DEFINITION OF THE BOLT SHEARING LOAD IN MULTIPLE-ROWMETAL COMPOSITE JOINTS

Ya. S. Borovskaya, M. A. Glebova, V. I. Grishin
Central Aerohydrodynamic Institute, Zhukovsky Str., Zhukovsky, 140180, Moscow, Russia

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Abstract

A method of the definition of the bolt shearing load in multitier metal composite joints in design calculations of integral units of aircraft structures is presented. A comparison of the calculated results with experimental data of 3D modeling and calculation of connections in the ANSYS program is made.

The aircraft design consists of many components and elements such as aircraft skin, ribs, spars, and others. To connect them, bolts, rivets, adhesives, welding, and other technologies are used. In order to provide the transport ease, the inspection, the repair, or replacement, the aircraft units are manufactured with the possibility of disassembly. To connect the detachable units, the bolts are typically used. These compounds appear to be the weakest design elements; so, for reliability control of joint areas, the greatest attention is given to the assembly procedure while designing. The connection failure in operating can occur due to various reasons, e.g., because of the influence of assembly stress, the presence of stress concentrators, deviations from the production technique, or as a result of combination of these factors. The influence as a whole is very difficult to be assessed. These factors directly affect the strength of bolted connections, with the appearance of various cuts, splits, and the presence of irrational designed cutouts and edging strips, as well as forms of conjugated fragments of construction lines being constituted a particular danger.

In most critical cases, the emergence of stress concentration is a consequence of reducing the cross-sectional area of product resulting when drilling holes for bolts.

The weight efficiency of the design [1, 2] to be achieved and the successful implementation of the strength potential of composites in aircraft structures to be realized by careful designing of compounds are of a particular importance. This challenge is even more relevant for composite materials than for metallic compounds, since reinforcing elements of the composites are usually very fragile. In addition, the composites generally do not possess a significant ability to transfer loads, which is inherent to plastic materials.

Thus far, the technology of manufacturing compounds for aircraft metal structures is well established. Types of damage of joints of composite elements are practically equivalent to those that occur for metallic compounds, but the behavior of composite materials differs from that of similar metallic compounds for the following reasons: relatively high vulnerability of the material associated with a high stress concentration in the edges of holes; splitting of the composite in the transverse directions because of its multilayer structure, etc.; and the inhomogeneity of the composite properties and the peculiarities of its interaction with metal fasteners, significantly different from the interaction with metal fasteners.

Shown in Fig. 1 are the typical forms of mechanical failure of composite compounds. These are cuts, gap on the net section, and the combination of gap and cut. Destructions due to cuts and shear in the plane of cut can also be combined. Sometimes, along with composite bearing, pulling out of the fastener head or breaking of the bolt from its bend occur.

The strength of a mechanical connection is determined as follows.

Bearing:

$$P = dt\sigma_{bear}$$

Cut:

$$P = 2\,[(e/d) - 0.5]dt\sigma_{cut} \qquad (1)$$

Break:

$$P = 2\,[(s/d) - 0.5]\,dt\sigma_{br}$$

Where: P is the breakdown force of bolting; σ_{bear} is the working bearing stress; σ_{cut} is the working tangential shearing stress; σ_{br} is the working tensile stress; d is the diameter of the fastener; t is the thickness of the composite side plate and e and s are the linear dimensions (see Fig. 1). The shear force [Eq. (1)] can also be used for the case of combined cut and sample break, if the working stress is known empirically.

When designing a classic structure diagram of thin-walled structures of the wing and fuselage, 2D simulation is effectively used. When calculating the shearing load of bolts, the compound can be modeled by a set of 2D plates and discrete elastic elements (tightening bolts, screws, rivets, metal pins). Adhesive joints, which of the calculation has its own features, can be also used as the fasteners. These same techniques are acceptable in calculation of the reinforcement plates that are used to repair parts been damaged in service. Obtained by means of expressions (1) in 2D simulation, the distribution of forces between the bolts can be used to assess the static strength of composite packets, and to take this distribution as a boundary condition in the study of the most loaded fasteners. In this case, the subsequent calculation using the 3D grid allows to determine the stress-strain state of a node connection, and to assess its durability.

There is a large number of studies devoted to the distribution of forces in bolted and riveted joints. For example, in [3, 4] for calculating the complex spatial bolted and riveted joints, the approach based on the following assumptions is substantiated: material of construction is elastic, and the possibility of elastoplastic operation of bolted or riveted joints is not assumed; friction between the individual elements of the connection is missing; and the principle of superposition of solutions (for stresses) is valid. Nevertheless, this technique is quite suitable for qualitative evaluation of the bonding strength, and can be used to predict their durability in the initial design stage.

When using the finite element method (FEM) to calculate compounds with numerous bolt or rivet connections, it is it is reasonable not to split the bolt body by the elements that

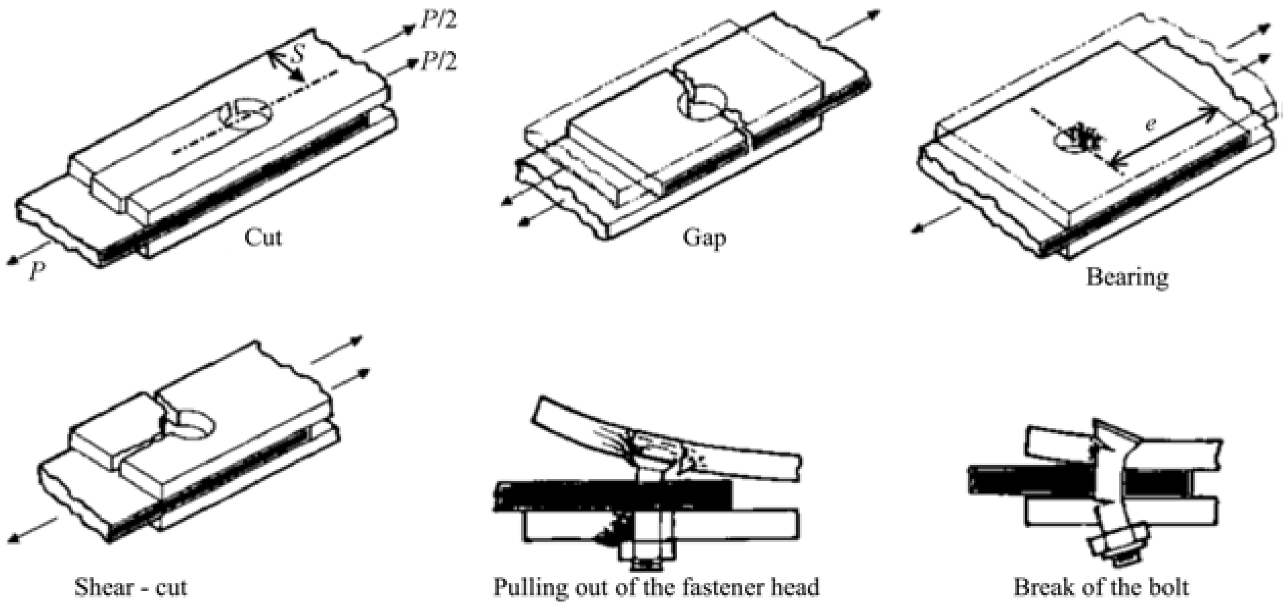

Fig. 1. Forms of destruction of composite compounds

would significantly increase the dimension of the problem, but to find the stiffness coefficients (compliance) of the bolts based on analytical or experimental dependences, and summarize them with stiffness coefficients of basic parts of the construction (Fig. 2). The relationship between the offset of the bolt ends (rivets) δ_b and shear load P acting on it is as follows:

$$\delta_b = CP \text{ or } K_b\delta_b = P, \qquad (2)$$

where $K_b = C^{-1}$.
Here, **C** is the attach point compliance, and K_b is the attach point stiffness.

Equilibrium equations linking the vector of node design δ with the vector of nodal forces $\mathbf{R} = \mathbf{R}_{up,l} + \mathbf{R}_b$ can be written as

$$K\delta = R, \qquad (3)$$

where the stiffness matrix $K = K_{up,l} + C^{-1}$, and the "up" and "*l*" denote the upper and lower parts of the compound (see Fig. 2).

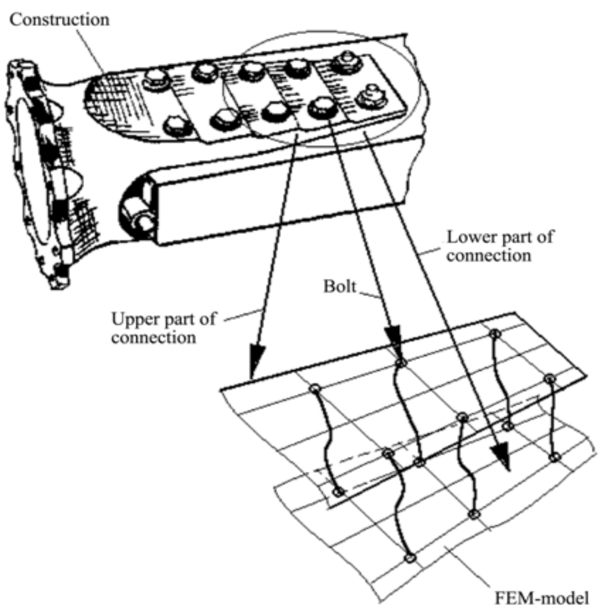

Fig. 2. FEM model containing discrete links

In constituting the stiffness matrix $K_{up:l}$ in (3), finite elements with linear or quadratic approximation of displacements in the plane of the element are typically used, and for calculating compliance of bolts **C**, calculated or experimental data is used.

Calculated and experimental studies while determining the compliance **C** of fasteners gave rise to use a number of experimentally verified relationships. Thus, according to [4] the compliance for metallic compounds is defined as the sum of the bearing deformation (δ_{bear}) of the package, bending (δ_{bend}), and bolt shear (δ_{shear}) as

$$\delta_{\text{bear}} = P\left(\frac{1}{E_b}\left(\frac{1}{2t_{\text{side}}} + \frac{1}{t_{pl}}\right) + \frac{1}{E_{pl}}\left(\frac{1}{2t_{pl}} + \frac{1}{t_{pl}}\right)\right),$$

$$\delta_{\text{bend}} = \frac{P\left(8t_{side}^3 + 16t_{side}^2 t_{pl} + 8t_{side}t_{pl}^2 + t_{pl}^3\right)}{\left(192E_b I_b\right)}, \qquad (4)$$

$$\delta_{shear} = \frac{P\left(2t_{side} + t_{pl}\right)}{6G_b F_b},$$

where F_b and I_b are the cross-sectional area and bolt mass moment of inertia, t is the thickness of the details, and the indices "side," "pl," and "b" refer to the side and central plates, and bolt, respectively.
For composite connections, elastic modules in the expressions (4) are recommended to be calculated by the expression

$$E_i = E_1 \cos^2\alpha + E_2 \sin^2\alpha,$$

where index i denotes either a lateral or central plate, and α is the angle between the direction of maximum modulus of composite packet E_1 and force P acting on the bolt, and E_2 is the elastic modulus.

The method described in this work is implemented by the authors as a program, FITCOM [3] and mentioned calculations were made with use of it. The results of the calculation of bolt shearing load received from the program FITCOM were checked by comparison with the results of shearing load in one of the joints of the wing obtained from the program ANSYS, which is used on a wide scale in aviation technology.

The reliability of the proposed numerical-analytical method is justified by comparing the results of calculations and data obtained in a series of experiments with two-, three-, four-row metal composite compounds, the scheme of one of which is shown in Fig. 3. Given in Table 1 are the dimensional characteristics of the samples. As the material of side plates, carbon fiber plates made from carbon monolayers of 0.12 mm thickness were taken. Elastic moduli used were E_1=13500 kgf/mm2 and E_{22} = 880 kgf/mm^2, with modulus of transverse elasticity G_{12} = 447 kgf/mm^2 and Poisson ratio μ_{12}=0.33. Side plates were attached to the central duralumin or steel plates with nonhidden bolts with shear strength τ_b = 60 kgf/mm^2, tightened to the axial stress of σ = 15 kgf/mm^2.

The elastic modulus of the aluminum plates is taken to be 7200 kgf/mm^2, Poisson ratio μ = 0.3, tensile strength $\sigma_b \geq$49 kgf/mm^2, and yield stress $\sigma_{0.2}$ = 35 kgf/mm^2. Plates of steel 30CrMnSiA had an elastic modulus of 21x103 kgf/mm^2, tensile strength σ_b=108 kgf/mm^2, and yield stress $\sigma_{0.2}$ = 75 kgf/mm^2. The destruction of all samples occurred as a result of the break of carbon fiber plates at an angle of ± 45 deg relative to the hole of the first most loaded bolt, measured from the junction zone of metal plates, as shown in Fig. 4.

Shown in Table 2 and in Fig. 5 is a comparison of calculated results with experimental data for a typical two-, three-, and four-row samples. The relative difference γ between theoretical and experimental results was determined by the following expression:

$$\gamma = \frac{K_{calc} - K_{\exp}}{K_{calc}} 100\%,$$

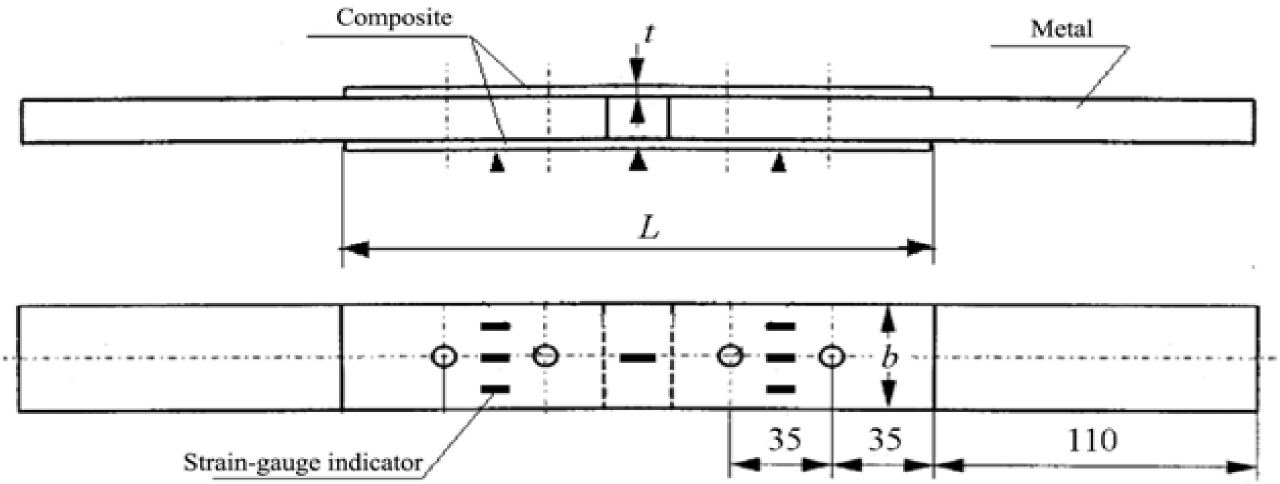

Fig. 3. Scheme of double-row metal composite sample

Table 1. Characteristics of samples

Sample type	Length of side plate *L*, mm	Width *b*, mm	Thickness of side CM plate *t*, mm	Bolt diameter *d*, mm	Number of bolts	Material of metal insert	Metal insert thickness, mm
1	200	40	3.72	8	4	Duralumin	14
2	210	40	2.4	6	6	Duralumin	10
3	270	40	2.76	6	8	Duralumin	12

Fig. 4. Typical destruction of the sample

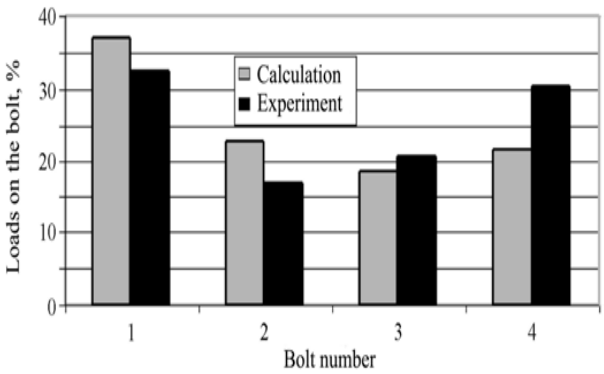

Fig. 5. Shearing load distribution of a typical four-row bolt compound

where K_{calc} is the relative value of bolt efforts in compounds in percentage terms, obtained by calculated (i = calc) and experimental (i = exp) means. Each experimental point was obtained based on average test results for three samples.

Composite wing panels were connected to the center section panels with two titanium fittings by titanium bolts, the diameter of which is 12 mm. Shown in Fig. 6 is a fragment of the joint designed in the CATIA system.

Mechanical properties of materials are taken as follows: the elastic modulus of titanium $E = 11\text{x}10^3$ kgf/mm^2, Poisson ratio $\mu = 0.3$ (Fig. 7). The composite panel is assumed to be orthotropic with the following properties of the package: E_x = 7716 kgf/mm^2, E_y = 3402 kgf/mm^2, E_z = 1280 kgf/mm^2, G_{xy} = 1918 kgf/mm^2, G_{yz} = 300 kgf/mm^2, G_{xz} = 300 kgf/mm^2, $\mu_{xy} = 0.46$, $\mu_{yz} = 0.3$, and $\mu_{xz} = 0.3$. In calculating the 3D model, the contact interaction between the details of the joint and the bolt bodies corresponds to the friction coefficient being equal to 0.15.

Table 2. Comparison between calculation and experiment

Bolt	Calculation,%	Experiment,%	γ, %
Sample 1			
1	54.8	57.7	-5.3
2	45.2	42.3	6.4
Sample 2			
1	41.6	42.0	-0.9
2	28.8	24.3	15.5
3	29.6	33.7	-13.8
Sample 3			
1	37.1	32.4	12.7
2	22.9	16.6	27.5
3	18.4	20.5	-11.4
4	21.6	30.5	-41.2

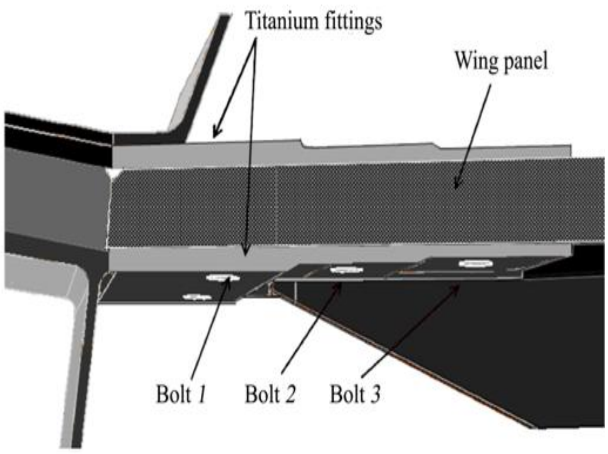

Fig. 6. Bolts numeration in the joint

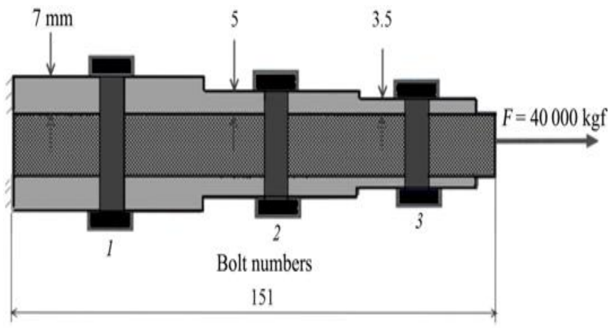

Fig. 7. Model of bolt joint of panels

The experimental results presented in Fig. 8 show that the first couple of bolts have 13.6–13.9% of the total load, and the second pair of bolts has 14.6–15.3% of the total load, while the third pair of bolts has 20.7–22% of the total load. The “asymmetry” of the results is associated with a small asymmetry of the model. Shown in Fig. 9 is the finite element scheme of the same joint, which was investigated by using the FITCOM program. A symmetric problem with three bolts was considered, and the external load was assumed to be 20 tons.

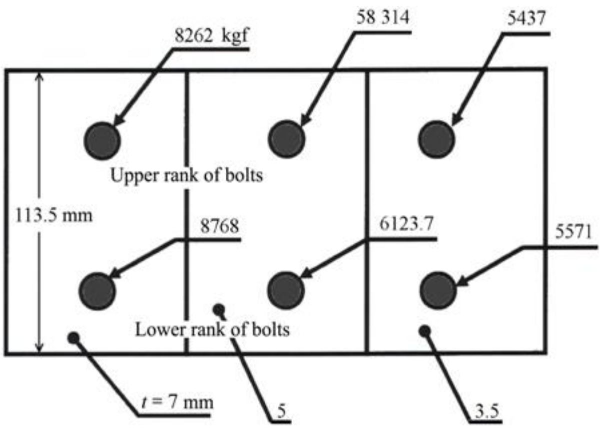

Fig. 8. Values of shearing load in bolt joints (ANSYS).

Shown in Fig. 10 and in Table 3 is a comparison between the compound shearing load obtained using the ANSYS and FITCOM programs. As appears from Table 3, the difference in the results obtained by averaging the shearing load in the bolt ranks using the ANSYS and FITSOM programs does not exceed 3.5%.

Thus, the technique of determining the bolt shearing load is developed in multiplerow metal compounds. The technique is based on the simulation of real 3D structures by simple 2D models with analytical relations for the description of discrete links.

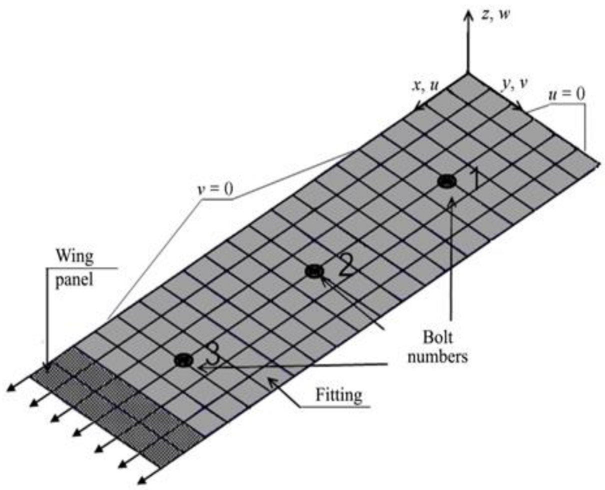

Fig. 9. Finite element model of the compound (FITCOM)

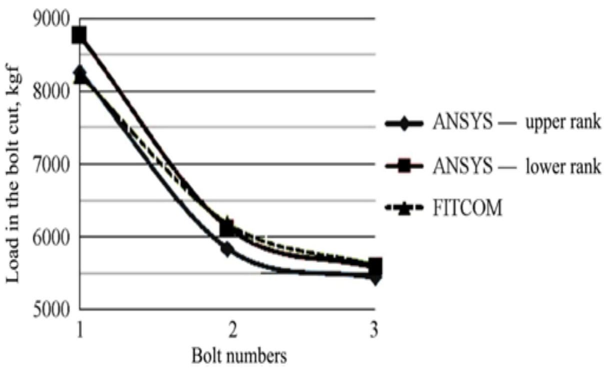

Fig. 10. Comparing the results obtained by the ANSYS and FITCOM programs

TABLE 3. Comparison of the results

Bolt number	ANSYS upper row, kgf	ANSYS lower row, kgf	ANSYS average, kgf	FITCOM, kgf	γ,%
1	8263	8768	8515	8220	3.5
2	5831	6124	5978	6180	-3.4
3	5437	5571	5504	5600	-1.7

A comparison of calculation results obtained by the FITCOM program with experimental data and the 3D simulation of real contact between compound details obtained by the ANSYS program indicates their good correspondence. The technique can be used in the design calculations of integral units of aircraft structures and the analysis of local strengths of complex details.

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