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CHARACTERISTIC AERODYNAMIC COEFFICIENTS AT
HIGH REYNOLDS NUMBERS

by

Jan Staszek

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CHARACTERISTIC AERODYNAMIC COEFFICIENTS AT HIGH REYNOLDS NUMBERS

Eng. Jan Staszek M.S.
Aviation Institute

Considerable changes in characteristic aerodynamic coefficients at high Reynolds numbers have been ascertained, in particular, in the transsonic range, and in the light of the latest wind tunnel tests and flight tests. Because of the generally accepted view that the effect of the Reynolds number exceeding the value $2 \cdot 10^6$ is rather small, these changes must be compared with the theoretical and experimental research presented in this article.

A change in aerodynamic coefficients with increasing Reynolds number is related to the structure and development of a boundary layer. These changes are rather advantageous, since the thickness of the boundary layer increases slower than the Reynolds number and as a result an increase in C_{zmax} was obtained with a simultaneous decrease in C_{xmin}, in particular in the presence of laminar airfoil sections.

However in the fifties the view was generally accepted that little could be expected in way of improvement of aerodynamic coefficients when the Reynolds number in wind tunnel tests is greater than the value $2 \cdot 10^6$, and that the agreement of these tests with the results of flight tests is sufficiently good for all practical purposes. American studies raised this limit to $6 \cdot 10^6$, however according to data available at that time the differences could not be great.

However Soviet tests demonstrated conclusively that changes in characteristic aerodynamic coefficients continue to occur with increasing Reynolds number. The 1 W 1012 airfoil section which was tested up to $Re=5.54\cdot10^6$ showed a further increase in the maximum aerodynamic lift C_z and changes in the coefficient C_m (Fig. 1), which suggested the necessity of conducting wind tunnel tests at higher Reynolds numbers. Subsequent studies have shown that this increase occurs even at $Re=8\cdot10^6$ and still amounts to about 7.5% (compared to C_z at $Re=5.54\cdot10^6$) with a further tendency toward an increase.

The effect of the tunnel scale on the obtained coefficients at increasing Reynolds numbers was also tested on that occasion. It turned out that the effect of the scale decreases commensurately with increasing Re, and that it is negligible and within the measurement error range at $Re=4.5\cdot10^6$. Even at smaller Re numbers differences occur at relatively large angles of attack, i.e. in ranges in which flow around the airfoil sections is, by the nature of things, more sensitive, when even minute deviations in performance cause great changes (Fig. 2).

Independently, further engineering progress in the construction of larger and faster aircraft revealed new phenomena and discrepancies with generally accepted views. The matter was further complicated since problems related to kinetic energy and viscosity of the air were compounded by problems involving air compressibility. Problems involving differences in estimates or measurement errors had to be approached more carefully and required greater accuracy. This was of fundamental importance, especially in the presence of more stringent requirements on the design for the purpose of obtaining the best possible cost and quality indicators during operation of the equipment (without delving on performance).

The C-5A aircraft, in which during wind tunnel tests the obtained critical Mach number was smaller by 0.02 than that obtained

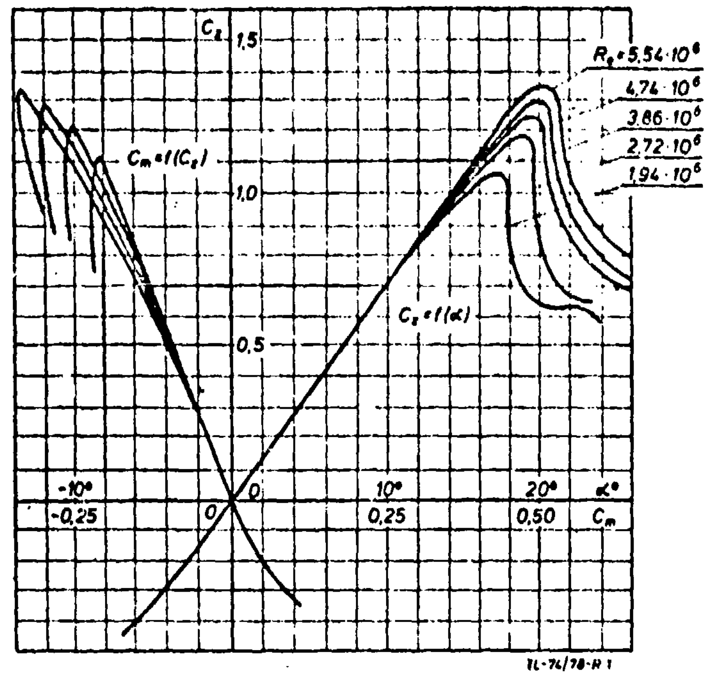

Fig. 1. Changes in C_z and C_m versus Reynolds number. CAGI IWIO airfoil section.

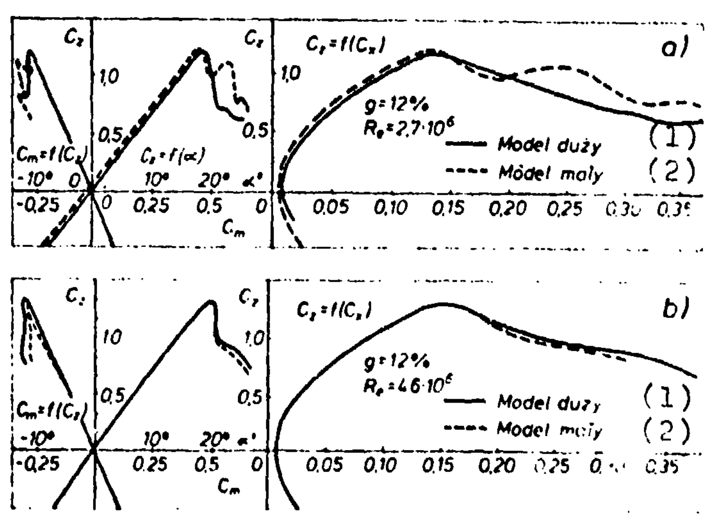

Fig. 2. Decreasing effect of scale with increasing Reynolds number. CAGI IWIO12 airfoil section.
Key: (1) Large model; (2) Small model

during flight tests (Fig. 3) can be mentioned as an example of such a situation. Since the aircraft was to be constructed in relatively large series, redesigning the wing (increasing the thickness of the airfoil section by 2%) turned out to be useful, which allowed reducing the weight of the airfoil by 3%. The fatigue properties were improved, the safety margin was increased, while all other performance characteristics remained unchanged (except the increase in payload lifting capacity). As a result of this the cost of the entire operation exceeded 900 million dollars and required the corresponding amount of time needed for the introduction of all necessary changes involved in initiating production.

Another example of errors in an estimation of extrapolated measurement (this time not associated with troubles) was the impossibility of determining correctly the drag in the rear part of the nacelle of a jet engine. Measurements performed in a wind tunnel up to $Re=18 \cdot 10^6$ and $Ma=0.9$ suggested a further increase in the drag coefficient while flight tests showed a distinct drop in the latter. Because of the relatively large cross section of the investigated fairing part of the nacelle and the lack of an estimate of the injector effect of the gas jet, the increased pressure in the rear part of the nacelle of the engine caused by interferences of the wind tunnel walls caused separation of the flow which completely distorted the results of measurements. Flight tests demonstrated the absence of flow separations and allowed correction of the drag coefficient which was decreasing commensurately with an increase in the Reynolds number.(Fig. 4).

Generally, the reason for the trouble is the lack of an estimate of the effect of the thickness and character of the boundary layer on the variation in aerodynamic coefficients. Obviously this layer is relatively thinner at higher Reynolds numbers, however the magnitude of the changes cannot always be foreseen in detail since these changes occur in the transsonic range in which the flow around the body is further complicated by the shock waves that are formed.

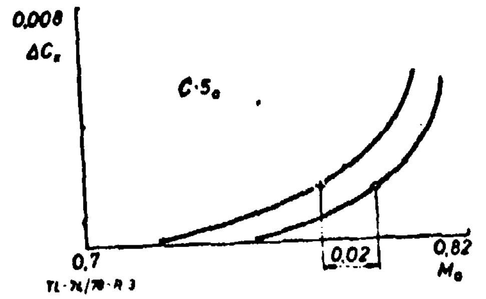

Fig. 3. Estimate of critical
Mach number from wind tunnel
tests: x - wind tunnel tests,
o - flight tests.

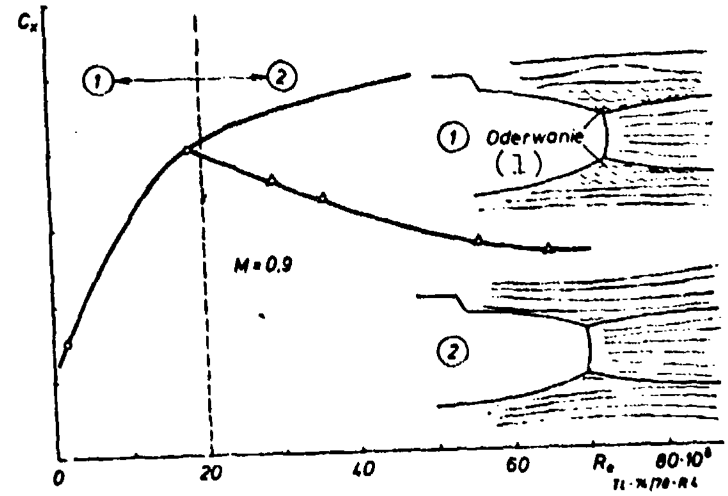

Fig. 4. Character of flow around and drag
coefficient of rear part of jet engine nacelle:
1 - wind tunnel test, 2 - flight test.
Key: (1) Flow separation

Trouble was revealed vividly during the construction of the C-141 aircraft, when the difference in the location of shock waves caused by the different thickness and shape of the boundary layer attained a value which exceeded by more than 20% the chord of the wing during a comparison with the results of flight tests at a considerably higher Reynolds number (Fig. 5). A different load distribution related to this fact caused an 11% change in the coefficient of the moment of the force (on the nose) which made it necessary to redesign the aircraft and caused a 9 month delay in the initiation of production (apart from the need to place over place over 180 kg balancing balast which lowered the aircraft's cost efficiency indicators).

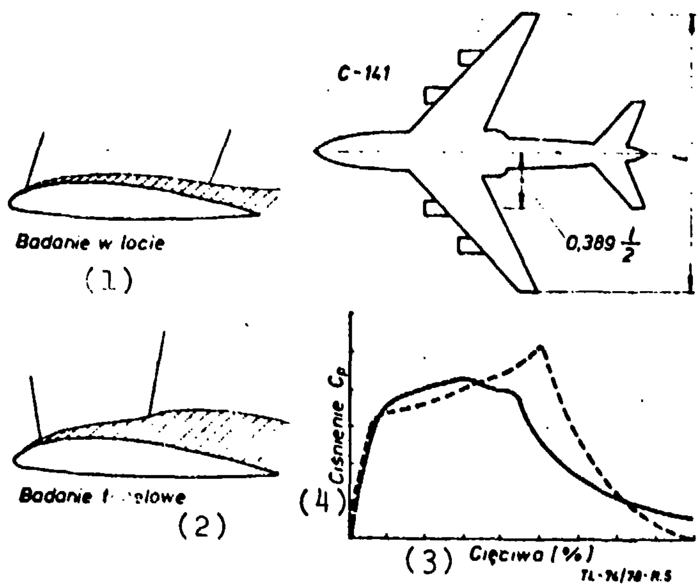

Fig. 5. Effect of character of boundary layer on location of shock wave.
Key: (1) Flight tests; (2) Tunnel tests; (3) Chord; (4) Pressure C_p

Attempts to determine the yawing moment of the aircraft M_y based on theoretical estimates of the location of the boundary layer and points at which a laminar boundary layer makes a transition to a turbulent boundary layer or on wind tunnel tests at smaller Reynolds numbers are in complete disagreement with reality (i.e. flight tests at high Reynolds numbers). Fig. 6 gives a comparison of data obtained using this method with the results of flight tests at real Reynolds numbers. The drawings encompass the results of calculations of the coefficient C_m for two wing sections at the distance 0.389 L/2 and 0.637 L/2 from the axis of symmetry of the aircraft (L denotes the wingspan). The results of the wind tunnel tests differ distinctly from calculated data both in regard to the values of coefficients and the shape of curves. The results of flight tests at high Reynolds numbers on the order of magnitude $100 \cdot 10^6$ deviate considerably from both magnitudes determined earlier. All data were obtained for the same Mach number and the same aerodynamic lift Ma=0.825, C_z=0.4.

More detailed data about the character of the turbulent boundary layer and its interaction with shock waves are needed during the design of aircraft. Most reliable information can be obtained by means of systematic investigations in the entire range of Reynolds numbers (from small to very large Reynolds numbers). In particular, this applies to the effect of the Reynolds number on the point at which the shock wave is formed, the location and the supersonic range region and the separation of streams near the trailing edge. Theoretical studies allow a prediction of changes in the location of the shock wave (Fig. 7), however these data must be supported and verified by wind tunnel tests and validated by flight tests. At any rate the designer must have at his disposal up-to-date data on phenomena taking place in the range of Reynolds numbers of interest to him.

The problem is so complicated that the shock wave does not always propagate to the rear commensurately with an increase in the Reynolds number. If, in addition, the above mentioned problems also involve aeroelasticity phomenona we may be dealing with a shock wave propagating to the front as shown in Fig. 8. Aeroelastic deformation

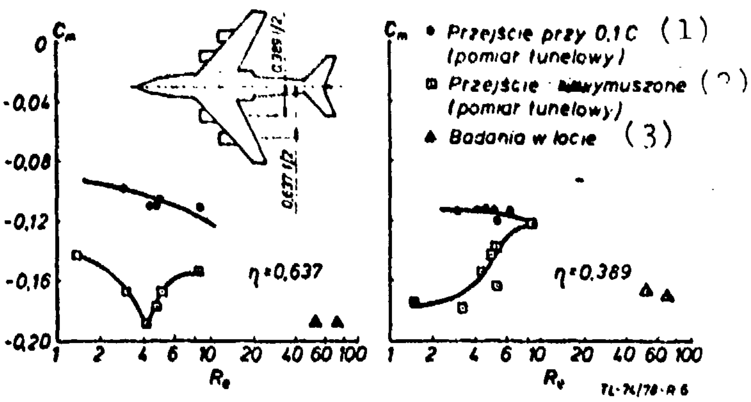

Fig. 6. A comparison of measurements of coefficient C_m from wind tunnel tests and during flight (Ma=0.825, C_z=0.4).
Key: (1) Transition at 0.1C (Tunnel measurement); (2) Transition not forced (Tunnel measurement); (3) Flight tests

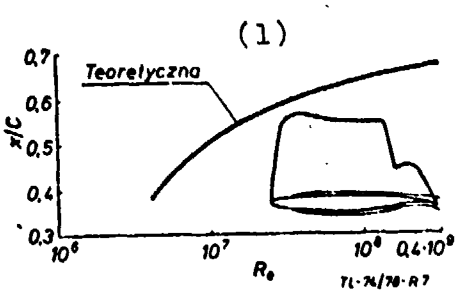

Fig. 7. Effect of Reynolds number on location of shock wave of supercritical airfoil section at Ma=0.8.
Key: (1) Theoretical

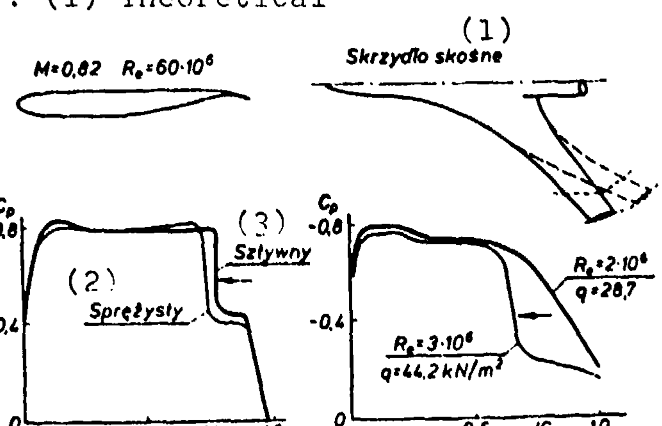

Fig. 8. Effect of elastic deformation displacement of shock wave.
Key: (1) Backswept wing; (2) Elastic; (3) Rigid

of the model of a supercritical airfoil section with a flexible trailing edge displaces distinctly the shock wave (compared with a rigid model) as shown in the left part of the diagram. In regard to the entire wing, an increase in the dynamic pressure corresdonding to a relatively small increase in Reynolds number (from $2 \cdot 10^6$ to $3 \cdot 10^6$) caused an elastic deformation of the wing resulting in a considerable displacement of the shock wave to the front which automatically changed the equilibrium conditions of the aircraft since a propagation of the shock wave to the rear was anticipated.

Wings with a supercritical airfoil section are much more sensitive to the Reynolds number than those used earlier, because the pressure gradients having an effect on the boundary layer are smaller. This problem was revealed by the results of two dimensional tests of a supercritical airfoil section which were compared with theoretical calculations. As a result of the analyses a technique was elaborated for approximate simulation of characteristics at the real Reynolds number for conditions which approximated actual conditions using LRC (Langley Research Center) wind tunnels. The transition line initially located near the leading edge of the wing is shifted artificially to the rear in such a way that the relative thickness of the boundary layer on the trailing edge of the wing is the same as that which can be expected under real conditions in the presence of a transition line located near the leading edge. Wind tunnel tests demonstrated that such simulation technique ensures very good agreement with the characteristics of the airfoil section at real Reynolds numbers.

Fig. 9 shows the change in the drag coefficient C_x as a function of the aerodynamic lift C_z for a wing with a supercritical airfoil section designed for a real Reynolds number at Ma=0.78 and a Reynolds number of the chord of the model equal to $2.26 \cdot 10^6$. The results are presented for conditions in which the transition line is located at a distance from the leading edge equal to 10% and 35% of the chord respectively. The calculations imply that a transition at 35% of the chord simulates approximately conditions under which a boundary layer exits in reality. From a comparison it can be seen that for the range of the coefficient C_z (which is close to that used in practice, i.e. about 0.6) the drag coefficient, when the transition is located near the rear, is smaller by about 0.005 than when it is located near the front. This difference is considerably greater than that which follows from the usual decrease in surface drag accompanying the location of a transition in the rear.

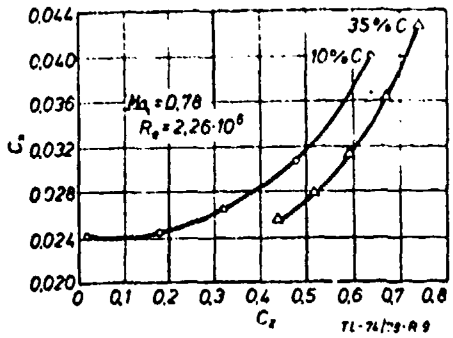

Fig. 9. Effect of point of transition of laminar boundary layer to turbulent boundary layer on airfoil section drag.

The importance of the part played by the viscosity of the medium can be clearly seen from the drawings in Fig. 10. The drawings represent the pressure distributions on a supercritical airfoil section at Ma=0.73 for three Reynolds numbers: $6 \cdot 10^6$; $40 \cdot 10^6$ and $400 \cdot 10^6$. It should be emphasized that at Reynolds number $6 \cdot 10^6$ which was considered as the limiting number beyond which changes no longer occur, the coefficients C_z and C_x differ basically from the values obtained for these coefficients at $Re=400 \cdot 10^6$. The increase in C_z is nearly 40%, whereas the decrease in C_x exceeds 43%. Calculations by the Korn-Garabedian medhod have shown that an increase in the coefficient C_z in the case of absence of viscosity may be as high as 93% for this airfoil section. The calculations were made according to a program for the analysis of transsonic flow taking into account the shift in the point at which the laminar boundary layer makes a transition to a turbulent boundary layer with a determination of the drag by the Nash and MacDonald method for a turbulent flow.

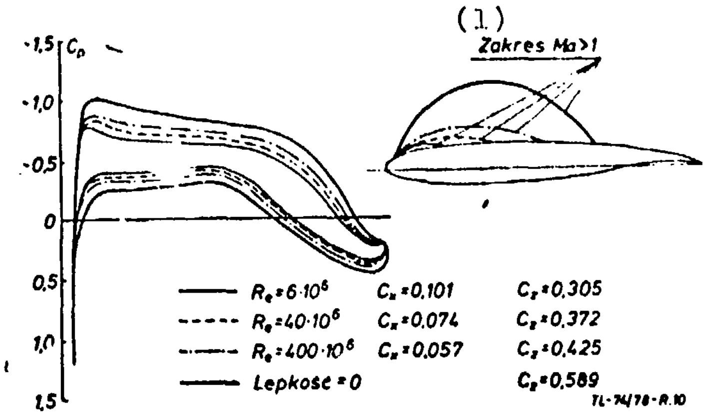

Fig. 10. Effect of Reynolds number on C_z and C_x of supercritical profile at Ma=0.73.
Key: Mach number range > 1

In the case of a supersonic range sufficiently effective to induce shock waves (for example Ma=0.759 in Fig. 11) an increase in Reynolds number automatically brings about changes in the pressure distribution and a shift of the shock wave, which entails the necessity of wing loading and balancing the aircraft. Clearly this matter is very important for the designer and it may cause a great deal of trouble.

The greatest difficulty during tests is correct prediction of the separation of the flow around the body. Generally, separation may occur near the leading edge or at the point at which the shock wave is formed if the flow about the body is transsonic. Each of these types of separations depend to a great degree on the Reynolds number, in particular the first type, because the character of the separation occurring on the leading edge depends to a great degree on the manner in which the laminar flow changes to a turbulent flow. The phenomenon is still not well understood and sufficiently analyzed as, for example, the sudden occurrence of separation pockets on the leading edge.

Separation of the flow around the body can also begin at the trailing edge and shift to the front commensurately with increasing angle of attack as long as the entire upper surface of the wing is not encompassed by it. Methods have been elaborated for calculating such flow around the body; however each time they must be confirmed by experiments. Criteria exist for the transition of a laminar flow to a turbulent flow which were partially successful in practice, however they are useless for supercritical airfoil sections at transsonic velocities. Fig. 12 presents changes in the pressure distribution, drag and aerodynamic lift at Reynolds number $6 \cdot 10^6$ for the airfoil section in Fig. 10 depending on the point of transition of the laminar boundary layer to a turbulent boundary layer. The region of changes of the transition point encompass the zone from the leading edge to 30% of the chord from it.

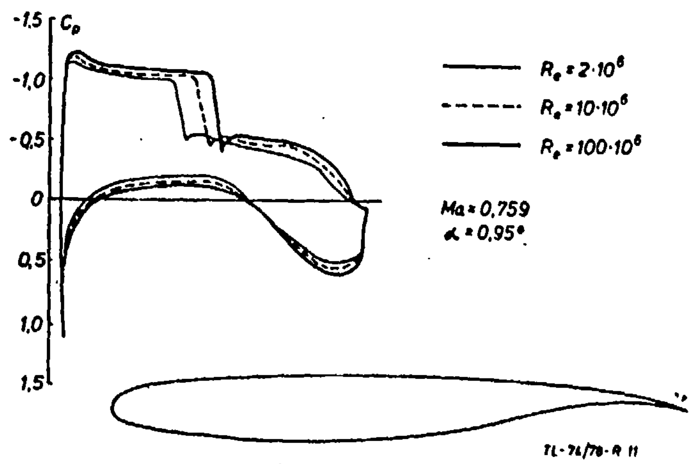

Fig. 11. Pressure distribution for typical supercritical airfoil section at various Reynolds numbers, the same Mach number and the same angle of attack α.

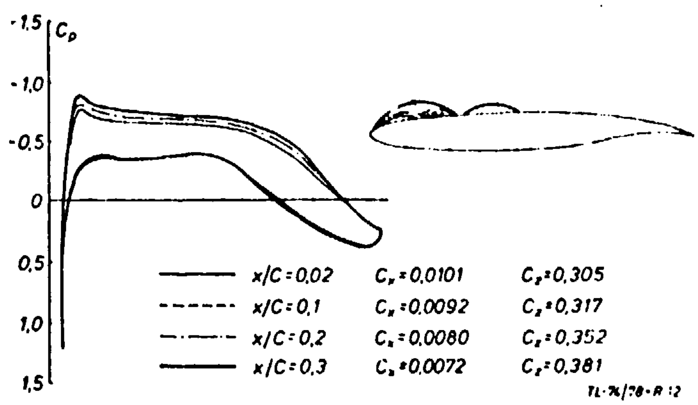

Fig. 12. Effect of position x/c of transition point of laminar boundary layer to turbulent boundary layer on C_z and C_x of supercritical airfoil section at Ma=0.73 and Re=$6\cdot10^6$.

A point worth noting is that when the transition point is located at a distance equal to 30% of the chord from the leading edge, the coefficients C_z=0.381 and C_x=0.0072 are very close to the values obtained at Reynolds number $40\cdot10^6$ shown in Fig. 10. From this fact the conclusion can be drawn that forcing of the point of transition of the laminar boundary layer to a turbulent boundary layer in wind tunnels with a small Reynolds number may provide representation of the flow around a body at high Reynolds numbers, since both natural and forced transitions should give a similar boundary layer. However even when a suitably forced location of a transition gives a good approximation of the thickness and character of the developing boundary layer and consequently also of the aerodynamic lift, we are by no means certain that the velocity distribution and consequently also the drag are also well represented. This problem must be investigated experimentally independently of the theoretical analysis.

According to available data on the effects of Reynolds numbers on the results of wind tunnel studies changes in the mean drag coefficient of a medium airliner were recalculated as a function of Re in the range up to $100 \cdot 10^6$. The results of this recalculation are presented in Fig. 13; the decrease in the drag coefficient C_x at $Re=100 \cdot 10^6$ to less than half its value at $Re=1 \cdot 10^6$ should be emphasized.

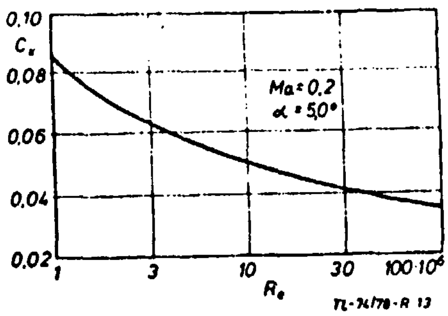

Fig. 13. Change in drag coefficient of medium airliner with increasing Reynolds number.

Present day American wind tunnels allow to obtain Reynolds numbers barely to about $30 \cdot 10^6$. The above mentioned data were obtained from flight tests which were compared very carefully with theoretical studies. The obtained Reynolds numbers, however, did not exceed $100 \cdot 10^6$; on the other hand they revealed many phenomena requiring closer study. Theoretical calculations and methods simulating flow in the presence of similar boundary layer proportions can only be used as an indicator of what may happen during flight at real high Reynolds numbers. Lately wind tunnels based on the technique of low gas temperatures in a wind tunnel (cryogenic technology) have been built in the United States. The model 0.3 meter nitrogen tunnel operating at temperatures on the order of 80 to 100 K, shows results which were so promising that the construction of a large wind tunnel was undertaken in which this concept was applied. Because the viscosity coefficient decreases substantially with a decrease in the temperature, a decision was made to also fill the new closed cycle wind tunnel with nitrogen and maintain a low temperature by vaporizing liquid nitrogen injected below the measurement space. The tunnel is propelled by a conventional fan and operates at barometric pressure. The Reynolds numbers obtained in the channel (on the order of magnitude 80 to $100 \cdot 10^6$) do not encompass entirely the envisioned range to be obtained in flight tests, however they allow coming closer to the unknown phenomena.

The problems that were touched on represent only incomplete and general information about their existence requiring deeper analysis and broader research. The problems indicate that in the range of high Reynolds numbers aerodynamics has still a great deal to learn, however understanding of this field by no means will be easy.

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