REVERSE ENGINEERING OF THE WING OF TU-154M AIRCRAFT

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Abstract

Using the available information as well as basic design principles, the structure of Tu-154M aircraft outer wing is recreated in this paper. The essential stress levels experienced under the limit loads are shown and the natural frequency is calculated. Some basic dynamic responses are also evaluated. This work is to be seen as the first step in recreating the whole wing as well as its performance under extreme conditions

Keywords *- Reverse engineering, Aircraft wing modelling, Wing dynamics.*

Streszczenie

Używając dostępnych informacji jak również podstawowych zasad projektowania, została odtworzona w tym artykule konstrukcja skrzydła zewnętrznego samolotu Tu-154M. Pokazane są podstawowe wielkości naprężeń pod obciążeniami użytkowymi a także jest obliczona podstawowa częstość własna. Zrobiona została także ocena niektórych efektów dynamicznych. Praca ta powinna być traktowana jako pierwszy krok przy odtwarzaniu całego skrzydła a także oceny jego zachowania w skrajnych warunkach.

Słowa kluczowe **–** *Inżynieria odwrotna, modelowanie skrzydła samolotu, dynamika skrzydła.*

1. INTRODUCTION

An engineer, who sets out to create a simulation of a mechanical event is usually faced with incomplete data that he needs to create a model. In the case of Tu-154M airliner the problem was especially acute because the original technical drawings were not available to this writer, so he could only rely on the information included in [1] and [2], photographs of the airframe, either original or disintegrated and various miscellaneous sources. One approach, which could be broadly called parametric design, was exemplified in the work of Zhang and Binienda [3]. They would assume a number of structural parameters (usually thicknesses) that seemed feasible and executed models containing those. They would then make the appropriate conclusions about the performance of the structure.

The approach employed here is quite different. While the same sources as before, i.e. [1] and [2] are used with the purpose of making the FEA model similar to the original structure, the wing is designed *ab ovo*. This is accomplished using the external geometry based on [4], known material properties and the same loads as assumed for the sister aircraft, Boeing 727-200. The differences between the FEA model so created and the real structure exist, largely due to convenience of modelling involved in our approach. Yet, it is expected that those deviations have little effect on the final, numerical results.

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2. PRELIMINARY DESIGN

The starting point was the external geometry of the plane as available from [4]. The material properties of the aluminium alloys used were

2024-T3: F_y = 293 MPa, F_u = 448 MPa and e = 16%,

7075-T6: F_y = 493 MPa, F_u = 545 MPa and e = 9%,

which are the averages of published data and which are similar to the original Russian alloys involved. The second of the above was used for stringers while the first for the remaining structural elements. The other important properties were E = 72000 MPa, ρ = 2800 kg/m3 and ν = 0.33.

The allowable stress for the first alloy was Fy, while for the second it was Fu/1.5 = 363 MPa, in order to have a minimum safety factor of 1.5.

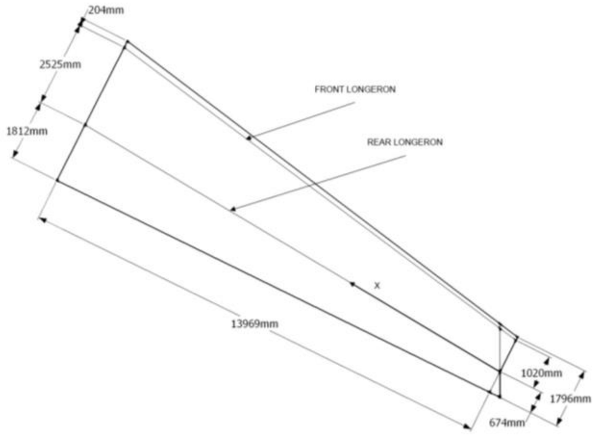

Fig. 1. Simplified outer wing, outline in plan view. The wing tip was slightly altered to make it parallel to the root section.

The take-off mass of the aircraft was 110.5 tonnes, while the landing mass was assumed as 78.6 t. The design was based on the average of the two, i.e. M = 94.6 t. The basic design condition was the overload factor of 4.0, or the limit load of 4 g. This results in the limit lift of 189.2 t per each of two wings and this is regarded as carried entirely by the wing surface. The projected surface area of the outer wing was calculated to be 0.5745 of the total and, accordingly, the outer wings would carry 108.7 t of lift each. The pressure on the wing surface was assumed to be uniformly distributed along span.

The projected structural area A (between the front and the rear longeron as in Fig. 1) and the average lift pressure p_o are, respectively:

$$A = 0.5(1020+2525)x13,969 = 24.76 \text{ m}^2, \quad (1)$$

$$p_0 = \frac{108,700x9.81}{24.76x10^6} = 0.0431 \text{ MPa} = 43.1 \text{ kPa}. \quad (2)$$

It is now a simple matter to determine the (structural) chord length c(x), shear force Q(x) and the bending moment M(x) as a function of distance from tip, x:

$$c(x) = c_1 + 0.10775x, \quad (3)$$

$$Q(x) = 0.5(c_1 + c(x))\, xp_0, \quad (4)$$

$$\mathcal{M}(x) = \left(\frac{x}{3}\right)\frac{c(x)+2c_1}{c(x)+c_1}Q(x), \quad (5)$$

where c_1= 1020 mm is the tip chord. With regard to chordwise pressure distribution, one must note that for thin profiles like one used here the aerodynamic center is at about 0.25 of the chord length from the nose. A typical moment coefficient is 0.1, which moves the line of the resultant pressure somewhat backwards. The resultant lift force is relatively close to the resultant of the assumed uniform pressure distribution between the front and the rear longeron. This was the reason behind assuming the latter distribution in the design. (One should note that even if the chordwise pressure distribution were drastically different, it would not affect the validity of Eqs. 3-5.)

The drag force was related to the engine thrust. Each of three engines develops a nominal thrust of 9500 kG, according to [1]. It was assumed that during steady flight the nominal thrust of two engines is involved, so there is a total of 19,000 kG of drag. By comparing the frontal areas involved it was estimated that each outer wing develops 10 % of the total drag. The drag force along the axis of the plane would therefore be P_D = 0.1x19000x9.81 = 18,640 N. But the leading edge is swept back by about δ = 37.30 and the drag, resulting mostly from pressure, is thus $P_D/\cos\delta$. When this pressure is uniformly applied over the nose part, its resultant is equal to pressure on the front longeron. The value of that pressure is relatively small, 6.16 kPa.

After the bending moments and the shear forces in various sections were calculated, one could make the initial choice of element sizes using simple equations of mechanics of materials as well the allowable stress levels stated before. The input for the Ansys [5] code was then created.

The scope of this presentation is limited to the outer wing only. For a conventional design of this part of the aircraft nothing else is needed, as the outer wing can be assumed to be rigidly fixed at its end. Yet, when the dynamic response is considered, the center wing should be included, at least in a simplified way, as it constitutes a deformable base for the outer wing.

3. FEA MODEL AND STATIC RESULTS

The skin along the span was divided into 7 segments, as shown in Fig. 2. (It is really 6 equal segments except that one on the left was divided in two.) The nose was subdivided into 3 segments. The properties are constant along the segments. The top and bottom skin at the root section is 7 mm thick. Modelling of stringers is depicted in Fig. 3.

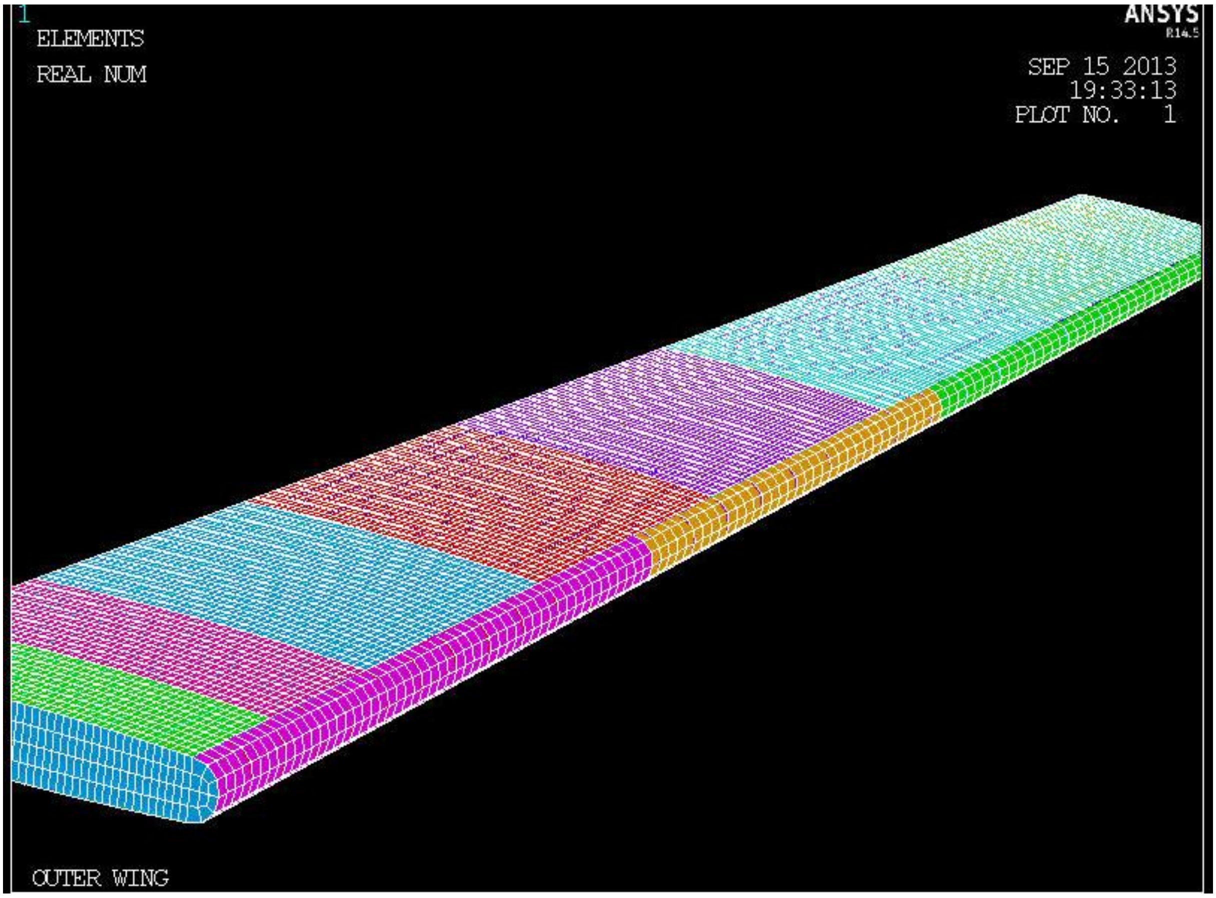

Fig. 2. Over-all view showing the top skin divided into seven chordwise segments as well as the nose skin divided into three segments.

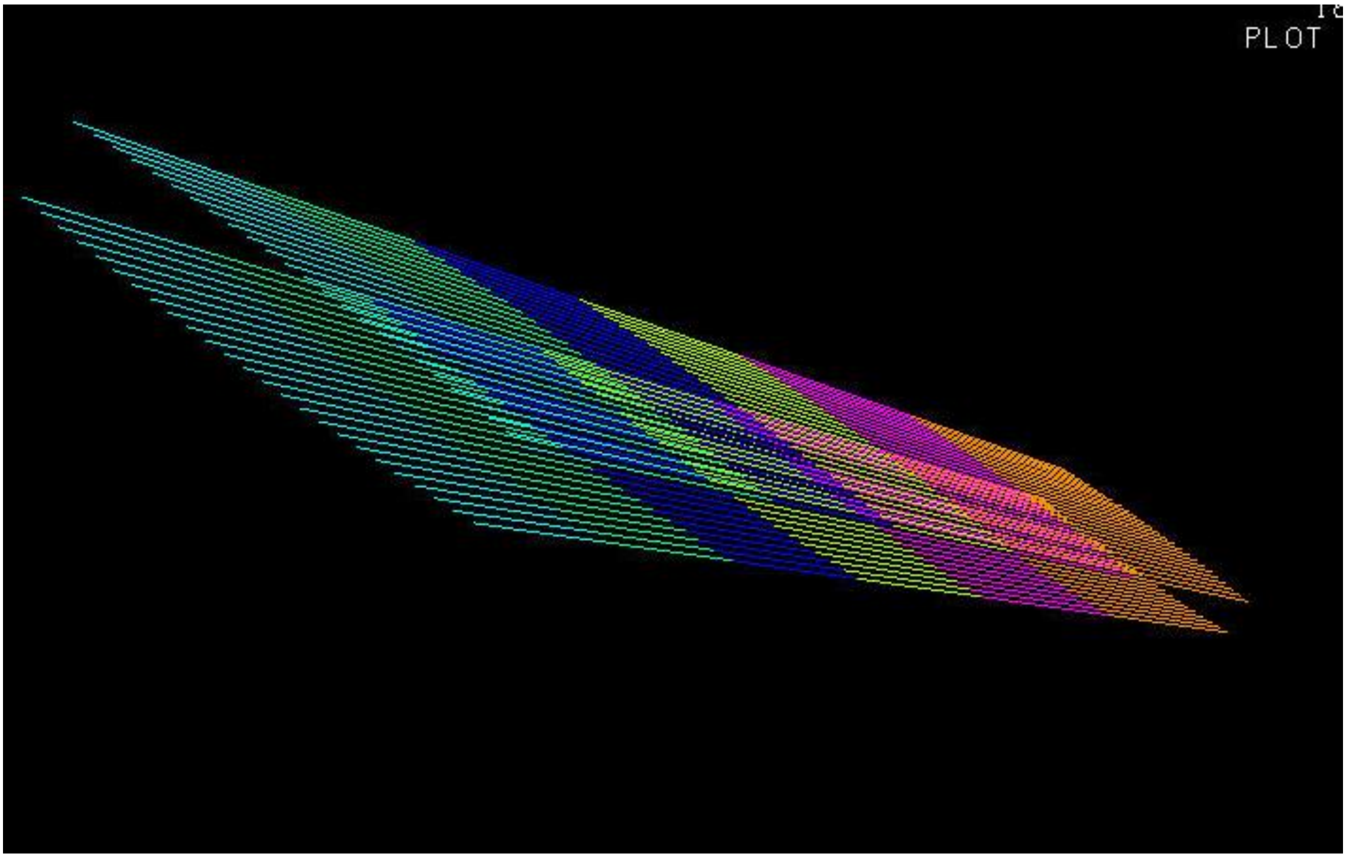

Fig. 3. **Six segments with different properties of stringers are marked. The left, right and center stringers serve as longeron caps.**

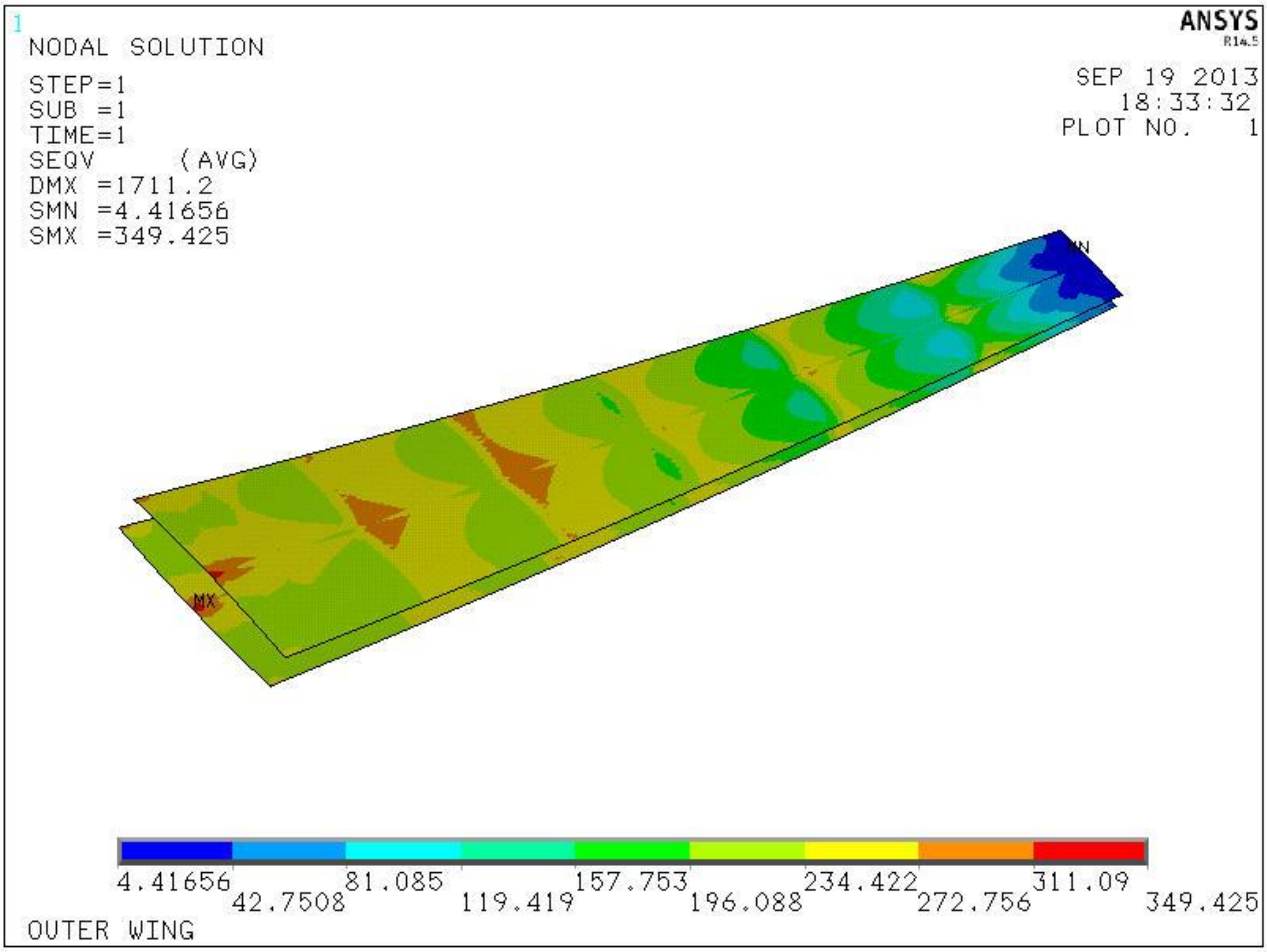

Fig. 4. Equivalent stress distribution in skin.

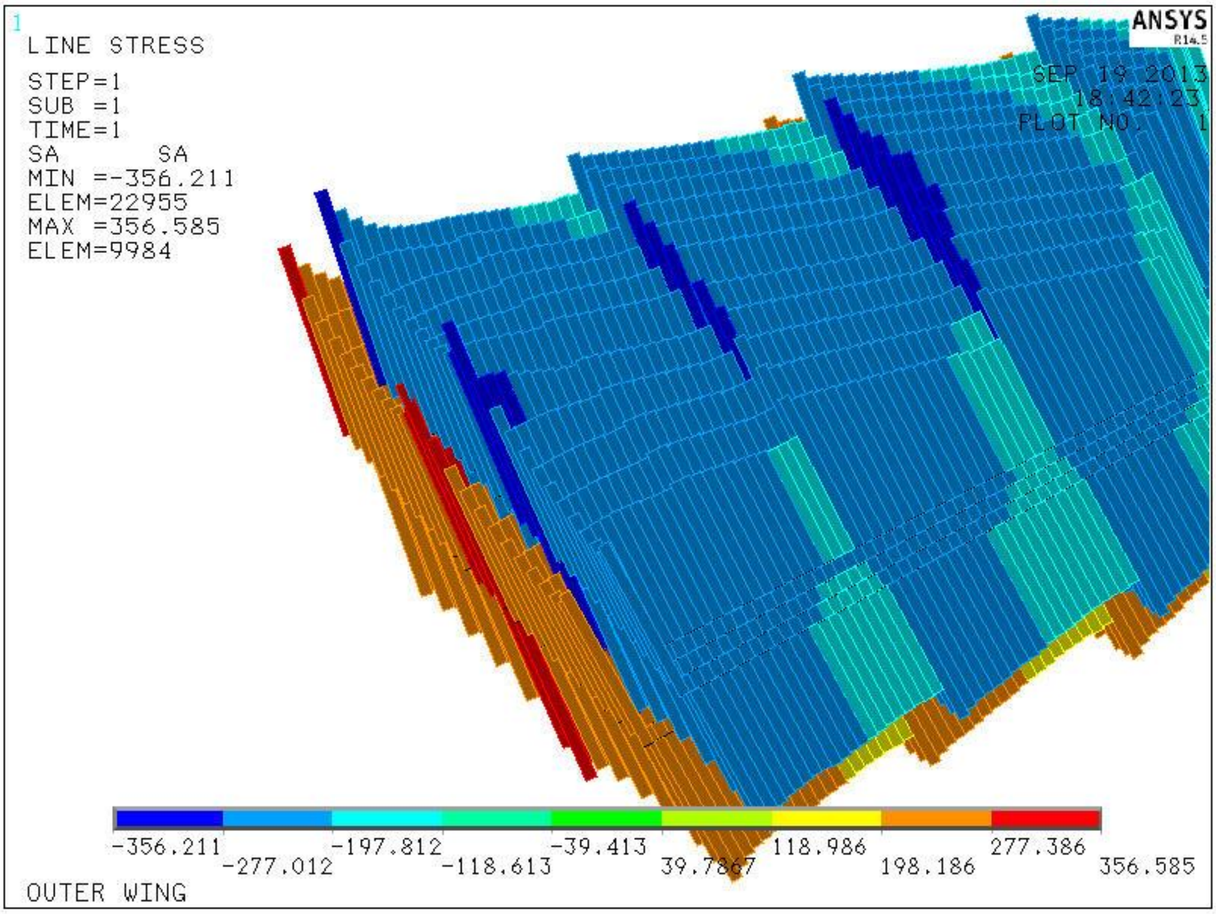

Fig. 5. Axial stress in stringers plotted at right angle.

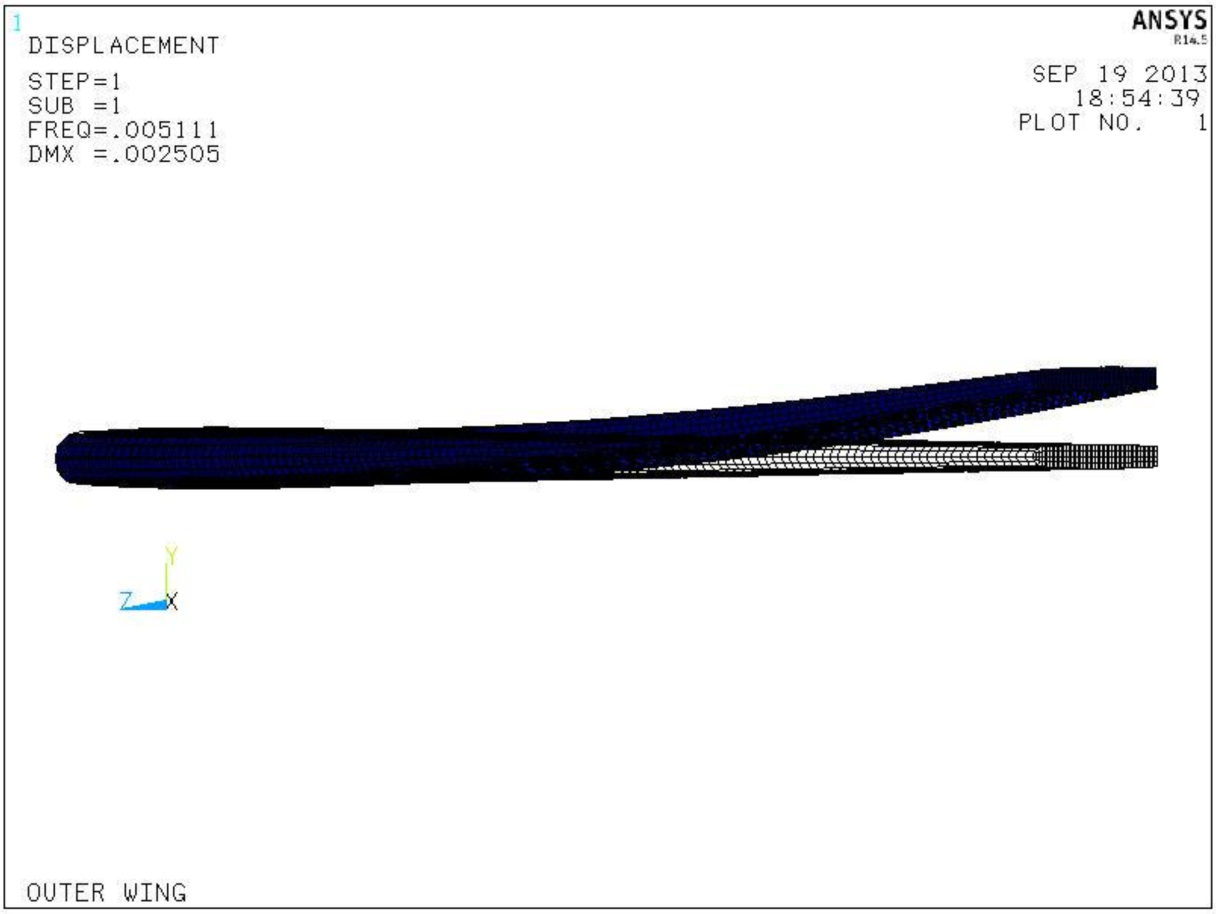

Fig. 6. **The first natural mode of vibration, bending, at 5,11 Hz.**

There are 24 spaces between stringers. (In the original structure most stringers were terminated on the way from the root to the tip. In this design we keep the number of stringers unchanged and reduce their properties when going towards the tip.) Ansys element Beam4 was used for beams and Shell63 for shells.

When estimating the inertial properties of the model, several types of non-structural masses were included. One was the rear part of the wing, and the other was to compensate for the deviation from a simple outline, near the leading and the trailing edge, to accommodate moving surfaces. Also, an allowance was made for various mechanical devices. The total mass of the outer wing was program-calculated to be $M_w = 2994$ kg.

When the limit lift along with the drag were applied to the wing, the maximum tip deflection was found to be us = 1711 mm. The equivalent (Huber-Mises) stress is shown below for the skin and a direct stress for the stringers. The vertical reaction at the root was 1065 kN, somewhat less than $4M_wg$.

The allowable stress in skin was set at 293 MPa. Fig. 4 shows that it is exceeded at places and the adjustments must be made. In stringers the allowable stress is set at 363 MPa and, according to Fig. 5, this is exceeded only at a few spots. The stress distribution problem can be addressed by adding material at some areas while removing it at others. The mode/frequency analysis was also carried out and the first natural mode, bending at 5.11 Hz is shown in Fig. 6 The results described above are preliminary only. The necessary changes to the model are intended to be carried out in the future..

4. DYNAMIC RESPONSE

As mentioned before, the outer wing under consideration is not really fixed at the root, but it is attached to the center wing, which may be viewed as the elastic base. While it is expected that the flexibility of that base is small, it will still make a difference in the natural frequency and the associated mode of deformation of the structure. Yet, some operations will be carried out below, primarily to demonstrate the nature of the response.

The dynamics part is simulated using the explicit LS-Dyna code [6], which is an efficient tool for dynamic response analyses. The shells are modelled as Belytschko-Tsay and the beams as Hughes-Liu elements.

Consider a dynamic application of pressure to the outer wing, treated as a tapered cantilever fixed at the base. To be more specific, pressure will vary as a step load, i.e. it will suddenly grow to the value of p_0 and remain there indefinitely. To have a direct comparison with static loading effects discussed above and, at the same time, to avoid unreasonably large response, apply p_0 equal to only 1/4 of the limit load used before. (One could question, of course, the reduction of drag to a fraction of that in a steady-state flight, but the immediate interest here is the comparison of static and dynamic effects.)

The peak displacement in Fig. 7 is 849 mm. The most convenient way of comparing this with the corresponding static displacement is to use the dynamic factor, which in this case is

$$\mathrm{DF}(u) = \frac{u_{\mathrm{d}}}{u_{\mathrm{st}}} = \frac{4\mathrm{x}849}{1711} = 1.985 \quad ,$$

where the factor of 4 comes from the use of 1.0 g rather than 4 g pressure loading. The reaction at the root peaked at 476,300 N. The dynamic factor for the reaction was therefore

$$\mathrm{DF}(R) = \frac{R_{\mathrm{d}}}{R_{\mathrm{st}}} = \frac{4\mathrm{x}476,300}{1.065\mathrm{x}10^6} = 1.79$$

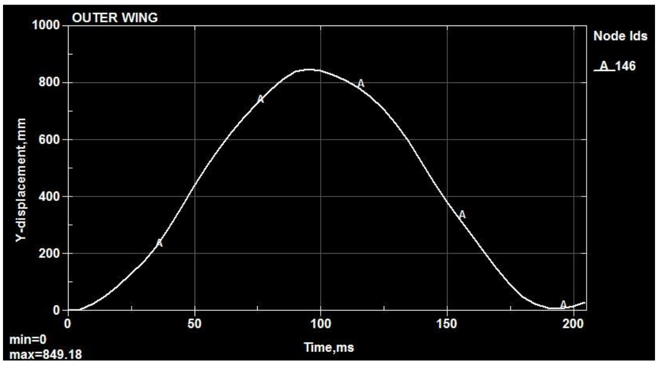

Fig. 7. Vertical displacement of the wing tip following the step loading of pressure.

Another dynamic event presented here is a quick vertical movement of the root of the wing. This has a form of the velocity pulse, as shown in Fig. 8, with v_0 = 5 m/s and t_0 = 97.8 ms, or one-half of the natural period. It is easy to calculate that the acceleration at the root is a = 10.4 g, while the maximum displacement of the root reaches 122 mm.

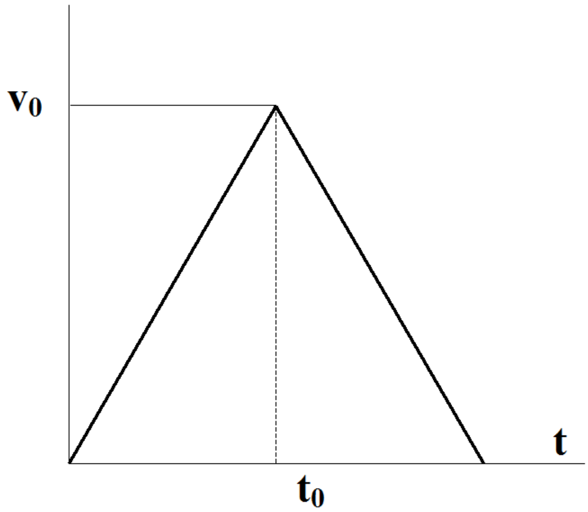

Fig. 8. Applied vertical displacement at the root section.

After the pulse was applied, the displacement response was as shown below and had a peak of 740 mm. This time the dynamic factor was DF(u) = 6.07. When reviewing the results, the reader should be aware that the two dynamic cases presented have different character. In the first the load was applied and increased everywhere by the same amount. This made the wing respond in a manner similar to that of an SDOF (single degree of freedom) system. In the second case there was no load applied to the wing at the beginning. Then, the enforced movement caused the stress wave to initiate at the root and propagate from there [7]. As Fig. 9 shows, it took some 30 milliseconds for the pulse to make itself felt at the wing tip.

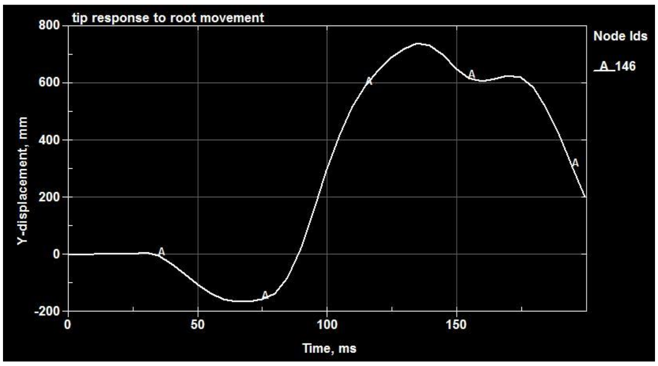

Fig. 9. Tip displacement caused by the root section movement.

5. SUMMARY AND DISCUSSION

The (reverse engineering) design of the wing of the Tu-154M aircraft described here can be viewed as the first step to determining the response of the wing as a whole. After the details of the external geometry were established, the structural elements were sized using hand calculations. This was the primary input into an FEA program, which helped to establish the stress levels associated with the limit load. The refinement of the design is planned in future work.

There was no mention of fasteners, i.e. rivets and bolts in reference to design of the wing. This is equivalent to assuming that fasteners are strong enough not to fail under the applied loads.

The analyses were carried out using the elastic material properties. Those will be modified when the true capacity of the structure needs to be assessed.

The wing was designed here to be as strong for down-bending as it is for up-bending and, as a result, the same stringers are used on the top and bottom surfaces. It is somewhat difficult to justify this for an airliner, which is not meant to be flying upside down. With the same gust forces, the up-bending will produce larger internal forces than down-bending because of gravity. The main justification for equal treatment of both surfaces lies in the simplicity of the design process.

In spite of the dynamic aspects of the system being incomplete (lack of the center wing or the elastic base of the outer wing) some analysis of the dynamic response was demonstrated. The extension of this work to various impacts, to which this wing may be subjected, will be carried out in the future.

APPENDIX

Since this work was submitted for publication and prior to its printing, a substantial progress was made. The inner wing (or the center wing) was included in the general model and the virtual test of the wing assembly was made. Then, a set of simulations of how this wing collides with the ground obstacles was performed. That obstacle was a 300 mm steel post impacted at its mid-height by the wing.

In the first test the post had 8 mm wall thickness and the point of contact was at 10,8 m from the plane of symmetry of the aircraft. The nose part of the profile was locally squashed, but the front longeron suffered only a minor denting. Next, the same 8 mm post was used to simulate the impact at a point closer to the wing tip; 13,4 m away from the plane of symmetry.

This time the front longeron was locally broken, but the wing comfortably survived. Then, a 12 mm post was used and a larger damage to the wing resulted. Finally, a 15 mm thick post has completely cut through the wing.

The 8 mm post mentioned above has some general similarity to the (400 mm diameter) birch, which was thought to be involved in the accident, but has its static strength at least 3.5x larger than the said tree.

It was also determined that cutting of the front longeron does not imply that the rest of the wing will break, at least not during the simulation and probably not for the next few hours of flight.

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