerical predictions with measurements obtained from the manufactured blade.

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МОДЕЛОВАЊЕ КОМПОЗИТНОГ КРАКА ЕЛИСЕ СЛОЖЕНЕ ГЕОМЕТРИЈЕ И ЛАМИНАТНЕ СТРУКТУРЕ

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За поуздано пројектовање композитних лопатица елисе неопходно је повезати геометријско моделовање, дефинисање структуре ламината, прорачунску анализу и припрему за производњу. Конвенционални CAD поступци најчешће се заснивају на еквивалентним запреминским моделима или поједностављеним облицима ламината, који не омогућавају прецизно представљање континуитета слојева, локалне оријентације влакана и промена дебљине на витопереним аеродинамичким површинама. У раду је представљена производно оријентисана методологија за тродимензионално моделовање и структурну анализу композитне лопатице елисе сложене геометрије и експлицитно дефинисане архитектуре ламината. Појединачни слојеви моделовани су уз очување аеродинамичке површине и уважавање редоследа слагања, дебљине слојева дуж распона лопатице. CAD модел представља геометријску основу за анализу методом коначних елемената при задатим условима оптерећења, одређивање масених карактеристика и положаја центра масе, као и генерисање геометрије појединачних слојева за потребе производње. Примена методологије на лопатици састављеној од влакнима ојачаних слојева и испуне показала је да се тачност приказа геометријских и материјалних карактеристика може очувати током пројектовања, анализе и припреме за производњу, без додатне реконструкције геометрије.