the order of 0.10 in. was obtained, indicating sufficient promise to warrant further development of the system.

The first natural flight test of the system was conducted in March 1986 on Cessna's 208 Caravan, a single-engine turboprop.[3] Testing was conducted over the northern plains of the midwest, but due to the lateness of the icing season only three test flights were made. For these tests as well as the NASA IRT tests, an early version of the system was configured. PEEK-surfaced, rubber matrix deicers were installed on the right inboard wing (approximately 12.5 ft long) and on the right strut (approximately 8 ft long), in place of the standard pneumatic deicers. The ice protectors were configured with a single leading-edge tube located over the leading-edge centerline, and were bonded to the aircraft's leading edge skin in a manner similar to conventional pneumatic deicers. Thickness of the ice removed was typically 0.25 in. or greater, with some thin ice removal noted in the vicinity of the impulse entry ports. The testing underscored the need to tailor tube size and location to the specific airfoil geometry.

It was desired to make the PIIP system available for commercial aircraft, as a low-power alternative to bleed-air. The PIIP surface, while suitable for many applications, did not possess the rain erosion resistance required for commercial aircraft, and was therefore replaced with titanium in 1986.

A number of tests were conducted in Lockheed's icing tunnel in Burbank in late 1986 and 1987. Articles for these tests were titanium-surfaced, but still contained a rubber matrix and were bonded over metal leading-edge skins. These tests series revealed difficulty with the system in removing "slush" ice.

Since this test series, development of the system has focused on tailoring the surface dynamics in order to be able to remove effectively the difficult thin and wet ice. At the same time we have moved away from an external, skin-bonded construction toward a nonintrusive composite leading-edge embodiment, in which the ice protector and the leading-edge skin are integrally cured into a single leading-edge structure. The rubber matrix of the ice protector was replaced with a tough, flexible thermosetting plastic resin. Rain erosion testing of the new composite leading-edge construction using titanium skin demonstrated over 7 h @ 600 mph, in 1 in. per hour of rain, and 2-mm-diam. drops, with no damage. This far exceeds the rain erosion longevity of conventional aluminum leading-edge skins.

The 1988 activity culminated in a week of testing in the IRT at NASA Lewis Research Center. The system demonstrated 0.040-in. threshold thickness shedding capability in "dry" ice conditions and 0.090-in. shedding capability in wet ice or "slush" conditions.

Subsequent testing at the BF Goodrich Icing Tunnel as well as the NASA IRT of improved versions have demonstrated the low power requirements,[4] ease of intallation, and durability of PIIP. Life cycle tests have demonstrated in excess of 250,000 cycles on the ice protector and 2,000,000 cycles on the impulse valve, equivalent to the life of the aircraft on a commuter application.

Concluding Remarks

Over the last several years, there has been under development a new impulse-type mechanical ice removal system. This system is considerably different from other dynamic systems in that a pneumatic rather than an electrical impulse is imparted to the ice-accreting surface to remove accumulated ice. The system has demonstrated the capability to remove ice accumulations as thin as 0.030 and as small as 0.25 in. equivalent particle diameter,[4] rendering it viable for ice-sensitive airfoils and engine inlets, while retaining the capability to remove thicker ice of over 1 in. Life cycle tests have demonstrated long life capabilities, estimated to be equivalent to the service life of a commuter aircraft and more. Applications being studied for PIIP include: airfoils, tailplanes, and engine inlets for military, commercial, commuter fixed-wing or rotorcraft. PIIP is now being viewed as a replacement for the currently used gas turbine-generated bleed-air ice protection system.

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Minimizing Supersonic Wave Drag with Physical Constraints at Design and Off-Design Mach Numbers

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Introduction

SINCE the concept of supersonic area rule was introduced[1,2] the area-rule method has been the most successful and systematic method for the arrangement of vehicle components and the determination of the vehicle area distribution for minimum wave drag at a particular Mach number.

To date, the most widely used computer program incorporating the area-rule method is the Harris code, written by the Boeing Company and reported in Harris.[3] A useful feature introduced by Sheppard[4] for designing the fuselage area distribution with minimal wave drag was built into the Harris code. However, several drawbacks exist in Sheppard's design procedure.

1) It has no control on the difference between the volume of the baseline and optimum fuselage. This may give the volume of the optimum fuselage to be much less than a required value.

2) The "control points" on the fuselage impose the constraints that no area change at those points is allowed on the optimum fuselage. However, in most design situations, the area at these points may be allowed to change, but within an upper and lower bounds.

3) Sheppard's design procedure only gives the optimum area distribution at design Mach number. Therefore, among all area distribution and Mach number pairs, the designers may have to find a compromised solution by a trial-and-error procedure so that the performance at off-design Mach numbers are not drastically inferior.

4) In many design situations, only a particular section of the fuselage is allowed to be modified for the minimum wave drag. Sheppard's design procedure gives the area distribution along the whole aircraft only.

With the advent of super-speed computers and a general-purposed optimizer,[5] the above drawbacks in the Harris code can be solved by coupling itself with an optimizer and for-

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mulating a proper constrained minimization problem including objective function, constraint functions, and the design variables.

Mathematical Formulation

The problem of designing aircraft optimum area distribution with physical constraints for minimum wave drag can be seen in Fig. 1. That is, within a section started from Xs and ended at Xe of a given baseline aircraft area distribution $B(x)$, (represented by the solid line in Fig. 1), what is the optimum modification of area $A(x)$ that gives the minimum supersonic wave drag of the aircraft? While searching for the minimum wave drag solution, $A(x)$ must satisfy the conditions that:

1) The volume V provided by the area distribution $A(x)$ be greater than a required value.

2) At K numbers of locations Xk, $k = 1, \ldots, K$, $A(Xk)$ cannot exceed the given lower and upper bounds, Akmin and Akmax, $k = 1, \ldots, K$, respectively.

In the present formulation, the unknown $A(x)$ is approximated by the Fourier sine series as

$$A(X) = \sum_{i=1}^{n} ai \sin(i\pi(X - Xs)/L) \tag{1}$$

where $L = Xe - Xs$ is the length of the section.

The Fourier sine series in Eq. (1) gives three advantages to the present problem:

1) It is well known that any discontinuity of the area distribution and its slope of an aircraft configuration can induce flow separation and create additional drag. The sine series assures $A(x)$ to be a continuous and smooth function, which avoids these problems.

2) If both the baseline and optimum area distribution are continuous, it implies that $A(x)$ at Xs and Xe must be zero, as automatically satisfied by the sine series.

3) Instead of searching for $A(x)$, Eq. (1) redefines the unknowns to be n sine coefficients ai. This reduces the present problem from a continuous formulation to a discrete one that is suitable for numerical optimization.

For a given set of ai, the new area distribution $B\text{new}(X)$, can then be defined as

$$B\text{new}(X) = A(X) + B\text{baseline}(X) \tag{2}$$

and the wave drag coefficients Cdw are computed through the Harris code based on $B\text{new}(X)$. This is to say that, by defining ai as the design variables, the objective function can be evaluated through the Harris code. With these already defined design variables, the optimization problem in the present code is formulated as

Design variables ai, $i = 1, n$

Minimizing Obj

$$= \sum_{j=1}^{m} Wj[Cdw(Mj) - Cdw\text{baseline}(Mj)] \tag{3}$$

Constraints:

$$V - V\text{required} < 0 \tag{4}$$

$$Ak\text{min} < A(Xk) < Ak\text{max} \quad k = 1, \ldots, K \tag{5}$$

where $Cdw(Mj)$ is the wave drag of the optimal area distribution evaluated at Mach number Mj, Cdwbaseline (Mj) is the wave drag of the baseline area distribution evaluated at Mach number Mj, V is the volume enclosed by the area distribution $A(X)$

$$V = \int_{X_s}^{X_e} A(X)\, dX \tag{6}$$

Vrequired is the required volume specified, and Akmin and Akmax are the lower and upper bounds of the area at Xk for which $A(Xk)$ cannot exceed.

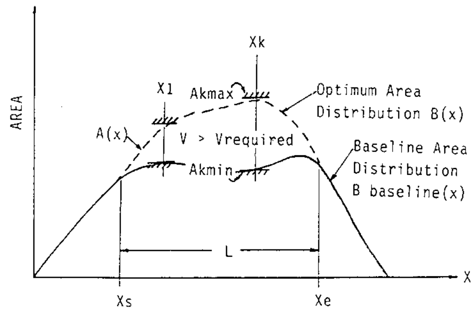

Fig. 1 Optimum area distribution with physical constraints.

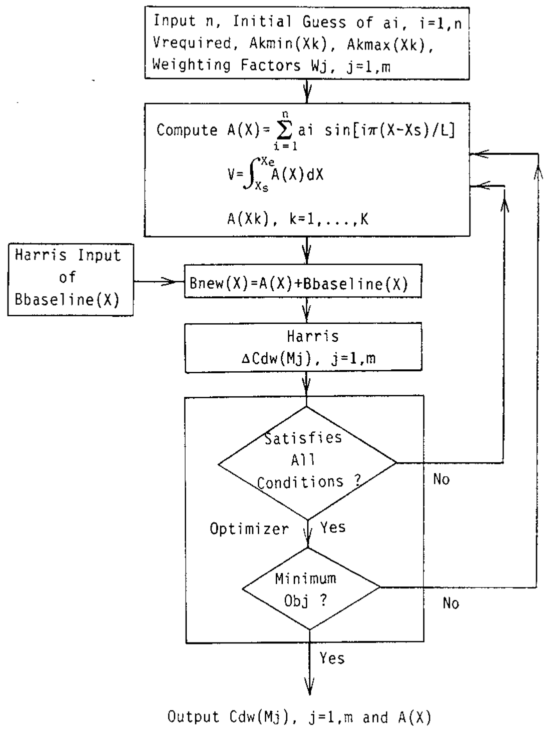

Fig. 2 Program architecture of the present method.

Eq. (3) expresses the objective function as the sum of the difference between the wave drag of the optimum and the baseline area distribution for m Mach numbers. Wj is the weighting factor of the jth Mach number. The weighting factor at design Mach number should be larger than others.

Now, Eqs. (3–5) are in the standard optimization form and can be solved by mathematical programming technique. The method of feasible directions[6] is selected for its ability of dealing initially infeasible design.

By coupling the Harris code with the optimizer, the program architecture of the present code is presented in Fig. 2. At the beginning, users must input the number of design variables n, initial guess of ai, and all required physical constraints. Usually ten design variables would be sufficient to obtain a converged solution. The initial guess of ais at zero value should be avoided to keep the optimizer from encountering the difficulty in normalizing the design variable. The Harris input contains the area distribution of the baseline configuration in terms of the original Harris input format. A typical run of the present method with ten design variables usually takes 25 CPU minutes on an IBM 3090-200E computer.

Results and Discussion

Given the feasible range of fuselage stations (F.S.), the desired volume, the design and off-design Mach numbers, the area distribution of a conformal tank is designed using the present method, as can be seen in Fig. 3. The result was

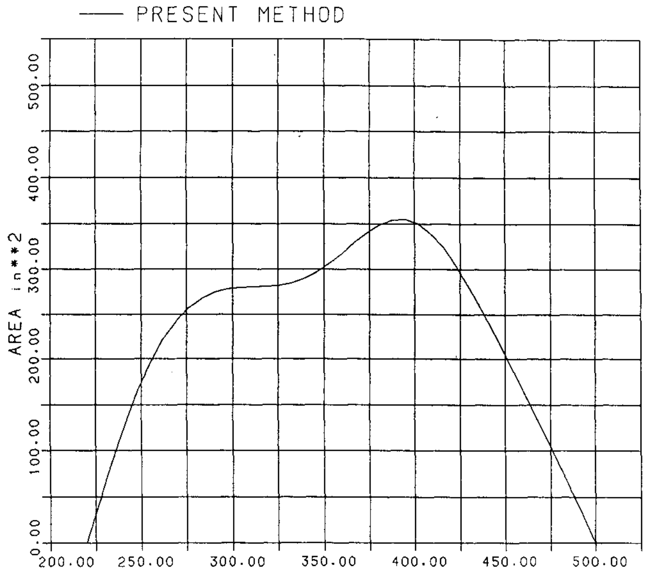

Fig. 3 Designed conformal tank area distribution.

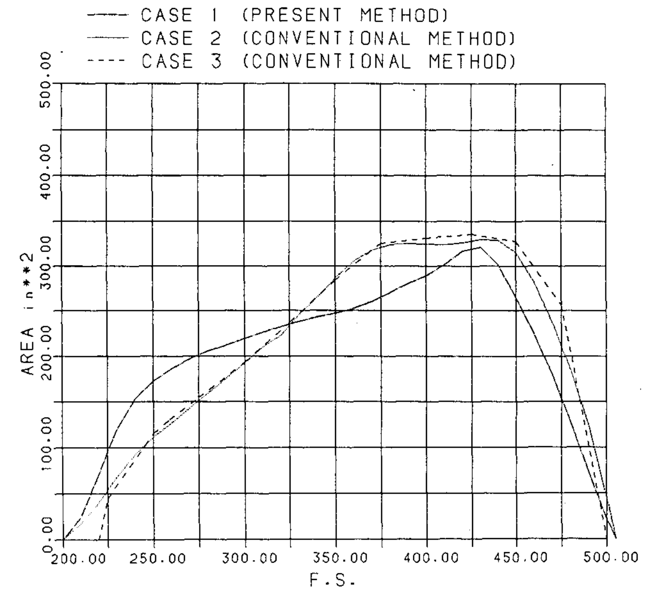

Fig. 4 Comparison of conformal tank area distributions designed by conventional method and present method.

Table 1 Comparison of drag increments for cases 1, 2, and 3. (Unit: counts)

Mach number	Design	Off-1	Off-2
Case 1	12.0	14.0	16.0
Case 2	19.0	19.0	20.0
Case 3	27.0	28.0	27.0

slightly modified by the lofting people to meet other design requirements such as structural constraints and subsystem arrangements. The modified area distribution is designated as case 1 in Fig. 4. Two other designs designated as cases 2 and 3 were obtained by modifying the results from Sheppard's design procedure based on designer's experiences.

All of these three cases were tested in the 4 × 4 ft high-speed wind tunnel (HSWT) of ARL at design Mach number and two off-design Mach numbers. The incremental drag coefficients obtained from the test are presented in Table 1. The wind tunnel data show that case 1 gives the lowest drag increment throughout the Mach numbers of interest. Note also, that case 1 gives least drag increment at design Mach number.

Conclusion

A computer program is developed for the design of optimum aircraft area distribution for minimum supersonic wave drag with physical constraints. This program can outperform the Harris code by its fully automated design process and its capability of dealing many physical constraints and balanced consideration of Mach numbers of interest. Wind tunnel test results of a conformal tank design shows that this method not only gives the lowest wave drag throughout the Mach numbers of interest, but also meets the objective of least drag increment at design Mach number.

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Suppression of Fatigue-Inducing Cavity Acoustic Modes in Turbofan Engines

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Introduction

MODERN turbofan engines use ducting systems to transport freestream air, bypass fan-air, and engine core primary air to pumps, generators, and heat exchangers. Typical ducting systems have upstream openings exposed to a flowing air field and are regulated by downstream valves (Fig. 1). Such ducting systems are cavities having acoustic modes that are often excited to high sound pressure levels (SPL) by the cavity shear layer when the downstream valve is closed. This can induce dynamic loads capable of causing fatigue failures in the ducting system and neighboring structures. Weight, cost, and design restrictions often force one to extend the fatigue life of such ducting systems by reducing the excitation of its acoustic modes rather than by raising its load-bearing capability. Three concepts of reducing shear-layer excitation of a cavity's acoustic mode are evaluated in this study using a full-scale turbofan engine test.

Shear-Layer Excitation of Cavity Acoustic Modes

When fluid passes over an open cavity, a shear layer of vortices may be generated at its upstream edge that roll across

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