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Reduction of Fuselage Form Drag by Vortex Flows

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Wind-tunnel tests were performed in the California Institute of Technology's 10-ft-diam tunnel in a parametric study of 1:17 scale models of fuselages of Boeing 747 and Lockheed C-5 aircraft. The purpose was to validate the theoretical model advanced in 1981, that pairs of vortex generators would reduce the form drag of transport aircraft. The longitudinal vortices generated by flat-plate lifting planforms in the lower aft regions of the fuselage prevent, or delay, separation by energizing the boundary layer and also by creating a strong transverse outflow from the plane of symmetry. The search for the minimum drag involved three vortex generator configurations, with three sizes of each in six locations, clustered in the lower aft regions of the fuselages at the beginning of the tail upsweep. The local Reynolds number that referred to the surface distance from the nose was about $\sim 10^7$. Fully developed turbulent flow was confirmed by noting insignificant drag differences between test runs with and without upstream boundary-layer tripping strips. Vortex generator planforms ranged from swept tapered, through swept straight, to swept reverse tapered wings, whose semispans ranged from ~ 50 to 125% of the estimated local boundary-layer thickness. Inboard actuators under the control of an externally mounted proportional digital radio controller varied the pitch angles of the vortex generators. While certain combinations of vortex generator parameters increased drag, other configurations, locations, and pitch angles of vortex generators reduced drag by 3% for the 747 and ~ 6% for the C-5, thus confirming the arguments that effectiveness increases with the angle of upsweep of the tail. Therefore, the greatest gains in performance are expected on aft-loading military transports. Incremental changes in overall aircraft drag and fuselage drag are approximately in the proportion 1:3. Overall drag reductions are expected to be ~ 1% for the 747 and 2% for the C-5 aircraft.

Introduction

TRANSPORT aircraft generally follow the form envisioned by Sir George Cayley almost 200 years ago, in having a wing, a fuselage, and an empennage. The volume of the fuselage is determined by cargo or passenger load requirements, with the overall length being fixed by a compromise between operational considerations and aerodynamic efficiency. In general, the aft ends of transport aircraft fuselages are tapered asymmetrically, with pronounced upsweep of the lower contour, to some minimum base area. The upsweep of the fuselage contour is exemplified by the Boeing 747, but it is much more pronounced in aft-loading aircraft such as and the Lockheed C-5. Profiles of these aircrafts' fuselages are shown in Fig. 1. Adverse longitudinal pressure gradients, caused by fuselage upsweep, result in rapid boundary-layer growth on the bottom of the fuselage, and eventually, separation of flow. A large volume of low-energy flow over the upsweep region leads to a high-momentum defect in the wake, and an increase in fuselage form drag.

Aircraft designs are being continuously refined to enhance their performance and operating economy. When the basic performance and operating requirements are defined, the configurations of the basic components are essentially fixed, and only minor shape modifications or additions of appendages are allowed. One of the more attractive approaches is the utilization of vortices to energize the boundary layer and to modify the outer inviscid flow. Attention has generally focused on aircraft wings. Pearcey[1] gives an excellent survey of methods for energizing boundary layers to delay or prevent flow separation. An example of boundary-layer control by means of relatively small vortices is the location of vortex generator arrays on Boeing 707 aircraft. Large-scale vortices on fighter aircraft are generated by means of canards, chines, leading-edge extensions, or leading-edge discontinuities. These can be seen on virtually all recent aircraft. Small vanes on the forward parts of the projecting DC-10 engine nacelles are yet another example of the utilization of vortex generators to control the boundary layers on aircraft wings.

Aircraft fuselages have received less attention, although there is some awareness of the possibilities of drag reduction and flow stabilization. An example of that is the replacement of the tail cone of the McDonnell Douglas MD-80 by an improved aerodynamic shape, which, according to *Aviation Week*,[2] reduces cruise drag by ~0.5%. On the bottom regions of highly upswept tails of aft-loading transport aircraft, the flow instabilities caused by flow separation and vortex shedding have been alleviated through the use of sharp discontinuities at the beginning of upsweeps and longitudinal strips. These generally incurred increased drag losses.

The present concept is a new approach to the reduction of drag of fuselages with upswept tails. It is known that the boundary layer on the bottom of an upswept part of the fuselage is highly three dimensional because of the strong retarding axial pressure gradient and inflow toward the vertical plane of symmetry. Wortman and Franks[3] showed in a comprehensive parametric study of three-dimensional flows that relatively small transverse outflow from the plane of symmetry delayed boundary-layer separation. Therefore, rather than merely energizing the boundary layer with an array of small vortex generators, it is much more advantageous to generate two large longitudinal vortices that induce transverse outflow at the plane of symmetry on the lower part of the fuselage. These not only energize the boundary layer, but also generate strong favorable transverse pressure gradients that move flow separation downstream, or eliminate it entirely. The concept was expressed in the form of a pair of vortex generators near the beginning of the upsweep of the lower contour of the fuselage. A proposal submitted in 1981 to the U.S. Air Force DESAT program,[4] although described as excellent by the Flight Dynamics Laboratory, was not supported. An attempt to reduce the aft fuselage drag was made later by the U.S. Air Force Flight Dynamics Laboratory.[5] However, this effort used pairs of vortex generators similar to those found on the vortex generator arrays on wings, and, thus, missed the significance of induced transverse outflow from the plane of symmetry. This paper describes the theoretical and experimental studies performed under U.S. Department of Energy Grant DE-FG01-86CE15277, which were summarized in a project report[6] that had very limited distribution. The invention was granted a U.S. patent[7] in 1991.

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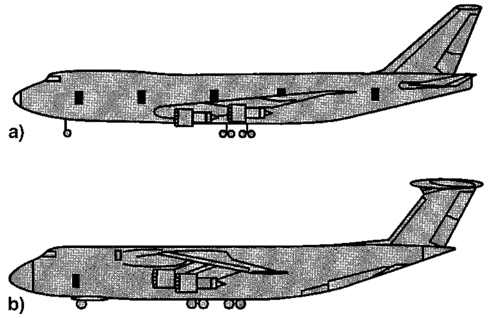


Fig. 1 Transport aircraft fuselages: a) Boeing 747 and b) Lockheed C-5.

Aerodynamic Estimates

Assessment of the significance of reductions of fuselage drag on the total aircraft drag involves difficult estimates of relative contributions of various components of drag. These are not only functions of the aircraft configuration, but also vary with the modes of operation. Torenbeek[8] presents drag polars for a representative transport aircraft. At the reference point of minimum total drag, the principal contributions to profile drag are identified as wings (39%), fuselage (32%), engine installation (14%), and tailplanes (15%). The vortex drag contribution to the total aerodynamic resistance is equal to ~18% of the profile drag. The influence on the total drag C_D of the fuselage drag $C_{D,F}$ is easily determined by logarithmic differentiation to be

$$\frac{\mathrm{d}C_D}{C_D} = F^{-1}\frac{\mathrm{d}C_{D,F}}{C_{D,F}} \tag{1}$$

The magnitude of the influence coefficient $F =$ (total drag/fuselage drag) is about 3.7. Thus, for this particular example, the reduction in overall drag is about 27% of the reduction in the fuselage drag.

Preliminary estimates of the required sizes of vortex generators must be derived from order-of-magnitude estimates because the theory for the interaction of longitudinal vortices with three-dimensional decelerating boundary layers contains too many uncertainties to permit precise calculations. Here, estimates are for $M = 0.82$ cruise at 40,000 ft, at a location $L = 160$ ft from the aircraft nose, and an effective fuselage diameter D at that station of 25 ft. The Reynolds number, based on the longitudinal distance, is $\sim 250 \times 10^6$. While realizing that the boundary layers under consideration are three-dimensional practical considerations dictate the use of two-dimensional relations given by Schlichting,[9] for first order of magnitude estimates it is estimated that the boundary-layer thickness $\delta = 1.2$ ft (36 cm), the displacement thickness $\delta^* = 0.12$ ft, and the momentum thickness $\delta^{**} = 0.11$ ft. The momentum thickness represents the momentum defect caused by viscous dissipation in the boundary layer, so that the total defect of momentum M_D near the tail is

$$M_D = \pi D \delta^{**} \rho U^2 \tag{2}$$

where ρ is the density and U is the local velocity. Aircraft vortex flows, particularly those generated by strakes and wing leading-edge extensions, were studied extensively by Wortman,[10] in response to questions regarding flow visualization. Attention was focused on vortex structure, interactions with viscid–inviscid flows, and Reynolds number effects. The study showed that intense, high Reynolds number vortices have very high axial flow velocities that do not exist in laminar vortices. The study presented an example that was taken from Jones,[11] who showed experimental data for delta wing vortex flows at a Reynolds number of 3×10^6, in which the ratio of axial to local velocity was 3. The velocity ratio was only 1 at a Reynolds number of 1×10^4, which is typical of small water-tunnel flow visualization tests. Awareness of these data necessitated large wind-tunnel models with high Reynolds numbers and fully developed turbulent boundary layers.

It is known that the diameter of the vortex at the trailing edge of a delta planform is approximately equal to the half span. Here, the flat-plate vortex generators operate at high angles of attack and produce nonlinear lift. If the shed vortex core diameter is taken as approximately equal to the chord c, so that the axial momentum flow in the vortex, M_A, is

$$M_A = \rho Z^2 U^2 \pi c^2/4 \tag{3}$$

where Z is the ratio of average axial velocity to the local freestream velocity. [Because the momentum thickness δ^{**} is given approximately by

$$\delta^{**} = 0.03 L Re_L^{-0.2} \tag{4}$$

where Re_L is the Reynolds number referring to the boundary-layer running length L at the location of the vortex generators.] The ratio of momenta is

$$M_A/M_D = (c^2/LD)\left(Z^2 Re_L^{0.2}\big/0.12\right) \tag{5}$$

For the range of conditions under consideration, the magnitude of $(Z^2 Re_L^{0.2}/0.12)$ is 3000–4000. Assuming $M_A/M_D = 1$, a reasonable estimate is

$$c^2/LD = 1/3{,}500 \tag{6}$$

For the vortex generators used in this program, the ratio of the height or semispan h to the mean chord c is ~2. If the fuselage is treated as a mirror plane, then the effective aspect ratio $A = 2\ h/c$ is ~4. Therefore,

$$(h/D)^2 = (A^2/4)[(L/D)/3500] \tag{7}$$

and $h = 0.0.086 \times D = 2$ ft. This is most likely a conservative estimate because it does not include the effect of the reduction of the inflow by the vortex tangential velocity.

These simple estimates indicate that a pair of 2-ft-high fins would be quite adequate to achieve significant drag reductions. Actually, wind-tunnel studies demonstrated that smaller fins could achieve significant drag reductions. A limited number of tests with four vortex generators did not show any gains, so that reductions in size are likely to be achieved with different configurations than those tested here.

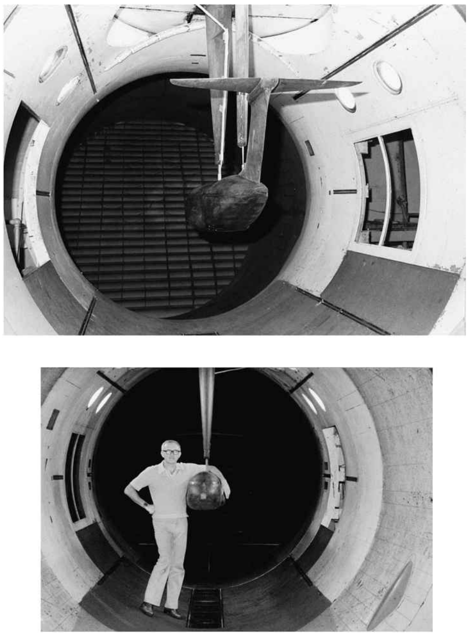
Experimental Program

The wind-tunnel experimental study was a parametric search for the optimum configuration of vortex generators, locations, and pitch angles. The principal parameters were 1) vortex generator configurations, 2) vortex generator size, 3) location of the vortex generator, and 4) pitch angle of the vortex generators. With three configurations in three sizes and six locations, the total number of runs could be 54 for each model, and each run contained at least five pitch angles.

Models

The models were ~1:17 scale replicas of the fuselages and empennages of Boeing 747 and Lockheed C-5A transports. Hollow, layered, clear pine wood construction was employed and structural stiffness of the ~13-ft long (4-m) hulls was augmented with steel channel section beams. The complete models mounted in the test section of the California Institute of Technology's 10-ft wind tunnel are shown in Fig. 2, a) showing the supports and b) illustrating size and test section blockage.

Budgetary constraints and practical considerations led to the decision to only test models of fuselages and empennages. From the vortex studies cited in the preceding text, it was known that to obtain representative vortex structures, the Reynolds numbers would have to be very high. This necessitated large fuselages that yielded Reynolds numbers of 10–15 $\times 10^6$ at the locations of the vortex generators. It was not practical to attach wings to the models at that scale of the fuselage. If a significantly smaller scale had been adopted, then, in view of the conclusions in Ref. 10, the relationship between wind-tunnel tests and full-scale results would be questionable.



a)

b)

Fig. 2 Models mounted in the wind tunnel a) details of supports (B-747), and b) cross section with C-5 model.

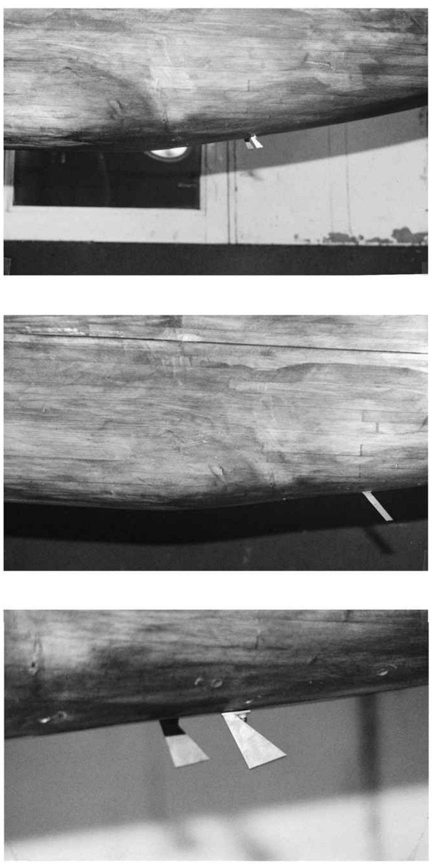


Fig. 3 Installation of vortex generators.

Actually, the results for models without wings should be very conservative because, in an actual configuration, the downwash from the wings will tend to mitigate the adverse pressure gradient. Without wings on the models that were tested, the adverse pressure gradient was greater, and thus, a greater demand was placed on the effectiveness of the vortex generators.

Details of the vortex generators installed on the B-747 model are shown in Fig. 3. The relative size of vortex generators may be

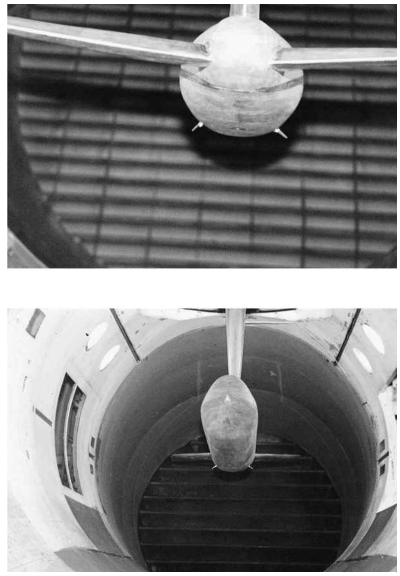
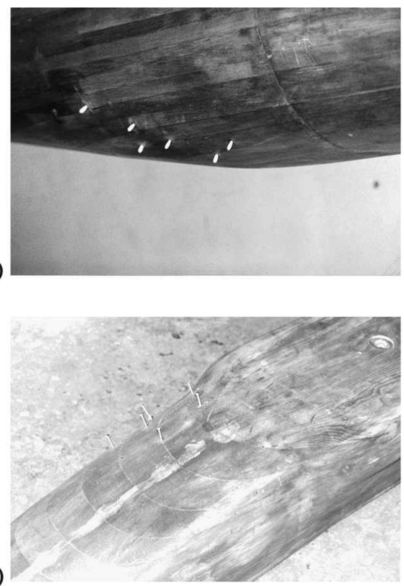


Fig. 4 Relative sizes of vortex generators.



a)

b)

Fig. 5 Locations of vortex generators on the a) B-747 and b) C-5.

gauged from the views of the models installed in the wind that are shown in Fig. 4. Locations of the vortex generators were determined by estimates of trajectories of the streamlines into the separated flow regions. Essentially, the selected locations represent subjective judgment that can only be verified in comprehensive parametric test programs. In view of this, locations of the vortex generators were not expressed in fuselage coordinates. Details of locations are shown in Fig. 5. The identifying numbers were assigned in ascending order in the aft direction from the most forward locations.

Vortex Generators

Three configurations of vortex generators, in three sizes each, were tested. These are shown in Fig. 6 with defining geometrical data and identifications of height and mean chord. The vortex generators were made of a 0.06-in. (1.5-mm) brass sheet, and were brazed to base plates and brass tube axles attached to the internal actuating mechanisms.

Controls and Actuators

Control of the pitch angles of the vortex generators was by means of a Futaba proportional digital radio transmitter located outside the

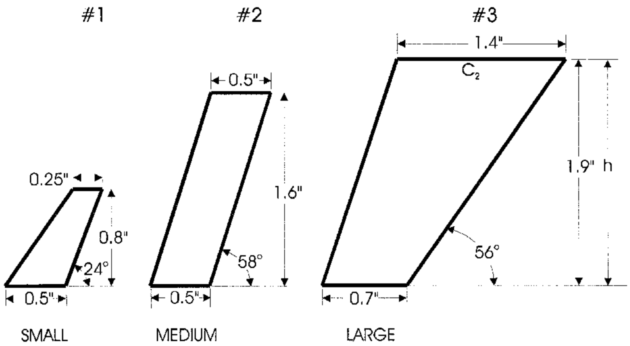


Fig. 6 Shapes and sizes of vortex generators with definitions of heights and mean chords. h = height, semispan; $c = (c_1 + c_2)2$, mean chord.

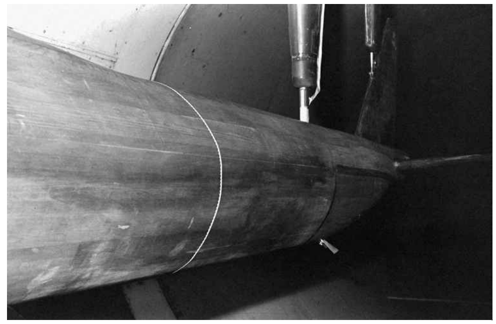


Fig. 7 Boundary-layer tripping strip.

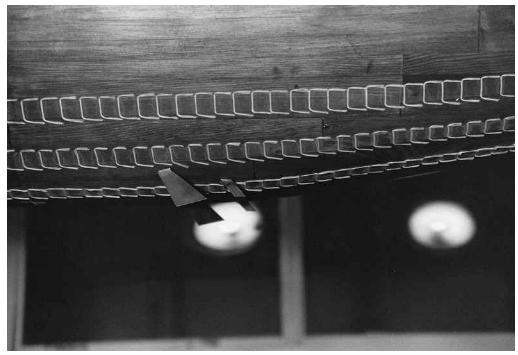


Fig. 8 Flow visualization tufts.

wind tunnel, and a receiver mounted on the internal wind-tunnel wall. Vortex generator pitch angles were varied by actuators operating rod and ball crank mechanisms. All actuating mechanisms were in removable lower fuselage sections, so that adjustments and repairs could be made without interfering with the mounting of the models in the wind tunnel.

Wind Tunnel

The tests were conducted at the California Institute of Technology recirculating flow 10-ft Wind Tunnel Facility, capable of speeds up to 250 ft/s. The facility has a six-degree-of-freedom model support system, and the balances have a resolution of 1 drag count at a level of reading of ~2000 drag counts, that were representative of readings in these tests. The accuracy was stated by tunnel personnel to be 0.1–0.2%. Repeatability was within 0.06% on consecutive tests, and within 0.1% for tests separated by several days. Because only relative magnitudes were of interest here, the drag counts were not converted into magnitudes of drag. On each test, the balance system was allowed to stabilize until variations in the readings ceased. In these tests, only lift and drag data were recorded, but pitching moments were checked frequently. The effects of pitch angles of the vortex generators on fuselage pitching moments were insignificant.

Test Aerodynamics

The Boeing 747 model was tested at a dynamic pressure of 30 psf because of some concerns related to possible aeroelastic problems. The much more rigid C-5 model was tested at 60 psf. The corresponding velocities are ~160 and 225 ft/s, respectively. At the location of the vortex generators, the Reynolds numbers that referred to the axial distance from the nose were 9.5×10^6 for the Boeing 747 and 11.2×10^6 for the C-5. To remove any doubts regarding the state of the boundary layer at the location of the vortex generators, a boundary-layer tripping strip, shown in Fig. 7, was used in some tests. No differences in drag could be detected between a dozen tests with and without the trip. The flow was therefore determined to be a fully turbulent boundary layer.

Procedure

After a pair of vortex generators was installed on the model, their pitch angles were set to an estimated +10 deg to the estimated local streamline, and the controller was set to 0. Positive pitch was defined when the leading edge of the vortex generator was inboard of the estimated local streamline. Calibration tests established that a 25% change in the controller setting resulted in a 10-deg change in pitch angles. With the exception of some check runs for repeatability and sensitivity around the best results, the pitch angles were changed in increments of 25% of the controller scale until at 100% the estimated pitch angle was −30 deg.

Flow visualization was performed by means of arrays of tufts, such as those shown in Fig. 8. The fluttering of the tufts near the bottom of the fuselage was reduced noticeably when vortex generators were rotated to their optimum pitch angles. These observations indicate that while even if there is no flow separation, the boundary layer on the upswept fuselage bottom is highly retarded and unsteady. Reduction or elimination of this unsteadiness by the vortex generators indicates that the concept of imposing longitudinal and transverse velocities on the boundary layer is an effective method for modification of flow.

For the Boeing 747 model, the boundary-layer overall, displacement, and momentum thicknesses at the location of the vortex generators are estimated to be 1.6 in. (4 cm), 0.2 in. (5 mm), and 0.15 in. (4 mm), respectively. For the C-5 model, the boundary-layer thicknesses are estimated to be 80% of the preceding values. Vortex generator heights ranged from $\sim\frac{3}{4}$ to 2 in. Even for the smallest vortex generators, the flow velocity at their outer tips was at over 85% of the local inviscid flow velocity. The vortices were therefore generated by flows with relatively high dynamic pressures.

Results

Wind-tunnel data that are summarized in Appendix C of the project final report (Ref. 6) are not shown here because of limitations of space. Selected values of the best configurations are shown in Tables 1 and 2. Lift and pitching moments, which are of secondary interest here, are not included because they were not influenced perceptibly by vortex generators. The data indicate that a 3% reduction in fuselage drag can be achieved for the Boeing 747, and almost double that reduction can be realized for the Lockheed C-5. These values do not necessarily represent the absolute maxima because of the limited number of configurations, sizes, locations, and pitch angles that could be tested. It is quite likely that greater drag reductions could have been found, but at the expense of additional locations and vortex generator configurations. The additional wind-tunnel test time would have exceeded the fiscal means of the present project.

The problem of finding optimum vortex generators is quite formidable, but not beyond the reach of a reasonable engineering effort. The data in Tables 1 and 2 show that comparable drag reductions can be achieved by different vortex generators in different locations. When optimum pitch angles were sought during the test by making small pitch changes around the apparent optimum value, it was found that the sensitivity was quite low. This reduces greatly the efforts to find optimum pitch angles.

Table 1 Selected B-747 drag data

Run no.	Vortex generator no.	Size	Station no.	Vortex generator %	Drag counts	% Change
1	——	——	——	——	2234	——
10[a]	3	L[b]	1	0	2169	−2.91
22	3	M[c]	4	0	2186	−2.15
29	1	S[d]	2	0	2185	−2.19
32	2	S	2	0	2166	−3.04
35	3	S	2	100	2177	−2.55
41	1	S	4	25	2177	−2.55
47	1	S	6	100	2183	−2.28
52	3	S	6	75	2183	−2.28
58	1	S	2	50	2181	−2.37
64[a]	3	L	1	50	2173	−2.73

[a]Duplicate runs 5 days apart. [b]Large. [c]Medium. [d]Small.

Table 2 Selected C-5 drag data

Run no.	Vortex generator no.	Size	Station no.	Vortex generator %	Drag counts	% Change
66	——	——	——	0	1812	——
80[a]	1	L[b]	1	0	1766	−2.54
81[a]	1	L	1	0	1768	−2.43
82	3	S[c]	1	0	1761	−2.81
83	3	M[d]	1	0	1737	−4.14
84	3	L	1	0	1735	−4.25
85	3	S	3	0	1755	−3.15
86	3	M	3	0	1749	−3.48
87	3	L	3	0	1743	−3.81
89	2	M	4	0	1735	−4.25
92	3	S	5	0	1738	−4.08
93	3	M	5	0	1736	−4.19
94	3	L	5	0	1744	−3.75
95	2	S	5	0	1743	−3.81
96	2	M	5	0	1710	−5.63
97	2	L	5	0	1733	−4.36
98	1	M	5	0	1734	−4.30

[a]Duplicate runs to check consistency. [b]Large. [c]Small. [d]Medium.

Full-Scale Implementation

Scaling of wind-tunnel results to full-scale actual aircraft applications can be accomplished in several ways. If purely geometrical scaling is adopted, then the height or semispan of the vortex generators would be ~2 ft (61 cm). Scaling on the ratio of vortex momentum to the boundary-layer momentum defect requires that

$$\left(A_v \big/ D\delta^{**}\right) = \text{const} \tag{8}$$

where A_v is the vortex generator area. In terms of Reynolds number,

$$Re_L^{0.2} A_v \big/ LD = \text{const} \tag{9}$$

With wind-tunnel $Re_L = 10^7$ and full-scale $Re_L = 25 \times 10^7$, the relation for scaling is

$$(A_v/LD)_{\text{Prototype}} = 0.525(A_v/LD)_{\text{Model}} \tag{10}$$

This leads to the conclusion that semispan of the vortex generators on full-size aircraft should be the square root of 0.525, or 0.725, of their relative size on the wind-tunnel models. Several other approaches to translating wind-tunnel vortex generator sizes to full-scale applications are possible, but they all lead to the conclusion that on Boeing 747 and Lockheed C-5 aircraft, the height of the vortex generators should not exceed 2 ft. An appreciation of the relatively small sizes that are required may be gained from the consideration that the Boeing 747 is ~220 ft long, and in the region of the vortex generators the effective fuselage diameter is about 25 ft. Relative size may be gauged from a scaled vortex generator shown in Fig. 9.

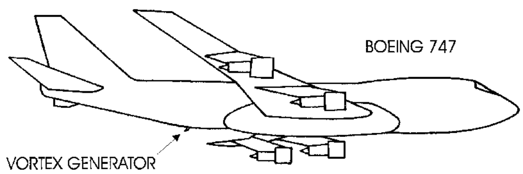


Fig. 9 Illustration of a possible installation of vortex generators on a Boeing 747.

At $M = 0.82$ cruising speed at 40,000 ft, the dynamic pressure is ~200 psf. A vortex generator with 2-ft^2 area operating at 30-deg effective angle of attack would be subject to an aerodynamic force of ~100 lb (440 N). Attachment of vortex generators to the existing fuselage skin and frame, by means of bolts and doubler plates, would not present any problems. If the vortex generators were made of fiberglass or aluminum, then a $\frac{1}{2}$-in.-thick, 2-ft^2 plate would weigh ~15 lb. With attachments, the vortex assembly weight should not exceed 25 or 50 lb per aircraft of the 747, C-5 class. A foam core–fiberglass skin structure would weigh less. Vortex generators fabricated out of composite materials would weigh even less, but their cost effectiveness would have to be determined in a preliminary design study.

Maintenance on the stationary plate-type vortex generators would be negligible. If variable pitch vortex generators were adopted to adjust the pitch angles with variations in flight conditions, then sensors, controls, and actuators would have to be employed. Such a system would not be expensive or complicated, but could require routine maintenance and functional checks. Presently, only fixed pitch vortex generators are being considered.

Discussion

The concept of employing a pair of relatively large vortex generators to influence both the inviscid flowfield and a retarded boundary layer on the upswept fuselage bottom differs fundamentally from the conventional small-scale devices that influence the boundary layer alone. In the applications to transport aircraft fuselages, the tangential component of the vortex flow is used to counter the inflow toward the plane of symmetry, whereas the strong axial flow energizes the boundary layer. Such strong interactions of an externally generated vortex with inviscid and viscous components of flow over aft regions of a fuselage are presently beyond analytical methods in aerodynamics, and it is necessary to perform wind-tunnel tests. Order-of-magnitude estimates and references to analogous flow situations were required to predict the size, configuration, pitch angles, and locations of the vortex generators. Demonstration of the validity of the concept required a test of large models in a large wind tunnel with an established reputation. The tests were designed for the California Institute of Technology 10-ft tunnel, and the model size was determined by the requirement that the region in which the vortex generators were located had to have a fully developed turbulent boundary layer. This was achieved with models of ~1:17 scale, and the Reynolds number referred to length that was ~10^7. The full scale flight Reynolds numbers of ~25×10^7 result in turbulent flows that are well represented by the fully developed turbulent boundary layers existent on the wind-tunnel models.

The usual pressures of budgets and schedules limited the number of configurations, pitch angles, and locations that could be tested. However, the results indicate that the effects are not very sensitive to variations of the preceding parameters, so that it is very likely that the results are fairly close to the optimum that can be achieved. The final optimization can most likely be realized in full-scale flight tests for the certification of the devices.

The cost estimates are approximate because a detailed design, fabrication, and installation study could not be performed within the scope of the program. It appears that a \$2000 contingency added to the best estimate of $3000 for the total installed cost should be adequate for any unforeseen circumstances. Even if the cost was doubled, it would still be insignificant in relation to the economics of operation, which could be realized through the use of vortex generators. The greatest problem in commercial deployment of the concept is the cost of the flight-test program that is necessary for Federal Aviation Administration certification.

Conclusions

The concept of installing relatively large vortex generators on the bottom, upswept regions of fuselages of transport aircraft to influence both the inviscid flow and the boundary layer, was validated in extensive wind-tunnel tests of Boeing 747 and Lockheed C-5 fuselage models. Total aircraft drag reductions of 1% for the former and 2% for the latter can be realized through the use of simple fixed pitch devices, whose installed cost is estimated to be less than 5% of the reduction in the operating costs of a Boeing 747 for the first month of utilization. All aft-loading military transports and most of the civilian airliners are seen as potential candidates for the installation of vortex generators.

Acknowledgments

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