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# LOW SPEED AERODYNAMICS FOR ULTRA-QUIET FLIGHT 

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TECHNICAL REPORT AFFDL-TR-71-75
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AIR FORCE FLIGHT DYNAMICS LABORATORY
AIR FORCE SYS'TEMS COMMAND WRIGHT-PATTERSON AIR FORCE BASE, OHO


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## FOREWORD

This report was prepared by The University of Tennessee Space Institute, Tullahoma, Tennessee, for the Aero-Acoustics Branch, Vehicle Dynamics Division, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, under Contract F33615-70-C-1762. The work described herein is a part of the Air Force Systems Command's exploratory developrent program to predict the noise environment of flight vehicles. The work was directed under Project 1471, "Aero-Acoustic Problems in Air Force Flight Vehicles", Task 147102, "Prediction and Control of Noise Associated with USAF Flight Vehicles". Capt. R. P. Paxton of the AeroAcoustics Branch was the task engineer.

The authors wish to thank their coil leagues Messes. Dieter Nowak, Jerry Coble, Frank Keeney, Ing Botchers, Gunter Schrecker, Jim Goodman and Lt. Douglas Marshall for their valuable assistance during the course of the research project.

The program was monitored by and under guidance of Dr. B. H. Goethert, Dean, The University of Tennessee Space Institute.
'manuscript was released by the authors on April 26, 1971 for publication as an A"FDL Technical Report.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.
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Asst. for Research \& Technology Vehicle Dynamics Division

A combined Aerodynamics, Acoustics and Bionics study was conducted in an attempt to discover novel mechanisms to reduce the noise associated with aircraft flight. The strigiformes orde: of birds, solected in the Bionics effort as possessing characteristics of silent flight, was studied extensively. Three mechanisms producing the potential for acoustic quieting were discovered as a result of this study. These are;

1. Vortex sheet generators,
2. Compliant surfaces and
3. Distributed wing porosity

An experimental program aimed at initiating full scale flight evaluation of these concepts was outlined. The detailed results of these studies are included.

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## LIST OF SYMBOLS

a

A Peak amplitude of vibration, ft.; Coefficient boundary layer, dimensionless

Aspect ratio, dimonsionless
c Local wing chord, fit. Characteristic length of an airfoll, ft

C Distance from camera to the floor, ft
Drag coofficient, $2 D /\left(\rho S V^{2}\right)$
Lift coefficient, dimensionless
Diameter or distance, ft.
Decibela, relerence $2 \times 10^{-4}$ microbar
Dimension of acoustic source, ft.; Drag, l.b.
$D X^{*}, D Y^{*}$
DZ*
i Charactoristic irequency, eps; Frequency of vibration. cps; Vortex frequency, cps
$f_{n} \quad$ Natural frequency, cps

F Acoustic pressure radiation function, lbs/it
$\mathrm{F}_{0}$
GPR
H
1 Length, ft.

1

L/D
m

M
Acoustic velocily, [t/sec

Equivalent parasite area, $f^{2}$

Forcing function, lbs
Gilde path ratio, dimensionless
Distance from flash unit to the floor, it

Lift per unit span, lbs.; Lift, lb
Lift-to-drag ratio
Mass per unit span, slugs per foot
Volume of test chamber, $f^{3}$
$X, Y, Z \quad$ Position vector coordinates, f+; Apparent rectangular
coordintes of owl, ft.
$X^{*}, Y^{*} \quad$ True rectangular coordinates of owl, ft.
$\mathbf{Z}^{*}$
$X_{1}$
$\alpha$
$\gamma$
$\therefore \quad$ Boundary layer thickness, ft.

| * | Boundary layer displacement thickness, ft. |
| :---: | :---: |
| $\therefore 2$ | Boundary layer momentum thickness, ft. |
| $\wedge D$ | Actual flight distance, ft |
| $\triangle t$ | Time interval between dual flash, sec |
| $\eta$ | Proportionality factor, dimensionless |
| $v$ | Kinematic viscosity, $\mathrm{ft}^{2} / \mathrm{sec}$ |
| $\rho$ | Alr density, slugs/ft ${ }^{3}$; Density of air, $1 \mathrm{~b}_{\mathrm{m}} / \mathrm{ft}^{3}$ |
| 0 | Standard deviation, dimensionless |
| $\varnothing$ | Glide angle, deg |
| (1) | Characteristic frequency, rad/sec |
| Subscripts |  |
| 1 | Conditions at first flash |
| 2 | Conditions at second flash |
| 1,2 | Specified positions or conditions |
| bl | Boundary 1ayer |
| c | Correlation; Coeflicient |
| ' | Friction |
| 0 | Remote Condition |
| t | turbulent |
| $\mathrm{U}_{1,2}$ | Associated with velocities 1,2 |
| vs | Vortex shedding |
| w | Wake |
| $\infty$ | Remote condition |

## INTRODUCTT ION

From a military standpoint it is desirable that aircraft operate in such a way as to avoid detection. One means by which aircraft are detected ts by the noise they generate and propagate to a receiver. Aircraft which have been designed to ininimize aural detection are called quiet aircraft. The primary noise associated with quiet aircraft is emitted from the propulsion system and from sources associated with the aircraft in flight (radiated noise from boundary layer turbulence, vortex shedding, wakes, cavities, vibrating panels, etc.). Within aircraft performance limitations, the propulsion system noise can be controlled through design and installation. When these sources are minimized, the noise associated with flight becomes the dominant factor. The attainment of an ultra-quiet aircraft requires that the sources of noise associated with filght be identified and that means of suppreasing this noise be developed.

A solution to silent flight is found in nature in birds of the order strigiformes, Species of these birds have evolved with a number of configurations each with the common characteristic of silent filght. Wings of these birds differ from other birds in that they have the following features: (a) a leading edge comb, (b) trailing edge fringe, and (c) a soft and porous upper surface. These characteristics are described in a paper by R. R. Graham (Reference 1).

The technical effort on this project was directed toward discovery of new phenomena and development of engineering information pertinent to the design of a quiet aircraft capable of flying at low altitude in an aurally imperceptible mode. The Bionics approach was talsen as the primary element of the research for several reasons. The field of ultramquiet filght is relatively new and basic studies were felt to be important in their potential for major advances. As in any other scientidic study, the precise description of a phenomena may not be as valuable as basic concepts which may arise when considering the conceptual elements of a natural system for use in an applied system. Thus the technical approach employing Bionics under the assumption that "the flight characteristics of the highly specialized Owl were evolved toward the same end as our application and could be simulated in a practical design", opened a new potential for discovery.

The objective defining the work performed under this contract was to conduct an investigation of the noise sources associated with unpowered winged flight and to determine methods of suppressing the nolse radiated from these sources. The basic nature of the flow associated with an airplane wing caused this
flight olement to be the local point of the study. The only types of flow noflected by concentrating on the wing alone aro those accoumpmaying wing/body interference, interaction of an airloil with a turbulont wake and possibly llow at very high Reynolds Numbers as would be experienced on a long fuselage. The basic mechanisms of noise generation to be studied were therefore reduced to:
(1) Boundary layer,
(2) Unstendy 1ift
(3) Wakes.

Low speed sailplane flight, between 30 and 100 mph provided the reference Reynolds Number range for the program.

Until recent years, the mechanisms which produce acoustic disturbances in aerodynamic flight were not woll known. Extensive research programs (for example, Reference 2-7) have been directed toward predicting and reducing noise in aeronautical systems. However, the ability to design to an acoustic power level requires further developments in noise suppression technology (Reference 8). It is felt that reduction of aircraft nodse to an acceptable level will preceed the ability to design to a particular level.

The natural filght syetems chosen for this research are said to possese tho quality of quiet flight. Significant similarities exist among the various species of owl to guggest the need for a concentrated study of their undque filght methods of reducing noise. Two representative specimens of live owls were obtained for this research.

This project was organized in a manner to best utilize the talents of a team of specialists in Bionics, Aerodynamics and Acoustics. In the oarly phases, the three teams worked in dependontly, both for speed and for broadening the scope of possible approaches. During the remaining phases, the teams worked as a unit, using the Bionics approach for naxrowing the possibilities and allowing concontrated offort on the most promising ideas. Though this report is assembled to reflect the three distinct elements, the contents of onch reflect the integrated program.

Finally, having discovered quieting mechanisms of silent flight, practical techniques wore to be applied to a typical sailplane airfoil section in preparation for future system installation on a piloted sailplane. Design specilications of these elements were also to be developed.

## BACKGROUND STUDIES

## 1. QUIET FLIGHT


#### Abstract

Flight within the atmosphere is accompanied by an interchange of energy between the vehicle and the surrounding air. The concept of a quiet aircraft requires the reduction of the noise produced by this interchange to a level not noticeable by unsuspecting but alert personnel.


The aural detection of an aircraft dopends on several factors. These include: (1) the intensity and radiation pattern of the noise generated by the aircraft, (2) the spectrum and real time character of the generated noise, (3) the distance separating the aircraft and the observer, (4) the atmospheric absorption, (5) the background noise present in the observer's environment, and (6) the sensitivity of the observer to the received noise. In Reference 2 , methods are presented for developing aural detection criteria considering the making offects of the background noise and the hearing sensitivity of the observer (Factor 5 and 6 above). Flgure 1 shows an aural detection curve for a daytime jungle. The aural detection levels presented have been modified by replacing the threshold of hearing curve given in Reference 2 with that given in Reference 3. As can be seen in Figure 1, an afert observer should be able to sense a sound preasure levell (SPL) of -4 dB for a 4000 Hz tone in the absence of background noise. However, he is restricted to a considerably reduced sensitivity at other frequencies. In addition, the presence of background notse tends to mask other noise signals and, as shown in Figure 1 , the masking level of the jungle background noise determines the aural detection level for frequencies over 125 Hz .
A.s indjcated in Reference 4, the aural detection range of an aircraft is determined by comparing the spectrum levels of the received noise signal from a given altitude with an oural detection level curve. The maximum amount by which the signal exceeds the detection curve determines the aural detection range. For example, consider the flight of a sallplane over an observer in a jungle environment. A one-third octave band spectrum of the received noise signal from such a mailplane at an altitude of 125 feet (from Reference 4) is Ahown in figure 1. In Reference 4, it is mown that the apectrum level of the sailplane's noise

[^0]simmal exceeds the detcetion level cuive by a maximum of 18 dB at 310 Hz with the rosulting aural detection range being approsimitely 1000 fect over a quidet jungle.

The capacily to provide aurally nondotectable ilight at a given altitude thus depends on the ability to attenuate the aircraft noise signature such that the received noige signal is below the aural detection lovel curve for a given environment.

## 2. SOURCES OF NOISE

The mady of ultra-quiet flight introduces the need to consider all of the potentinl types of noise. The characterm istics of directivity and multiplo source summing may amplify the influence of subtie contilibutors. The relative importance of the primary sources of notse has been studied for missions wherein silence is a prerequisite. Normally the propelloi/ongine noise is sufficiently larke to mask the remaining noise gources. However, with the recent successes in low speed propulsion system quieting, the airframe produced sources predominato. These are discussed in minimum detail in the following paratraphs.

## a. Vortex Cores

The vortex sheet which develops at a point of steep spunwise lift slope is nomally quiat strong and wrapped tightiy. Its core is downstream of an element auch as a wing or flap tip and is usually low in axial onergy because of a local flow separation upatream. The rotational energy of the core may also be quite low. The surrounding vortox sheet is at an elovated energy loval and induces high velocities at the interface boundaly. The potential nature of the main vortex implies no acoustio disturbancos, yot intoraction with a region of another onergy leval provides a nobse souree, dependent on the finelpiont mature of the interaction.

## b. Boundary Layor

As the vohicle boundary layer transitions to turbulent filow, unsteady pulsations occur and are transmitted into the surrounding field. The extent of this problem is a function of the aerodynamic conliguration, Reynoids Number, attitude and surface roughness. A broad band scoustic radiation signal ia generated by the resulting boundary layer sources. Control of noise would certainly accompany the ability to shape the boundary layer conflyuration.

## c. Vortex Nolse

The natural consequence of three-dimensional iffing surface performance is the generntion of an extensive sheet of diatributed vorticity. The exact character of this sheet is not well known for arbitrary wings and the individual vorticity elements may nave axial as well as lateral components. As they are in a partly viscous region, littio can be said of their closure. Thus, their life may be extremely short, being accompained by a rapid energy exchange. Periodicity is not always a property of this source, though for simple cases its effects may be described analytically.

## d. Unsteady Lift

Most forms of notse are accompained by an unsteady flow process. The development of unsteady 11 ft implies a changing circulation about a body which may be a consequence of a fluctuating induced angle of attack of a rigid aurface or a vibrating surface in a steaciy flow. The existence of such changing circulation about the airfoll may produce noise. Fluctuating oirculation may be excited by the classical Karman street or broad band turbulence introduced into wake at the wing trailing edge.

Though these contributors to acoustic disturbance ant in varying degrees and may be coupled by some aerodynamic interm face, they may be considered af independent for such a basic investigation.

## 3. BIONICS

Bionics is the study of nature's golutions to the problems of living syetems, suggesting the application of these solutions to man-made systems. Ultra-quiet flight may be presenting problems which fall in the realm of novel science. Upon searching the field of aeronautics for olutionm, the sailplane was first considered but as Figure 1 shows, it fall short of being imperceptible in a jungle enviromment.

The mitence of Bionics offers a well adapted and very effective aerodynamic matem for silent flight through the species of Owl. Over milition of yearm of development, this species has produced many specialized configurations with varying degrees of body solidity, aspect ratio, planform mape, $W D^{\prime}$, gross weight and performance. However, virtually every member of the apectes has the common characteristic of silent filght. This implies a successful development toward this common goal. If this is truly a primary objective of this evolution echeme, as it has
been described, considerable optimization has taken place and much can be learned through a detailed engineering and scientiffe study of the present spocies of owl.

Special emphasis will be placed in the study of the owl's wing-feather structure. For instance, ita leading edge contains amall feathers with the appearance of hooked combs. It is probable that these as well as other unique elements combine to reduce noise generation.

## SECTION III

ANAI,YSIS AND DISCUSSION

## 1. MAJOR NOISE SOURCES

The prediction of acoustic disturbances originating in low speed aircraft flight wam studied. Referencen 6 through 11 show that the technology is not completely developed. In the early phases of the project it was found necessary to establiah a level of importance on the major noise aources. A method described in detail in Appendix I was umed. The results are shown In Figure 2. Both the boundary layer and unsteady lift noise components were found to warrant the first consideration. The wake generated noise was found to be of negligible consequence In the low speed lange of intereat in this research.
2. UNIQUENESS OF THE 1.WI,

For many years, the owl has been proclaimed to be a silent flyer. His three elements of uniqueness are:
(.1) The leading edge comb,
(2) The trailing edge fringe,
(3) The downy upper surface of the feathers.

In addition, a wing porosity exists which is a consequence of the soft feather structure.

In some way these elements must work together to suppress audible acoustic disturbances originating from the turbulent boundary layer and unstoady lift as well ar the other possible sources.

The Owl in known to have three distinct phases of flight which must be silent. These are;
(1) The flapping, or propulaion and guidance phase,
(2) The gliding phase, and
(3) The phame of conflguration change during touch-down.

All of these phases must be very silent as his prey is wary and the Owl filght is comparatively quite slow and clumsy. This research was concentrated primarily on the gliding phase though the
contribution of the owls unique qualities to the other phases was always kept under scrutiny.

## 3. SEQUENT IAL, APPROACH

The goals of the project were set out in the beginning to be obtained by the following approach:
(1) Develop an acoustical measurement system capable of measuring very ahort duration, low lovel aooustic properties.
(2) Obtain and calibrate a reverberation chamber large enough to test the Owl's silent gliding flight phase.
(3) Study the unique characteriatica of the Owl filght with a team compored of Aerodynamiciats, Acousticians and Bionics experta, using Zoologiats for consultation in certain specialized areas.
(4) Irain the Owls to tly in a manner acceptable for recording the spectral dietribution of his radiated noise.
(5) Develop an acoustic data reduction technique for for ahort duration noise measurements.
(6) Develop a wind tunnel with mufficiently mooth flow to test mounted winges of a smal species of owl.
(7) Perform wator tunnel experiments, using a sailplane airfoll section, to search for practical means of simulating the notse suppression mechanism of the Ow1.
(8) Teat other spocies of birds to verify the quiet nature of Ow] filight.
(9) Modify the wings of the living owla and acoustiecally monitor their filght to show the effect of removing the quieting mechanisms.
(.10) Define the ailencing mechanismf of the Owl and make recommendations for allplane experiments to verify their validity.

In depth descriptions of this approach are included in Appendices I-V.

## 4. FINDINGS

When the first acoustical data reduction was completed for the Owl flights, it was seen that the Owl had a significant acoustic radiation. The interesting part was that the noise spectrum was shifted strongly toward low frequencies range. In fact, it was seen to nearly match the human perception threshold curve at a distance of three meters. The owl does not radiate the broad band type noise developed by sadiplanes as depicted qualitatively in Figure 3 and discussed in depth in Appendix IV.

Appendix $V$ shows the acoustic quieting of the Owl to be accompanied with a very poor flight performance. For example, the ilit-to-drag ratio of the owl is lesg than five while the albatross may reach peak values above thirty.


#### Abstract

It was thus decided to search for boundary layer and unsteady lift noise suppression mechanisms which were not necessarily aerodynamically efficient but could shift the acoustical notse spectrum to low frequencies as accomplished by the owl. The wind tunnel studies aided in this matter. The tests of the mall owl wing mounted in a wind tunnel showed a wing instability to occur at approximately 15 cycles per aecond. This experiment is described in Appendix II. This frequency was seen to correspond to the peak noise radiation measured in the reverberation chamber for the live Owls, Strobe light studies of the small wing mounted in the wind tumel showed an aeroelastic compliance of a rather complex nature at this frequency. Reference 12 wam consulted concerning boundary layer turbulence attenuation by compliant surfaces. The result was that compliant surfaces mhift the boundary layer turbulence frequency spectrum to lower levels.


A curve from Reference 13 is shown in Figure 4. It shows the turbulent boundary layer attenuation of a compliant surface with reference to a solid surface. This curve is qualitatively the same as the attenuation which would accompany the comparison of the Owl with the eailplane referred to the same total noise energy level. This conciusion promoted the water tunnel studies described in Appendix III. Figure 5 shows the compliant nature of the owl wing in gliding flight. The flutter of the feathers on the upper surface was borne out in further wind tunnel and gliding flight tests.

The value of the leading edge comb to the silont flight of the Owl was discovered. In esmence, the leading edge comb is a vortex shoet generator (not a classical vortex generator) which works in combination with the leading edge slot and tip feathers to promote attached laminar flow over the entire outer half of the wing span. The mechanimm was thoroughly probed and is described in detail in Appendix II. A gketch of the boundary layer streamline pattern im hown in Figure 6. It must be emphasized that the region of counter-rotating flow is not separated but reversed laminar flow which fills the void
induced by the divelgtif streamilnes ncar the leading edge tolot. The prossure drop created by the flow around the slat adjacent to the vortex sheet fenerator allows the nir to easily nofotiato the 180 degree turn and dow smoothly in the pattern shown. It is hard to depict the three-dimensional flow in a sketch; however, approximately four millimeters above the wing the flow is in the chordwise direction and quite comparable to that over a conventional airplane wing. These conditions can exist only with the aid of tho special vortex sheet. The wake is very thin at the tiailing odge. The weak nature of the induced tratilng vortex sheet implies a fentle lift diop-ofif towad the tip. The very complicated flow pattern found here was seen to allow the existunco of laminar tip llow up to an argle of attack near 30 degrees. This is unusual porformance for an airfoil with an unswopt thin leading edge. lhe fact that the stagnation line is along the comb at small angles of attack seoms to offer an additional dom sign advantage for high mpeod airplunes.

An explodad view of the leading edge feather possessing the vortex shoet generator is shown in Figure 7. Note the sol't downy substance on the upper surface. This material also covers the region of contact between the feathers. No proof of its function could be made. Howover, two qualitios ware lound which possibly shed light on their function. Those are:
(1) The downy surface allows a undioxm film of aid to past botweon the feathers so that an evenly distributiad porosity could be produced. This liow between the feathers could produce a chordwter boundary layer thickening for reduced nodse at the trailing edge.

The second explanation, is:
(2) The downy surface produced lubrication which allowed the winks to bo quitety folded into an assaujt. position right belode touchdown as shown in Fifure $H$. Tho wings on the live owls could bo torcelully oxtended and elosed with a very oidorly und quiet motion of the silding feathers. As another polat of intorest in figure 8, note the dijferent atidtudes of the leading edge slat as depicted in he Hhadow on the tloor.

In effect, the second possibility was not fell to be important in the gliding filisht phase other than its contribution to compliance.

One other interesting observation of the owl's flight was his virtual disregard for the position of his lege. figure $\theta$ shows such a configuration. Of the numerous photographs takon, his leg position seems somewhat arbitrary. If this is the case in his quiet flight mode he must possess a versatile silencink mechanism. This could be in the foxm of the long, compliant and very fine feathers attached over his legeandinhis base regiol.

## 5. APPLICATIONS TO SAILPLANES

A desifn fur application of the silencting mechanisms of the Owl to an existing sailplane was not simple. In fact, aiter considering the influence of tip modifications to the lateraldirection stability and control as well as to the stall characteristics of the vehioie, recommendations concerning the applications of tip vortex sheet generator will be withheld pending exhaustive wind tunnel evaluation. This is due to the lack of existing analytical capability for calculating the three-dimensional flow invalving boundary layer interaction. However, it was felt that though the structure of the flow induced by the application of compliant surfaces was extremely complicated, the immediate application to a sailplane for flight testing would not be excossively dangerous.

A series of water tunnel experiments were conducted, as described in Appendix III, to determine a simple, lirst look at the influence of a type of compliance on the acoustic and aerom dynamic performance of the gailplane. A splitter plate was extending from the trailing edge of the wing and a "bent" length was tound. This length was $12 \%$ of the local wing chord. Fifure 10 shows how the splitter plate, if mounted flexibly from the trailing edge, might simulate a conipliant surface, Notice thot as the plate is moved up, the point of turbulent boundary layer transition moves forward on the upper surface and rearward on the lower surface. Conversely, as the plate is moved down, the reverse occura, The ilmitation on the success of a sailplane moidification progiam will depend on low well the splitter plate can regulate the flow in this lashion. No doubt, compliance extending further onto the wing would have had more influence. However, eince so little was known of the phenomena for quieting, such an extensive study was felt to be beyond the scope of this research. A limition on the splitter plate approach is that the overall wing atreamwise movements compared to those of the splitter plate will be very small. The owl's compliance is such that the entire wing will deford under the influence of the trailing edge motion.

The observatione made in the gtudy of the splitter plate were as follows: As the turbulent boundary layers developed on the upper and lower surfaces a closed loop aerodynamic control system was developed. When tho ajrstreams along the upper and lower sulfacen coalesce at the trailing edge, an alternate vortex shedding develops. This may occur at a single frequency or over a broad range of frequencies, depending on the Reyriolda Number, Figure 11 shows this action for a cylinder. The same type of aystom develope behind an airioil excent the prequency in higher. The experiments by Strouhal[15]show this to be

$$
\begin{equation*}
f_{0}=0.185 \frac{V_{\infty}}{C} \tag{1}
\end{equation*}
$$

where $f_{o}$ is the vortex frequency in cycles per second, $V_{\infty}$ is the velocity in feet per second and $d$ is the cylinder diameter in feet. For application to the airfoll, d becomes the boundary layer thickness at the trailing edge.

By adjusting the stiffness of the trailing edge attachment the frequency of oscillation of the splitter plate can be tuned to a predominant wake frequency and thus be driven to amplitudes of at least the thickness of the boundary layer. The prediction of this amplitude will be limited because the variation in total pressure across the path of the plate travel will add a nonlinear aerodynamic spring rate.

The amplitude of motion assuming a constant, quasi-steady, aerodynamic spring rate would be;

$$
\begin{equation*}
A=\frac{F_{o} / m(2 \pi f)^{2}}{\left\{\left[1-\left(f / f_{n}\right)^{2}\right]^{2}+\left[2 \gamma\left(f / f_{n}\right)^{2}\right]^{2}\right\}^{1 / 2}} \tag{2}
\end{equation*}
$$

where $f$ is the forcing frequency, $f_{n}$ the natural frequency, $m$ is the effective plate mass (including the local air mass) and $\gamma$ is the damping factor. It was seen that the prediction of the amplitude would be difficult if the nonlinear terms were included.

Consider the design of the splitter plate attachment where the hinge has small damping. In this case, Equation 2 may be approximated by;

$$
\begin{equation*}
A=\frac{F_{o} / m(2 \pi f)^{2}}{l-\left(f / f_{n}\right)^{2}} \tag{3}
\end{equation*}
$$

This equation says that if the forced frequency of vibration, $f$, of the splitter plate is above the system natural frequency, the displacement will be out of phase with the force. If the forcing frequency is small, compared to the natural frequency they will be in phase. In a very rough sense, this action implies an atcenuation of the unsteady lift at high frequency and an amplification at low frequencies in a broad band turbulence wake through the phasing of the camber change with the induced downwash.

For the purposes of analysis of this approach, assume the attachment spring constant to be zero. The average velocity in the region of motion is $V_{\infty} / 2$ and the lift curve slope for the splitter plate is $2 \pi$. For harmonic motion, the oscillating lift on the splitter plate becomes

$$
\begin{equation*}
L=\frac{\pi a c \rho V_{\omega}^{2}}{4} \operatorname{Sin}(2 \pi f) t \tag{4}
\end{equation*}
$$

per unit span. Where $\alpha$ is the effective angular deflection, $c$ is the chord length, $\rho$ the air density, ithe vibration frequency and $t$ the time. If the effective system mass and the lift are assumed to act at the $2 / 3$ chord position of the splitter plate, the natural frequency of motion becomes

$$
f=\frac{1}{2 \pi} \sqrt{\frac{\mathrm{dL} / \mathrm{d} \alpha}{2 / 3 \mathrm{~cm}}}
$$

or, using Equation 4,

$$
\begin{equation*}
f=\frac{V_{\infty}}{4} \sqrt{\frac{\rho}{\pi m}} \tag{5}
\end{equation*}
$$

Next, by setting the natural frequency of the splitter motion Equation 5) equal to the wake vorticity frequency (Equation 1),

$$
\begin{equation*}
0.185 \frac{V_{\infty}}{d}=\frac{V_{\infty}}{4} \sqrt{\frac{\rho}{\pi m}} \tag{6}
\end{equation*}
$$

From this, the mass per unit span of the splitter plate becomes

$$
\begin{equation*}
m=0.6 \rho d^{2} \tag{7}
\end{equation*}
$$

Even fur thick boundary layeds, this equation implies the need for a very light weight splitter plate if a flexible hinge is used. Even if the plates were made from a light material such as balsa wood, some additional spring stiffening would still have to be added at the hinge line to increase the system's natural frequency to the level of Equation 1.

By assuming the net boundary layer thickness at the trailing edge to be around one-tenth foot and assuming a flight velucity of 100 ft . per second, the fundamental wake frequency would be around 180 cps . Therefore, a range of frequencies up to fairly high values should be employed.

The preceeding anslysis was very rough and provides only a starting point for design. The avialable theory is not sufficient to predict such parameters as spifter plate chord length, mass distribution or elasticity distribution, though a more rigorous analyses might provide more ingight into the motion trat might be expected.

## 6. PROPOSED FLIGFI TES'I PROGRAM

Based on the above findings, the following flight test program is suggested in order to evaluate the splitter plate concept of compliance simulation.

Strips of $1 / 16$ inch thick balsa wood with a chordwise dimension of 0.12c should be hinged to the trailing edge in a very flexible manner. Tho grain of the wood should be allgned In the streamwise direction. The spanwise dimension would be one toot. In the first trials, several filght tests should be run using only a few strips. being mounted at about the $1 / 4$ span position in the same manner on both wings.

These first tests will determane the durability of the plates and the attachment techniques. A good grade of cloth tape should be sufficient to hold the plates in place. It is expected that the second bending mode may be set up in the platos so their life apan may not be very long. A broad speed range should be traversed in the sajiplane with merely visual observations being made during and aiter each flight. Should the attachmont and performance of the plates be satisfactory, additional ones may be installed incrementally between high speed runs. As more plates are added, the top flight speed should be reduced as a flutter prevention precaution. In no case should the plates be jnstalled outboard of the $75 \%$ span or the inboard edge of the aileron, whichever is the smaller without a detailed flutier analysis.

Acoustic measurements may be made upon establdahing the new configuration. Strips of shim stock may be added to incieaso the stiffness. Bul agin, the stiffening should be started in. board and extended to the outboard regions of the wing with small fincrements of stiffenting being made between flights. A small amount of canber may be added to the splitter plates if another flow position of the plate is desired.

## SECTION IV

## CONCLUSIONS

Extensive aerodynamic and acoustic studies were made of the owl wing in an attempt to isolate the mechanisms which roduce his noise level. The research succeeded in exposing three mechanisms which could aid in supressing noise due to ungteady iift and turbulent boundary layer. These are:
(1) Laminar, attached flow over the outboard half of the wing produced largely by the influence of vortex sheet generators.
(2) Shift in frequency spectrum, compared with solid surfaces, to the lower range by the action of compliant gurfaces.
(3) A thickening of the boundary layer and a reduction in the velocity gradients at the trailing edge by the action of a porosity diatributed over the wing.

Splitter plates at the trailing edge were found to strongly influence the boundary layer transition and mixing near the trailing edge. A flight test program was proposed to evaluate the spiitter plates in reducing noise. Care must be taken to isolate the effects of the splitter plate alone and the degree of compliance it simulates.

Figure 1. Aural Detection in Jungle Environment.


Figure 2. Relative Acoustic Amplitudes of NACA $\quad \mathbf{6 5} \mathbf{5}_{2}$


Figure 3. Qualitative Noise Levels


Figure 4. Percentage Attenuation of Compliant Piate with Rewpeot to Hard Plate Spectra of Turbulent Energy at Y/Sw0.0033.


Figure B. Photograph Showing Trailing Edge Vibration.

Figure 6. Streanline Pattern Over Upper Surface of Owl wing.


Figure 6 (Conc luded). Flow with Comb Removed.



Figure 8. Double Exposure Showing Changing Flight Contidyuration.


Flgure 8. Photograph Showing Arbitrary Leg Position.


Figure 10. Influence of Splitter Plate Attitude on Boundary Layer Transition


Figure 11, Schematio of Vortex Flow Around a Circular Cylinder.

## APPENDIX I

## RELATIVE STRENGTHS OF MAJOR AERODYNAMIC SOURCES FOR AN AIRFOIL IN LOW SPEED FLIGHT

## 1. INTRODUCTION

a. Background

The purpose of this work finds its origin in the increasing interest in achieving silent flight. It is limited to the investigation of the aerodynamic noise caused by the turbulent boundary layer, lift fluctuations caused by vortex aheddiny at the trailing edge, and the wake resulting from the movement of the wing form through the air. For this work an engineless, high performance sailplane, the Standard Austria SH1, was used as an example. The wing cross section of interest was the Eppler 266 laminar flow airfoil.

## b. Objective

The goal of this study was to identify the notse sources ol the Eppler 266 airfoil as contained in the wing planform of tho Standard Austria SH1 and to determine their relative magnitudes in the law subsonic regime. The Reynolds number and Mach number rangen were restriuted to approximately $1,200,000$ to $3,750,000$ and 0.05 to 0.185 , rexpoctively. The primary objective in accomplishing this goal was tro qualitatively predict the frequency epectrum and the relative power output of the sound generated by turbulent boundary layer, the lift fluctuations caused by the vortex shedding at the tialiling odge of the alifoil, and tho wake by theoretical means. This study should then provide the basis for continued investigation and the future flight test of the sailplane to record the nodse genexatod by an area of the wing (excluding the wing tip, aileron deflection, and fuselage effects). By correlating the results of this study and future teate, the actual noime level of the predominant sound source should be identiliable.
C. The Approach

This study relies heavily on the theoretical and experimental efforts of such notables in the area of turbulent boundary layer research as Harrison, Doak, Sharland, Willmarth, and others. The rewults of their mathematical solutions to the noise problem are presented without elaborating on the actual mechanical details of how they were derived. Likewise, the
experimental results from wind tunnel and water tunnel tests recorded in open iiterature are used without explaining the techniques through which the data were obtained. The theoretical equations for determining the acourtic output resulting from vortex shedding and the wake are also presented in a similar manner, with the exception that values for the frequency of the vortex shedding and the boundary layer thicknese at the trailing edge of the airfoll are obtained by original efforta. The frequency of the vortex shedding is obtained through flow visualization techniques provided by the hydraulic analogy of low subsonic two-dimensional flow represented on the water table to steady air flow. The boundary layer thickness is obtained theoretically with a numerical integration of the potential theory velocity distribution over the airfoil. Finally, the data received from the above techniques and experimenta recorded in the iiterature are placed in the appropriate equations in order to predict the relative noise levels generated by the boundary layer, vortex shedding, and wake. These theoretical predictions should make possible the qualitative overview of the Eppler 266 crose seotion as a notwe generator.

## 2. GENERAL THEORY AND GOVERNING EQUATIONS

## a. The Acountic Output of the Boundary Layer

The pressure fluctuations at a far field point in a dilow due to the radiated sound from a ource with an arbitrary surface shape (refer to Figure 12) is proportional to the following integral [16]

$$
\int_{S} \frac{f^{\prime}\left(\bar{x}^{\prime}, t^{\prime}-\frac{\sqrt{\bar{x}-\bar{x}^{\prime} \perp}}{a_{0}}\right)}{\left|\bar{x}-\bar{x}^{\prime}\right|}
$$

where

$$
F\left(\bar{x}^{\prime}, t^{\prime}-\frac{\left|\bar{x}-\bar{x}^{\prime}\right|}{a_{0}}\right)
$$

1a the mource function and

$$
t^{\prime}-\frac{\left\lfloor\bar{x}-\bar{x}^{\prime} \mid\right.}{a_{0}}
$$

is the retarded time between source fluctuation and field point response. With the assumption that the field point is a large distance from the source such that $|x|>D$ and the dimension of the source is small compared to the wavelength of the sound generated, one may replace the retarded times

$$
t^{\prime}-\frac{\left|\bar{x}-\bar{x}^{\prime}\right|}{a_{0}}
$$

by

$$
t^{\prime}-\frac{|\bar{x}|}{a_{0}} .
$$

Hereafter the retarded time

$$
t^{\prime}-\frac{|\bar{x}|}{a_{0}}
$$

Will be referled to as $t$. However, when the source function is random, such as that which exists in a turbulent boundary layer, statiatical methods become necossary in order to determine the pressure at the fiold point. The autocorrelation is the statistical tool available to characterize the function. In the space domaln the autocorrelation compares the source function at a point to the source function at suriounding pointe at a specific time and thus enables the formulation of a mean value for the source function in the neighboghood of the point. This mean is lodmulated from $F\left(\bar{x}^{\prime}, t\right)$ and $F\left(\bar{x}^{\prime}+\wedge \bar{x}, t\right)$ in the fullowint operation:

$$
1 / s \int_{0}^{S} F\left(\bar{x}^{\prime}, t\right) \cdot F^{\prime}\left(\ddot{x}^{\prime}+\bar{x}, t\right) d S^{\prime}=\bar{F}\left(\bar{x}^{\prime}, t\right) \cdot F\left(\bar{x}^{\prime}+n \bar{x}^{\prime}, t\right)
$$

and since $\bar{x}^{\prime \prime}=\bar{x}^{\prime}+\mid \bar{x}$, then

$$
\overline{F\left(\bar{x}^{\prime}, t\right) \cdot F\left(\bar{x}^{\prime}+\Lambda \bar{x}, t\right)}=\overline{F\left(\bar{x}^{\prime}, t\right) \cdot F\left(\bar{x}^{\prime \prime}, t\right)} .
$$

Doak [17] has taken these general concepts and the worls of Curle [18] and Lighthill [19] and established an expression for the source function of a turbulent liuid nn an infinite plane.

From this function Doak derived a formula estimating the jower radiated from a turbulent ooundary layer on a flat plate at jow Mach numbers:

$$
\begin{equation*}
W=\frac{1}{6 \pi f_{o} a_{o}^{3}} \int_{S} \int_{S} \frac{\partial p}{\partial t} \overline{\left(\bar{x}^{\prime}, t\right) \frac{\partial p}{\partial t}\left(\bar{x}^{\prime \prime}, t\right) d s^{\prime}} d s^{\prime \prime} \tag{1}
\end{equation*}
$$

where $\partial p / \partial t\left(\bar{x}^{\prime}, t\right)$ and $\partial p / \partial t\left(\bar{x}^{\prime \prime}, t\right)$ represent the source function at time $t$ at the points described by position vectors $\bar{x}^{\prime}$ and $\bar{x}^{\prime \prime}$, reapectively. The differentials $d S^{\prime}$ and d $S^{\prime \prime}$ indicate the neighborhoods of $x^{\prime}$ and $x^{\prime \prime}$ and are integrated over the entire surface, $S$, of the body.

One of the key assumptions upon which the derivation depends is the presence of low Mach numbers, and/or that the turbulent eddies in the fluid are small compared $t$ the wavelength of the sound generated. Doak demonstrates that, in fact, the Mach number and the ratio of the turbulent eddy size to the acoustic wavelength are identical dimensionaless parameters. Thus the presence of low Mach numbers allows-as Lighthill [19] has pointed out--the, neglect of the different times of the emission of waves from $\bar{x}$ and $\boldsymbol{x}$.

The integrated covairance term

$$
\int_{S} \frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right) \frac{\partial p}{\partial t}\left(\overline{\mathbf{x}}^{\prime \prime}, t\right) d s^{\prime \prime}
$$

may, according to Doak, be expressed in terms of a correlation area such that

$$
\begin{equation*}
\int_{S} \frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right) \frac{\partial p}{\partial t}\left(\bar{x}^{\prime \prime}, t\right) d s^{\prime \prime}=\left\{\frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right)\right\}^{2} s_{c}\left(\bar{x}^{\prime} ; \frac{\partial p}{\partial t}\right) \tag{2}
\end{equation*}
$$

where

$$
\left\{\frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right)\right\}^{2}
$$

denotes a mean square of the source function at a point at time $t$ and where $S_{c}\left(\overline{x^{\prime}}, \partial p / \partial t\right)$ is a correlation area over which this mean square pressure fluctuation has been dexived and depends only on the point in question and the source function at that point.

The equation for power output, $W$, may now be written:

$$
\begin{equation*}
W=\frac{1}{6 \pi \rho_{0} a_{o}^{3}} \int_{S}\left\{\frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right)\right\}^{2} S_{c}\left(\bar{x}^{\prime} ; \frac{\partial p}{\partial t}\right) d S^{\prime} \tag{3}
\end{equation*}
$$

According to Sharland [20]

$$
\left\{\frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right)\right\}
$$

and $s_{c}$ as a function of $\left(\bar{x}^{\prime} ; \frac{\partial p}{\partial t}\right)$ may be replaced by

$$
\left[\overline{p\left(\bar{x}^{\prime}, t\right)}\right]^{2} \cdot \omega^{2} \cdot s_{c}\left(\bar{x}^{\prime}, p\right)
$$

where $\omega$ is the characteristic frequency of the pressure fluctuation. Substitution into Equation (3) for

$$
\left\{\frac{\partial p}{\partial t}\left(\bar{x}^{\prime}, t\right)\right\}^{2} s_{c}\left(\bar{x}^{\prime}, \frac{\partial p}{\partial t}\right)
$$

renders the form

$$
\begin{equation*}
W \% \frac{1}{6 \pi p_{0} a_{0}^{3}} \int_{s}\left[p\left(\bar{x}^{\prime}, t\right)\right]^{2} \cdot \omega^{2} \cdot s_{c}\left(\bar{x}^{\prime}, r\right) d S^{\prime} \tag{4}
\end{equation*}
$$

Since for flight test, the microphone will be so situated that it will predominantly receive the noise generated from the bottom of the airfoil, the assumption will be made that the airfoil is nearly symmetrical and thus the bottom surface area is equal to one-half of the total surface area. Therefore, the sound radiated by the turbulent boundary layer into the lower semi-space should be one half of Equation (4):

$$
w \cong \frac{1}{12 \pi \rho_{0} a_{0}^{3}} \int_{S}\left[p\left(\bar{x}^{\prime}, t\right)\right]^{2} \cdot \omega^{2} \cdot s_{c}\left(\bar{x}^{\prime}, p\right) d S
$$

Here $\overline{\left[p\left(\bar{x}^{\prime}, t\right)\right]^{2}}$ may be interpreted as the mean square pressure fluctuation of the turbulent boundary layer. According to Harrison [21], the quantity

$$
\left[p\left(\bar{x}^{\prime}, t\right)\right], \frac{1}{2} P_{o} U_{o}^{2}
$$

is equal to a constant for a strouhal number, based on the boundaryllayer displacement thickness $\AA^{*}$, less than or equal to 0.2. Harrison obtained a value of $9.5 \times 10^{-3}$ for this constant
which is high according to the measurements of Willmarth [22] and others. A generally accepted value for this constant in the Mach number range of this study is $6 \times 10^{-3}$ and $w 111$ be utilized in the appropriate equations. This meaps that $(\bar{p})^{2}=36 \times 10^{-6} q^{2}$, where $q$ is the dynamic pressure, $1 / 2 \mathrm{PO}_{0}^{2}$.

According to Bull and Willis [23], the value for $\mathrm{w}^{2} \cdot \mathrm{~S}_{\mathrm{C}}$ should be of the order $1 / 2 \mathrm{U}_{0}^{2}[20]$. Substituting these values Into Equation (5) gives:

$$
\begin{equation*}
W=1.2 \times 10^{-7} \frac{P_{0}^{U_{0}^{6}}}{a_{0}^{3}} \int_{S} d S^{\prime}=1.2 \times 10^{-7} \frac{P_{0}^{U_{o}^{6}}}{a_{0}^{3}} S_{b 1} \tag{6}
\end{equation*}
$$

Where $S_{b l}$ is the lower surface of the wing covered by the turbulent boundary layer.

This equation for estimating the acoustic power output of the turbulent boundary layer on a flat plate should be appli.cable to the lower surface of the Eppler 286 airfoil. Doak [17] states that a surface may be considered locally flat if the radius of curvature of the surface is large compared to the wavelength of the gound generated by the turbulent boundary layer, Since the lower surface of the airfoil could be approximated by a circle with a large radius, the surface may be considered locally flat except possibly at low frequencies of 100 cycles per second and below.

## b. Lift Fluctuation Caused by Vortex Shedding as a Noise Source

The boundary layer noise, according to Sharland [20], will be rather small in comparison to a larger order noise source created by the fact that for a plate of finite aize, the "larger scale vorticity in the boundary layers on the two sides of the plate is not instantaneously symmetric." He contende that vortex shedding will occur at the trailing edge and, therefore, iff fluctuations are present. Sharland atates that Equation (5) may be simplified when one realizes that the pressure fluctuations may be thought of in terms of local lift fluctuations per unit area. This idea relies on the assumption that the normal pressure fluctuations on the surface are large compared to the tangential stresses. Therefore,

$$
p\left(\bar{x}^{\prime}, t\right)=C_{L} q=C_{L}\left(\bar{x}^{\prime}, t\right) \frac{1}{2} \rho_{0} U_{0}^{2}
$$

and

$$
\frac{\partial \mathrm{p}}{\partial t}\left(\bar{x}^{\prime}, t\right)=\frac{\partial C_{L}}{\partial t} \frac{1}{2} \rho_{0} u_{0}^{2}
$$

assuming the fluctuating component of the velocity is small when compared to the mean velocity. Then

$$
\left[\overline{\frac{\partial \mathrm{p}}{\partial t}\left(\bar{x}^{\prime}, t\right)}\right]^{2}
$$

may then be written

$$
\frac{1}{4} \rho_{o}^{2} u_{o}^{4} \overline{\partial C_{L}} \frac{\left.\left.\bar{x}^{\prime}, t\right)\right]^{2}}{}
$$

Substituting back into Equation (5) resulte in the following:

$$
\begin{equation*}
W=\frac{P_{0}}{48 \pi a_{0}^{3}} \int_{S} U_{0}^{4}\left[\frac{\partial C_{L}}{\partial t}\left(\bar{x}^{\prime}, t\right)\right]^{2} S_{c}\left(\bar{x}^{\prime}, \frac{\partial C_{L}}{\partial t}\right) d S^{\prime} \tag{7}
\end{equation*}
$$

Assuming then that

$$
\left[\frac{\partial C_{L}}{\partial t}\left(\bar{x}^{\prime}, t\right)\right]^{2} \cdot S_{c}\left(\bar{x}^{\prime}, \frac{\left.\partial C_{L_{L}}\right)}{\partial t}=\overline{\left[C_{L}\left(\bar{x}^{\prime}, t\right)\right]^{2}} \cdot p^{2} \cdot s_{c}\left(\bar{x}^{\prime}, C_{L}\right)\right.
$$

where $f$ is the characteristic frequency of the lift fluctuations and

$$
\overline{\left[C_{L}\left(\bar{x}^{\prime}, t\right)\right]^{2}}
$$

the mean square of the fluctuation lift coefficient, then

$$
\begin{equation*}
w=\frac{P_{0}}{48 \pi a_{0}^{3}} \int_{S} U_{0}^{4} f^{2}\left[C_{L}\left(\bar{x}^{\prime}, t\right)\right]^{2} \cdot s_{c}\left(\bar{x}^{\prime}, C_{L}\right) d S^{\prime} \tag{8}
\end{equation*}
$$

The description of this integral in terms of flow parameters necessitates a cextain degree of knowledge about the $f l o w$ mechanics by which the lift fluctuations are produced. Sharland [20] contends that the source strength can be ascertained by considering the order of magnitude of the parameters involved. He atates that the lift fluctuations can be related to time fluctuations in the boundary layer thickness at the trailing edge of the airfoil. On this basis, it in muggested that the root-mean-square of the fluctuating lift coefficient should be of the order of $-1 / 5$ power of the Reynolds number [20]. The frequency of the lift fluctuations mould be approximately the same as the frequency of the vortex shedding at the trailing edge. Sharland [20] contends that the correlation area as a function of ( $\bar{x}^{\prime}, C_{L}$ ) should be "governed by the size of the larger eddies at the trailing edge."

Thus if

$$
\left[C_{L}\left(\bar{x}^{1}, t\right)\right]^{2} \simeq R e^{-0.4},^{*}
$$

then

$$
\begin{equation*}
W=\frac{P_{0}}{48 \pi a_{0}^{3}} \int_{S} U_{0}^{4} \cdot P^{2} \cdot(R e)^{-0.4} S_{C}\left(x^{\prime}, C_{L}\right) d S^{\prime} \tag{9}
\end{equation*}
$$

Since $\left[U_{0}^{4} f^{2}(R e)^{-0.4} S_{C}\left(x^{\prime}, C_{L}\right)\right]$ is asaumed convtant during the integration,

$$
\begin{equation*}
w \simeq \frac{P_{0}}{48 \pi a_{0}^{3}} U_{0}^{4} \cdot f^{2} \cdot(R e)^{-0.4} S_{c} \cdot S_{V E} \tag{10}
\end{equation*}
$$

where $S_{\text {gs }}$ is the entive lower murface of the wing.
Since $f$, the characteriatic center firequency of vortex shedding, is not attainable from existing theory for otreamilne bodien, it muet be ascertained experimentally in Section 3 of this jeport an experiment im deacribed which utilizes the flow viaualization cheme provided by a water table in order to eatablish this shedding frequency. The correlation area $\mathrm{B}_{\mathrm{c}}$ may be eatimated if the larger eddies at the trailing edge are assumed to be approximately the same mize as the boundary layer

[^1]thickness at the trailing edge. Since the experimental moasuroment of the actual boundary layer thickness is extremely difficult. a theoretical approximation of this thickeness is necessary. The method of obtaining this thickenss is discussed in Section 4 of this report.
c. The Acoustic Output of the Wake

Since no direct data exists for the estimation of the nolse produced by a wake, qualitative efforts to predict the noise generated by this sound source have been made. Franken [24] has argued that the wake of a subsonic vehicie is a region of high shearing forces, flow separation, and turbulence. Franken notes that these are the identical flow characteristics which occur in the jet stream of a gas reaction motor. Thus he makes an analogy between the two in qualitative terme. He observes that in jet radiation the velocity profile ia directed downstream away from the jet engine and the most intenae mound radiation is likewies downstream. However, aince in wake turbulence the velocity profile is directed upatream towards the moving vehiole, he reasons that likewise the most intense sound radiation should be directed upatream as depicted in Figure 13.

Franken contenda that the acoustio output of the wake is a small fraction of the power contained by the mechanical system generating the wako. The power of this syatem may be estimated by the equation

$$
W=\eta D U_{0}
$$

Where $D$ is the drag, $U_{0}$ is the free stream velocity (forward apeed of the vehicle), and $\eta$ ia a factor of proportionality. The drag for a aliplane may be written in terme of an equivalent parasitic area $\mathrm{I}_{\mathrm{p}}$ and the dynamic pressure $q$ [3].

$$
D=f_{p} q=f_{p}\left(\frac{1}{2} \rho_{0} U_{o}^{2}\right)
$$

where

$$
\begin{equation*}
f_{p}=\frac{W}{q(G P R)}\left[1-\frac{G P R}{q}\left(\frac{W}{S}\right)\left(0.01158+\frac{0.3185}{A R}\right)\right] \tag{11}
\end{equation*}
$$

and GPR is the glide path ratio at a specific velocity, $\bar{W}$ i:s
the total weight of the aircraft and occupant, $S$ is the total wing surface area, and $A R$ is the aspect ratio. Then

$$
W=\eta U_{0}\left(\frac{1}{2} \Gamma_{0} U_{0}^{2}\right) f_{p}
$$

According to Franken, $\eta$, based on the analogy with subsonic jets, is assumed to be of the order $10^{-4} M_{0}^{5}$ where $M_{0}$ is the free stream Mach number. Substituting for $\eta$ renders the inal form for the acoustic energy of the wake:

$$
\begin{equation*}
W \cong 10^{-4} M_{0}^{5} U_{0}\left(\frac{1}{2} \rho_{0} U_{0}^{2}\right) f_{p} \tag{12}
\end{equation*}
$$

Intereatingly enough, this producea an eighth power velocity dependence for the acountic power output from a wake.
d. Frequency Spectra

## (1) Boundary Layex

The turbulent boundary layer frequency pectrum, according to Skurdzyk and Haddle [25], can be asmumed to arime from a Gaumeian (a random diatribution function) energy apectrum. The pressure apectrum is then derived by a series of integrations and the result resemblea a Gaussian spectral distribution with a pronounced drop off above the frequency described by $f=U_{0} \mathcal{I N}_{s}$ where $U_{0}$ is the free stream velocity and $f_{s}$ is the smallest boundary layer thicknem.

From the derivation of the low frequency apectrum, a patch of turbulence may be thought of as being a pulse with a diameter equal to the width of the turbulent patoh. If this patch is considered to have a width approximately equal to the thickneas of the boundary layer, then the opectrum should turn out to be conetant up to apace wavelength approximately equal to the largent boundary layer thickneas and then decreame as (sin $x / x$ ). The high frequency mectrum in upposed to behave as specified by the "equilibrium laws of turbulence," of which the Kolmogorov law predicts that at high frequencies the energy apectrum decreases inversely as $3 / 2$ the power of the mace wavelength. Thus the apectral denwity may be expected to be nearly conatant at low frequencien from $U_{0} / \hbar 1$ (where $\mathcal{F}_{1} 1$ denoter the largent boundary layer thicknema) up to a hich frequency of $\mathrm{U}_{\mathrm{o}} / \mathrm{s}$ a and then decreae approximately an the inverie of the $3 / 2$ power of
of the wavolength. Thorofore, the spectura density of the boundary layer noise in the audible range of the human car would be oxpected to be the result of broad band type notse.

A sizable amount of consideration has been given to the solid stationary circular cylinder in a flow [26]. The kenemal findings of these investigations show that the vortex sheddins from the cylinder due to the viscosity of the flow medium creates an unsteady lift force on the cylinder, causing an opposite circulation and hence a lift. This liuctuating lift Henerates a distinctive sound field perpendicular to the flow
 the dianeter of the oylinder. A weaker sound field of twice the frequency ut the noise generated by the fluotuating lift and parallel to the flow is produced by the fluctuating drag rosultm inf firom the lift variation as ghown in Figure 14. For un aid". foll in a flow, one would expect a similar acourtice phomomenon: the most intense sound, due to the fluctuating ift, being radiated perpondicular to tho flow white a much veaker sound field, due to the fluctuating drag being spherically radiatod parallel to tho flow, At higher angles of attrok the sound spectrum produced by the aireoil should conter about a precominant charncteristic frequoncy, while at lower ankjos of attack, as more randomeses appears in the fiuctuatjons, a babader band notse should appont.
(i3) Wake
'Tho wake of the cylinder at low Reynolds numboxe (300 to 10, (00) results in the rafularly spacod vortices of tho karman sheot. At hifher Reynolds numbers the Karman sheot diselpates rapidly and the orderly propression of individual vortices disappeaz. According to highthill [1U], the velocity lluctuntions In the turbulent wake produce a field of quadrapolo nolse gouncer which radiate broad band noise whooe intensity varios as U 8 , $1, i k e$ wise, the Irequency spectrum of the noise generated by the movinf airfoil should rellect the presence of wide band noiso wilh ho paxticular frequency prodominating.
e. Comparison of the Equation for the Acoustic Output of the Aerodynamic Nois'a Sources

Consider the ratio of the acoustic output due to lift fluctuations caused by vortex shedding to the acoustic output of the boundary layer, 1.e., $W_{\text {Vs }} / W_{b l}$ :

$$
\begin{gather*}
W_{v a}=\frac{\rho_{0}}{48 \pi a_{0}^{3}} U_{o}^{4} f^{2}(R e)^{-0.4} S_{c} S_{v a}  \tag{10}\\
W_{b l}=1.2 \times 10^{-7} \frac{\rho_{0} U_{o}^{6}}{a_{0}^{3}} s_{b l} \tag{6}
\end{gather*}
$$

Now forming $W_{V B} / W_{b l}{ }^{\prime}$

$$
\begin{equation*}
W_{v a} / W_{b 1}=\frac{17.4 \times 10^{4}}{\pi}\left\{\frac{f}{U_{0}}\right\}^{2} S_{c}\left(R e^{-0.4}\right) \frac{S_{V_{B}}}{S_{b 1}} \tag{13}
\end{equation*}
$$

$S_{y s}$, the mound radiating area due to the $11 f t$ fluctuations, is the' entire area of the lower portion of the wing, $S$. $S_{b l}$ is that area of the bottom of the wing covered by the turbulent boundary layer and in equal to $S\left(1-x_{1} / C\right)$ where $x_{1} / \bar{c}$ is the traneition point of the boundery layer from laminar to turbulent flow expremed am a deoimal iraotion of the mean aerodyanmic chord $\overline{0}$. Therefore, $S_{V E} / S_{b 1}=1 /\left(1-x_{1} / \tau\right)$.

If the correlation area in ancumed to be the eize of the ladger turbulent eddies at the trailing edge of the airfoil $[20], S_{c}$ can then be approximatod by a ciroular area, the width of which im of the mame scale an the trailing edge boundary layer thickneme, 5 , on the lower mide of the aimfoll:

$$
s_{c}=\pi(5 / 2)^{2}=\frac{\pi}{4} 5^{2} .
$$

Substituting in Equation (13) for $S_{V a} / S_{b 1}$ and $S_{C}$ renderm the final form,

$$
\begin{equation*}
W_{v=} / W_{b 1}=\frac{4.35 \times 10^{4}}{\left(1-x_{1} / \bar{c}\right)}\left\{\frac{1}{U_{0}}\right\}^{2} n^{2}(R e)^{-0.4} \tag{14}
\end{equation*}
$$

Now conmider the ratio of the ooumtic output due to the ifit fluctuation caused by vortex medding to the acourtic output of the wake, $1,0 . W_{v a} / W_{W}$ :

$$
\begin{equation*}
W_{v=}=\frac{P_{0}}{48_{\pi a_{0}^{3}}} U_{0}^{4} f^{2}(R \theta)^{-0.4} g_{c} E_{v a} \tag{10}
\end{equation*}
$$

$$
\begin{equation*}
W_{W}=10^{-4} M_{0}^{5} U_{0}\left(\frac{1}{2} f_{0} U_{o}^{2}\right) f_{p} \tag{12}
\end{equation*}
$$

Now forming $W_{V B} / W_{W}$ where $S_{c}$ is again considered to be $\pi / 4 i^{2}$,

$$
\begin{equation*}
W_{V B} / W_{W}=1.04 \times 10^{2}\left\{\frac{f}{U_{0}}\right\}^{2} \frac{(R \theta)^{-0.4}}{M_{0}^{2}} n^{2} \frac{S}{f_{p}} \tag{1.5}
\end{equation*}
$$

where $f_{p}$ is the parasitic aroa, As proviously mentioned, fipr for a sailplane may be expressed by Equation (ll). The gilde phth ratio (GPR) of the Standard Austria SH1 is 32:1 at a moed of 57 milem per hour. The welight of the pilot and aircraft for the preliminary filight test was approximately 715 pounds. Using these values and an aspect ratio of 16.7 in Equation (11) rendex. an fy of 1.12 equare foet. Subatituting the area of the wing for $S$ ( 146 quace peot) and 1.12 square feet for $f_{p}$, then

$$
\begin{equation*}
W_{V} \neq W_{W}=1.36 \times 10^{4} \frac{(R e)^{-0.4}}{M_{0}^{2}}\left\{\frac{f}{U_{0}}\right\}^{2} \pi^{2} \tag{16}
\end{equation*}
$$

The cqmanon unknown parametors for both Wva/W and Wve/Wipl are $\left\{t / U_{0}\right\}^{2}$ and $r_{1}^{2}$. Once theme parametert are known then the relative btuengths of the aerodynamic notme mources may be qualitatively Judged.

## 3. THE WATER TABLE EXPERIMENT

a. Iheory Bohind the Experimental Approach

The purpose of the water table experiment is to ascertain the quantity $\left\{f / U_{o}\right\}^{2}$ by entabliahing the characteristic from quenciem of vortex shedding at the trailing edge of the airfoll at various angles of attack. To accomplish this task the flow pattern about the sailplano wing is simulated by the twodimenslonal llow of the water table. The characteristic frequency of the vortex ehedding at the twailing edye of the aidfoll in water may then he correlated to the frequency in the air through the Strounal number. The Strouhal number is a dimensionless parameter which demcribes flow periodicity. The number is defined by $S t \approx P 1 / U_{0}$ where $f$ is the frequency of the flow, 1 is a characteristic length, and Uo is the free stream velocity. For thim etudy the characteristic length is the boundary layer dimplacement thickneme, $F^{*}$, at the tiailing edge of the airioil. By matching the Strouhal number in the
air and on the water table, the frequencies may be correlated:

$$
\frac{\mathbf{f}_{a 1 r^{8}}^{*}}{U_{a 1 r}}=\frac{\mathbf{f}_{\text {water }}^{8}}{\delta^{*}} \text { water }
$$

11

$$
\bar{n}=\frac{\varepsilon^{*} \text { water }}{8_{\text {air }}^{*}} \cdot \frac{1}{U_{\text {water }}}
$$

then

$$
f_{a 1 r}=\bar{n} f_{\text {water }} U_{a i r} .
$$

In thim manner the Strouhal number compenmates for the differencen in velooity and in the Reynoldm number of the two flow . The mianing parameter $\left\{f_{a i p} / U_{a i p}\right\}^{2}$ for the comparison of the acountic output of the aerodynamic noime mources 1 s thus equal to $\left\{\boldsymbol{n} \mathrm{f}_{\mathrm{water}}\right\}^{2}$. Hence, the purpone of the water table in to provide a flow regime from which the irequency of the vortex ahedding at the trailing edge of the airfoll may be correlated to the frequency of the vortex ahedding or lift fluctuation in uctual filght at different velocities. The mathod of obtaining valuea for $\mathrm{F}^{*}$ water/K*air in discumed in part (d) of thim mection.

The determination of fair requirem that the angie of attack of the airfoil on the water table be varied am one attempte to mimulate different velocities in the air. Ihum the determination of how variations in angle of attack in actual flight are related to velocity changen is important to the correlation of the periodic flow dieturbancea at the trailing edge to the acourtic output of the airfoll.

It ia pomaible to relate angle of attack to velocity through the $11 f t$ coefficient. If one knows how $C_{L}$ varieu with angle of attack, then the velocily can be obtained through the equation:

$$
L=C_{L} \frac{1}{2} \rho U_{o}^{2} s .
$$

Aseuming lift, (L), equal to weight, (W), then

$$
C_{L}=\frac{2 W}{\rho U_{0}^{2} s}
$$

'Thus il' ome knows the velocity and how $C_{L}$ vaides with angle ol athack, then it is known how the elfective natio of nttack varios with velocity. Sinee the water table fives only a two-dimensional model of the flow regime and the effect of induced velocities due to tho vortex syatem of the threondimensional wing is not prosent, then the offective anyle of attack is equal to the feometide dinfle of attack. For this experiment table i definos the ankle of attack nocegsary on the water table to simulate a certain ungle of attack in tho air associated with a specific velocity. The dorivation ol the $C_{1}$ versus angle of attack curve givon in this table was developed experimentaliy from the specified fildor in filight.

## b. Nechanics of the Experiment

## (1) The Water Tnble

For this experiment, the water table located din the hydrow dynamics laboratoly of Tho Univeraity of Tennessee Space Inetitute was utilizod. The main table assembly consists of a 3 foot, s lach by 6 foot tost section mado of 0.75 inch thick ploxiglass, 5 finch high walle, a sotilimg and a dischargo tank, and a 2 dinch thick fibious mesh located in the upstiream odye of the test soction (roter to Figure 1B). The fibrous mash smoothed out the flow and removed bubbles which formed on the water's surface in the gettilng tank belore thoy could be transpostod downstream. Reciroulation of the water from the dischadge tank to the setitilng tank 1 saccomplishod by a Weimman wator pump with a horisepowor oloctric motor. The maximum capacity of tho pump is GBO fallons per minuto. The water table astembly is attached to an iron framo which has built-in floor serews that enable the adjustment of the slupe of the table. Water for the table is suppliod by $\quad 2$ inch water pipe located just above the discharge talk. Once the proped level of water is obtalned on the test section, the llow may be adjusted by changinf thu flow rate throurh the rocirculation pump. Ihis adjustmont it accomplished through the uso ol throttling valvos on the pump.
(2) The Model

The model was constacted lyom a large soction of 2 inch thick plexiglass. A patterll of the Eppled $2 G 6$ ciosas section was produced from a 4-1/2 inch seale drawing which was photoriaphically enlarged to a 27 Lach chord longth. This chord length $i$ os approximately the upper limit lor a mudel on the water table and atill preserve the flow without wall interference affectink the near lield around the model. The larye chord length was chosen in order to obtain the closest posesble Reynolds number to those expected in nctual filight. The fattern was then placed on the plexiglase and cut, ground, and sandod to amooth close tolorance finish. A 7 inch onlargement of the $4-1 / 2$ inch sodjo
drawing is included in Figure 16. The inner unshaded airfoil is the NACA $652^{414}$ which was employed by the Standard Austria $s$ 1963. The outer airioil is the Eppler 266 which is utilized by the later Standard Austria SH1 1964.

## (3) Lighting Technique

A light source consisting of a bank of aix mandard fluorescent tubes was placed underneath the water table in ordex to illuminate the teat section. A large mheet of translucent tracing paper was taped underneath the plexigiame tewt eoction. Thim paper gerved two purposes: first, it diffumed the iight, thus producing, a good background for picture taking; mecond, a reference grid and lines to demignate mpecificianglea of attack orientations were drawn on the paper. The model could then be aligned and realigned ovar these ilnem, thum providing a aimple, accurate method of varying the ancie of attack. The grid proved to be of great value when reviewing the piotures of the model in the flow by pxoviding a reference by which the mole of the dimturbanoes oreated in the flow might be jurged.

## (4) The Dye

A dye wan umed in order to vinualize the atreamlines clome to the model. The dye for thí experiment wan potaneium permangamate cryatala dimmolved in water. This molution is a dark red liquid which was easily photographed when introduced into the flow. Once the model wam properiy aliened at the corroct angle of attack and poisitioned in the center of the temt section approximately 14 to 18 inchem downatream of the fibrous menh (in order to minimize wall effects and to oapitalize on the moothest possible flow), mooth, stendy flow of the desired velocity achieved, and the demired water level attained (approximately $1-7 / 8$ inchos at the leading edge of the model and $1-15 / 16$ incher at the trailing edye), then the dye wam introduced into the flow. The dye wai allowed to enter perpendicular to the ilow in a steady atream through mall rubber tubing connected to a gravity feed dye dimpenmer located high on the laboratory wall. When an overall view of the $f$ low near the model wan desired, the dye was allowed to onter the water about 2 inches in front of the leading edge of the model in much a poaition that the dye would pane through the etacnation point and then spread out over both the top and bottom murfices of the model. If juat the flow at the trailing edge wan to be viewed, then the dye was allowed to enter the flow next to the lower aurface of the model, approximately 3 inchen upatream of the trailing edge.

The two cameras used in this experiment wero the Minoltin K7 8imm for motion pictures and the 35 mm Agfa Karat for still photographs. There were two basic photographic methods utilizing these cameras which were especially adaptable to the water table. Although the table is equipped with a large 5 foet by 6 feet mirror mounted at a 45 degree angle over the test section, the method of simply mounting the cameras on a tripod and taking photographs directly into the large mirror was not applicable atince much amallex focal lengths were required for close-up lenses, focal lengths of approximately 12 inches and smallen bocame necesadry. Focusing directly into the large mircor could not produce focal lengths less than 36 inches. To solve tho problein, two pieces of $1 / 2$ foot by 8 feet by $1 / 2$ inch thick blexiglass were placed $7-1 / 2$ inches above the test section. Jho camaras were mounted to ono while a 4 inch by 6 inch mirror was mounted at a 45 degree anglo on the other (refer to figure 1.7 ). For atill photosraphs the 35 mm camera was then focused into tho mirrol with the focal length being varied by changing the disw tance from the midror to the camera. The entire lenyth of the model could be photographed in detail by moving the mixror and camera from position to position alomg the plexiglass supports and adfusting the focal length.

For motion pictures of the $f 10 w$, the Minolta $K 7$ was mounted to the plaxiglass support by a screw at the base of the hand grip and tilted to a position perpendioular to the area of tho flow dosired to be photographed. A support cable from the hand krip was then attached to the top of the large mirror mounted above the table (not shown in Figure 17). With the close-up lens attached, the camera could be foucsed manuatily by lookinf: throuph the viewfinder and adjusting the "zoom" lens, Mhis camera arrangement was especially advantageous because it allowed for the simple modification of the field of view around the model while still maintaining excellent support of the camera.
(6) Measurement of the Water Velocity

In order to obtain the water velocity, a simple but relim able method is apparent. Directly above the test section a muter wick was placed parallel. to the flow. Small piecen of paper were then placed on the water surface upatieam of the meter atiok and allowed to be transported duwnatream by tho Hlow. The time the paper took to traverge the length ol the meter gtick was timed with a stop watch. The times of several runs at the same throttle setting were then averaged and the result divided into the one meter distance to establish tho velocity. This approach seemed quite satisfactory since tho run times generally fell in the near vicinity of one another.

The Reynolds number is described by three parameters: $U_{0}$ (the free etream velocity), 1 (the characteristic length of the model, and $v$ (the kinematic viscosity of the fluid). In this experiment, al. 1 three parameters could be varied. Since Re $=U_{0} l / v$, increasing $U_{0}$ and 1 will increase Re while increasing $v$ will. decrease Re. The Reynolds number may be modified through the free stream velocity by adjusting the throttilng valves on the recirculation pump and by increasing or decreasing the table slope with the floor screws. The length of the model may also affect the Reynolds number. In addition to the main 27 Inch model, an 18 inch and an 11-1/2 inch model were constructed. By placing these different models on the table, the Reynolds number may be modified simply and quickly, finally, since kinematic viscosity is a function of temperature, the action of the pump's stator blades against the water will increase the temperature of the water with time and thus modify the Reynolds number. Since inttle control is possible over this variable, one can only record the increase of the temperature with a thermometer and thus note this automatic change in the Reynolds number.

## c. The Sti.11 Photograph Experiment

The purpose of the still photograph experiment is to obtain the transition point of the boundary layer from laminar to turbulent flow for the 27 inch model on the water table. This information will then be used to obtain theoretically the boundary layer displacement thickness at the trailing edge of the model in the water flow.

For this experiment, $t$, angle of attack of the model was varied from -2.3 to 10.4 degrees. Different water velocities and water temperaturea were used to vary the Reynolds number on the table. Both black and white and color pictures were taken of the model. Some of the best photographs of the transition region resulted from color pictures of the model when dye was injected into the boundary layer by a hypodermic syringe. Even though the Reynolds number was varied between $1.38 \times 10^{5}$ and $2.56 \times 10^{5}$, for angles of attack between -2.3 and 5.9 degrees, the transition region remained at approximately 21 inches downstream from the leading edge of the 27 inch model. This figure represents a transition of about 0.78 of the chord length.
d. The Motion Picture Experiment

The frequency of the vortex shedding at the trailing edge of the airfoll was studied for angles of attack of $\mathbf{- 2 . 3 , - 1 . 9 \text { , } , ~ ( 1 )}$
$2.8,4.25$, and 5.9 degrees. The close-up lens on the 8 mm camern allowed the inspection of approximately a 3 inch diameter aroa inanediately behind the crailing edge. At the higher angles of attack the pattern of shedding vortices is quite pronounced and dissipates relatively slowly when compared to the smaller vortices which dissipate immediately at the two negative angles of attack. However, even at 5.9 degrees, the vortices have generally decayed beyond recognition as diatinctive eddies by the time they reach the limit of the camera's field of view. For the three positive angles of attack the vortices ade approximately $1 / 4$ to $3 / 8$ inch in diameter when first formed downstream of the trailing edge, but soon decay to larger and rather formless swirls of approximately 1 inch in diameter within the time it takes for the vortices to be conveyed $1 / 10$ to $1 / 12$ foot downstream. The vortices at the two negative angles of attack are smaller and their cycle between creation and decay are much more didficult to follow. These vortices dissipate very rapidly and are not nearly as well formed. This 1 s to be expected, however, since these vortices result from a flow duncture at the trailing odge which is far more symmetric than those for the higher angles of attack where separation occurs on the upper surface and creeps toward the leading edge as the stall angle is approached. Thus at the positive angles uf attack the counterclockwise rolling up of the flow from the bottom surface of the airfoil just downstream of the trailing edge is quite diatinctive, while at the negative angles of attack this formang of the vortices ds not nearly as visible.

The data presented here $1 s$ the result of over 300 feet of 8 mm film being placed on a microifilm reader and inspected frame by frame to arrive at the various frequencies for the stated angles of attack. The results of these efforts are presented in Table II and Figure 18.

The frequencies of the vortex shedding for the same anglo of attack at the velocities shown in Table $I I$ appear to vary by the retio of the velocities. Logically, one would expect this correlation to exlet alince the boundary layer structure lor the various velocities are quite nearly identical since the Reynolds numbers are nearly matched. Therefore, by matching the struuhal number, then

$$
\frac{{ }^{\mathrm{I}} \mathrm{U}_{1}^{\delta}{ }^{*} \mathrm{U}_{1}}{\mathrm{U}_{1}}=\frac{{ }^{\mathrm{f}} \mathrm{U}_{2}{ }^{\delta}{ }^{*} \mathrm{U}_{2}}{\mathrm{U}_{2}}
$$

and since

$$
\kappa_{\mathrm{U}_{1}}^{*} \simeq \kappa_{\mathrm{U}_{2}}^{*}
$$

then

$$
\mathrm{i}_{\mathrm{U}_{1}}=\mathrm{f}_{\mathrm{U}_{2}} \frac{\mathrm{U}_{1}}{\mathrm{U}_{2}}
$$

where the mbecripta $U_{1}$ and $U_{2}$ denote the reapective values associated with the two velocitien.

## 4. BOUNDARY LAYER THICKNESS

This mection deals with the method by which the boundary layer thicknesmes at the trailing edge of the Eppler 266 aixfoil can be theoretically determined with the aid of the computer. This study is concerned only with the boundary layer that is formed on the lower surface of the airfoil; however, the method im applicable to both top and bottom eurfaces. Once the computer program is established, this method allows for the varym ing of the Reynolds number, lift coefficient, axid flow medium (apecificaliy air and water).

The basic formula for computing the various boundary layer thickneasen is found in Schlichting's Boundary Layer Theory [27] in the chapter dealing with the incompreselble two-dimensional turbulent boundary layer with a pressure gradient. This equation deal.s with the momentum thickness and from it the boundary thickneas and diaplacement thicknems may be somputed if the velocity profile is assumed to be of the form $U / U_{\infty} m(y / \delta) 1 / n$, known as a power law, In this case of the power law, the boundary layer thickness may be expressed in the following ratios:

$$
\delta^{*} / \pi=\frac{1}{1+n}, \delta_{2} / n=\frac{n}{(1+n)(2+n)}
$$

where $5^{*}$ is the boundary layer displacement thickness, $\delta$ f.s the boundary layer thickness, and $\mathrm{F}_{2}$ is the momentum thickness.

Schlichting [27] provides the following formula for the momentum thickness which is the basis for the various desired thicknesses:

$$
\begin{equation*}
\delta_{2} / l=\left(\frac{U}{U_{\infty}}\right)^{-3}\left\{c_{1}^{*}+\left(\frac{c_{f}}{2}\right)^{(n+1) / n} \int_{x_{t} / l}^{x / l}\left(\frac{U}{U_{\infty}}\right)^{3+2 / n} d\left(\frac{x}{l}\right)\right\}^{n /(i+n)} \tag{17}
\end{equation*}
$$

Here (U/Us) denotes the potential theory velocity ratio which
varies over the surface of the airfoil. But in Equation (17),

$$
\begin{equation*}
\mathbf{c}_{\left(-\frac{1}{2}\right)}(n+1) / n=A\left(-\frac{U_{x} \mathcal{L}}{v}\right)^{-1 / n}, \tag{1+3}
\end{equation*}
$$

where A is a constant established by the $n$ which is chosen to represent the boundary layer velocity profile (assuming a constant profile along the entire length of the turbulent boundary layer) and

$$
\begin{equation*}
\mathrm{C}_{1}^{*}=\left[\frac { 1 } { 2 } \mathrm { c } _ { \mathrm { f } _ { \ell } } \left(\int_{0}^{\mathrm{x}_{\mathrm{t}} / \ell}\left(\frac{\mathrm{U}}{\mathrm{U}_{\mathrm{c}}}\right)^{5} \mathrm{~d}\left(\frac{\mathrm{x}}{\mathrm{f}}\right)^{1 / 2} \mathrm{l}(n+1) / \mathrm{n}\right.\right. \tag{19}
\end{equation*}
$$

where cfle is the coefficient of skin friction for a flat plate at zero angle of attack with a Reynolds number (Re) defined by Uwl/v, However,

$$
\begin{equation*}
c_{f_{l}}=\frac{1.328}{\sqrt{R e}}+\frac{2.326}{R e} \tag{20}
\end{equation*}
$$

and $x_{t} / \mathcal{i}$ is the transition point from laminar to turbulent flow as a nondimensional fraction of the chord length.

For turbulent flow $4<n<6$, where $n m 4$ is valid for low Reynolds numbera and $n=6$ is valid for high Reynolds numbere. The corresponding constant $A$ for $n=4180.016$ and for $n=6$ is 0.0076 . Choosing $n=0$ for the Reynolds numbers in the one to three million range and linearly interpolating the constant $A$ between the values far $n=4$ and $\mathrm{r}=6$ produces an $\mathrm{A}=0.01 \mathrm{l}=0$. Jox the Reynolds numbers on the water table, $n=4$ is chosen with an $A=0.016$.

The remaining values which must be known to establish the dosired boundary layer thicknesses are the velocity distribution along the bottom of the ailefoll and the transition point lyom laminar to turbulent ilow in the boundary layer. First, consider the velocity distribution along the bottom surface ot the airioll.

The velocity ratio, $U / U_{w, ~ d i s t r i b u t i o n ~ i s ~ c o m p o s e d ~ o f ~ t h r e o ~}^{\text {a }}$ parte. Firet is the U/U $\mathrm{U}_{\mathrm{s}}$ distribution over the basic thickness form at zero angle of attark. This will be denoted by $\mathrm{U}_{\mathrm{o}} / \mathrm{U}_{\mathrm{u}}$. Any thickness ratio may be obtained from the nerrest thicknest form by inear scaling with the following equation [28]:

$$
\begin{equation*}
\left(\frac{U_{o}}{U_{\infty} t_{2}}=\left[\frac{\left(U_{o}\right.}{\left.U_{\infty}\right)_{1}}-1\right] \frac{t_{2}}{t_{1}}+1\right. \tag{21}
\end{equation*}
$$

The second component corlesponds to the design load digtribution of the mean line and will be denoted by $\Delta U_{0} / U_{\infty}$. For any other camber or design iift coefficient the $\Delta \mathrm{U}_{0} / \mathrm{U}_{\infty}$ may be obtained by a direct ratio. The third component corresponds to the additional load distribution due to angle of attack and will be designated by $\Delta U_{a} / U_{\infty}$. For velocity dictinbutions at ilft coefficients other than the design lift coeif $\because: \quad \mathrm{ht}$, the ratio may be determined by multiplying the $\Delta U_{\alpha} / U_{\infty} a:$ iio dasign lift coefficient by $\left(C_{L}-C_{L_{1}}\right)$, Thus, the final intio for the lower auxface is

$$
\begin{equation*}
\frac{U}{U_{\infty}}=\left(\frac{U_{0}}{U_{\infty}}-\frac{\Delta U_{0}}{U_{\infty}}-\frac{\Delta U_{a}}{U_{\infty}}\right), \tag{22}
\end{equation*}
$$

and for an airfoil which is built up from another basic camber and thickiess form:

$$
\begin{equation*}
\frac{U}{U_{\infty}}=\left\{\left[\left(\frac{U_{0}^{0}}{\left.U_{\infty}^{( }\right)}-1\right] \frac{t_{2}}{t_{1}}+1\right\}-(\text { camber } \operatorname{rat} 10) \frac{\Delta U_{0}}{U_{\infty}}-\left(C_{L}-C_{L_{1}}\right) \frac{\Delta U_{n}}{U_{\infty}}\right. \tag{23}
\end{equation*}
$$

For this etudy, the velooity dietribution on the lower surface of the Eppler 266 airfoil is conetructed from the thickness form of the NACA 66,-012 with new thicknesm of $17 \%$ [ 29]. The mean inne in the NACA 66 with a camber ratio of $0.0326 / 0.060$. Thum the new denign $11 \mathrm{I}^{\mathrm{t}} \mathrm{cosificient}$ im:
( $0.0326 / 0.060$ )(NACA 66 mean line deaign coefficient)
and is equal to 0.413 .
The inal unknowns for evaluating the trailing edge boundary layer thicknemes at various velocitiea are the tranaition points from laminar to turbulent flow in the boundary layer. Using Equation (24) for the instability pointa for a symetrical airfolil rendse a reasonable approximation for these pointe [23]:

$$
\begin{equation*}
x_{1} / c=0.0537\left(C_{L}+1.02\right)-0,0493\left(C_{L}+1.36\right) \log R e \tag{24}
\end{equation*}
$$

Using Equation (24) in the 11 ft coeffioient range between 0.097 and 0.960 render the following values, atetated in Table III.

For the range of the various $11 \mathrm{f}_{\mathrm{t}}$ coefficients above and a constant Reynold number of approximately $1.845 \times 10^{5}$, it was determined that transition on the wator table remained approximately conatant at 0.78 of the model'm chord length. This figure was arrived at through a photographic etudy of the airfoil on
the water table when the llow was seeded with dye (refer to Section 3).

With this information, a computer progran was constructor using Schlitiching's equation for the momentum inickness which necessitates the numerical integration of the velocity distijbution over the lower surface of the alrioil. The results of this procedure are later presented in this section in the foim of tables and graphs (refer to Table IV and Figures 19 and 20).

There is a weakness in the assumption that tho velocity profile of the turbulent boundary layer ia constant along the entire length of the wing section. In reality, as the flow encounters an adverse pressure gradient, the profile will change. Tre profile closely resembles a power law soon after transition, but is modified by an adverse pressure gradient in such a maner that the proifle can be approximated by a power law with a smaller $1 / \mathrm{n}$ term (refer to Figure 21). The result is a change which lessens the steepneas of the profile. In order to check the magintude of the eflect of this changing protile on tho answers given by the computer, the exponent of the power law was varied.

The amount that the exponent mould vary in order to simulate actual flow conditions depenta upon the sevarity of the actual adverse pressure gradient. In Figure 22 the velocity distribution is plotted for a lift coefficient equal to 0. 960 , 0.635 , and 0.135 . Theme three lift coefficientereflect a widespread sampling of the velucity regime in which the aailplane filem. From Figure 22 it is easily seen that the flow begines to decrease in velocity_at $x / \bar{c} m 0.7$ for all three $11 f^{t}$ coof.ficiente. Thus from $x / \bar{c}=0.7$ to the trailing edge of the alrfoil the flow encounters an adverse pressure gradient. For the ifft coefficiente of 0.635 and 0.135 , a nearly zero pressuid gradient exists from the trangition point to $70 \%$ of the chord. For a lift coefficient equal to 0.96 , a gentle favorable paesm sure gradient existe from tiansition at $x / \bar{c}=0.4$ to $x / \bar{c}=0 . \sigma$, where a_near zero preselure gradient exista until $x / C=0.7$. From $x / c=0.7$ to the trailing edge all three encounter approximately the eame moderate adverse presaure gradient. On the basis of the near zoxo presmure gradient from transition to $x / c=0.7$ and for the relativoly moderate adverse pressure gradient which is experienced by the flow-for this $11 \mathrm{f}^{\mathrm{t}} \mathrm{f}$ coefficient rangen-from $x / \bar{c}=0.7$ to the trailing odge, it is assumed that the actual profile of the boundary layer willat all times lie between, the imiting power laws of $\mathrm{U} / \mathrm{U}_{\infty}=(\mathrm{y} / \mathrm{s}) 1 / 5$ and $\mathrm{U} / \mathrm{U}_{\omega} m\left(\mathrm{y} / \mathrm{r}_{\mathrm{i}}\right) 1 / 4$. Table V compares the values given by the computer for the boundary layer displacoment thicknese for the power law exponent equal to $1 / 4$ and $1 / 5$. On the basia of the comparison shown in Table $V$, it will be aseumed that $\beta_{5}$ constant velocity profile equal to the power law $U / U_{\infty}=(y / \delta) 1 / 5 w 111$ produce a high entimate of the actual boundary layer thicknesses. This ostimate is utilized during the remainder of this study.

## 5. RESULTS AND CONCLUSIONS

Now that the neceseary information is available to establish values for $\left\{\mathrm{nf}_{\text {water }}{ }^{2}\right.$ and $5^{2}$, the comparigon of the relative importance of the aerodynamic sound mources of the Eppler 266 airfoil throughout the flight velocity range of the Standard Austria SH1 if possible with aid of the ration $\mathrm{W}_{\mathrm{Ye}} / \mathrm{W}_{\mathrm{bl}}$ and $W_{V g} / W_{W}$ derived in Section 2. Tables and plots of Wysi $W_{b l}$ versus Velocity and $W_{V a} / W_{W}$ versus Velocity ure presented in Table VI and in Figures 23 and 24.

From these ration of acoustic energies, a qualitative summary may be made of the aerodynamic sound sources of the Eppler 266 adrfoil
(1) When one speakn of what is the predominant sound source, the velocity regime of the flight must be coneidered.
(2) In general, the most predominant sound source in the range from 60 to 210 feet per second appears to be that of the unsteady lift caused by vortex shedding at the trailing edge of the airfoil.
(3) At the lowest and the highest speeds in the aircraft' velocity range, however, the turbulent boundary layer appeara to be an important contributor to the overall acoustic output of the airfoil.
(4) At low speeds, the wake appears to have very $11 t \mathrm{t}$ le effect on the total acoumtic output of the alrfoil, but as the velocity exceeds 150 feet per second, the wake rapidly becomes more influential to the overall sound energy generated by the airfoil moving through the atmosphere. At velocities above the range of the standard Austria SHl, the wake may possibly be the most. predominant aerodynamic source of conventional subsonic aircraft.
(5) One would expect the frequency apectra of the sound Reneratod by the aircraft in the approximate velocity range of 74 to 100 feet per second to show peake at the characteriatic frequencies of vortex ahedding, but also to reflect the broad band nature of the sound generated by the tur-. bulent boundary layer. At the higher frequenciea, however, the influence of the braad band nature of the wake, turbulent boundary layer, and the increased randomness of the lift fluctuations predominates.

It should be noted that although the equations used for the acoustic output of the turbulent boundary layer ( $\mathrm{W}_{\mathrm{bl}}$ ), wake ( $W_{w}$ ), and lift fluctuations ( $W_{v a}$ ) are dimensionally correct, they have been applied to a very specific situation and thoil real significancs lies in their relative values with one anothes rather than their absolute magnitudes. Again, the derivation of Equation (1) is highly dependent on the presence of low Mach numbers and thus would lose validity in a higher velocity regime than that considered in this study. Also, a characteristic frequency of vortex shedding from atreainline body would not be expected unless highly unsymmotrical flow in the boundary layers of the upper and lower surfaces at the trailing edge ia present. At the higher velocities where mall angles of attack are present, the increased randomness of the unateady lift forces makes the concept of a characteristic frequency more difficult to apply. Thus a time average of the fluctuations is the best estimate that can be expected for such orientations. Therelore, for any exact calculations of the absolute acoustic output at angles of attack near zoro lift, this method becomes questionablo. However, for a first order approximation of the relative source strengths, this formulation of $W_{v s}, W_{b y}$ and $W_{w}$ should provide a good deal of insight into the parameters affecting the achievement of ultra-quiet flight.
$\overline{\mathbf{x}}^{-}=\overline{\mathbf{x}}^{\cdot}+\Delta \overline{\mathbf{x}}$


Figure 13. Qualitative Comparison of Noise Radiation from a

Figure 14. The Sound Fields Due to Oscillating Lift and Drag on a
Circular Cylinder in a Flow.

Figure 15. Schematic Diagram of water Table.

Figure 16. The Model Shape.


a. Configuration for still Photographe

b. Configuration for Motion Pictures

Figure 17. Photographic Techniques.

Figure 18. Frequency of Vortex Shedding versus Angle of Attack.


Figure 19. $F_{\text {water }}^{*} /{ }^{*}$ *ir ut the Tralling Edge of the Airtoil veraus $C_{L}$.

Figure 20. Boundary layer Thickness at the Trailing Edge of an
Airfoil and a Fiat plate versus Veiocity in Air. Airfoil and a fiat Plate versus veiocity in Air.

Figure 21. Modification of the Boundary Layer Profile as an


Figure 22. Velocity Dintribution Over the Lower Surface of the Airfoil for $C_{L}$ Equal to $0.96,0.635$, and 0.135 .


Figure 23. $W_{v s} / M_{b l}$ versus Velocity.


TABLE I

RELATIONSHIP BETWEEN AIR VELOCITY AND ANGLE OF ATY'ACK NECESSARY ON THE WATER TABLE

| Velooity in Air (fpm) | $C_{L}$ | Effective Ancle of Attack on the Table |
| :---: | :---: | :---: |
| 66.8 (atalled | 0.960 | $10.4{ }^{\circ}$ |
| 68.8 (unstalled) | 0.980 | S. 9 |
| 74.3 | 0.780 | 4.25 |
| 82.3 | 0.635 | 2.8 |
| 178 | 0.138 | -1.9 |
| 210 | 0.097 | -2.3 |

TABLE 11

| $\begin{gathered} \text { Angle } \\ \text { of Attack, } \\ \text { Degrees } \end{gathered}$ | $\begin{aligned} \mathrm{V} & =0.549 \mathrm{fps} \\ \mathrm{Re} & =1.39 \times 10^{5} \end{aligned}$ | $\begin{gathered} V=0.567 \mathrm{fps} \\ \mathrm{Re}=1.435 \times 10^{5} \end{gathered}$ | $\begin{gathered} V=0.679 \mathrm{fps} \\ \mathrm{Re}=1.845 \times 10^{5} \end{gathered}$ | $\begin{gathered} V=0.740 \mathrm{fps} \\ \mathrm{Re}=1.875 \times 10^{5} \end{gathered}$ |
| :---: | :---: | :---: | :---: | :---: |
| 5.9 | -- | 2.50 cps | -3.1 cps | --- |
| 4.25 | 2.86 | 3.00 | 3.63 | 4.1 |
| 2.8 | -- | 3.21 | 3.8 | -- |
| -1.9 | -- | 2.79 | 3.23 | - |
| -2.3 | - | 2.57 | 3.00 | -- |

TABLE III
THEORETICAL TRANSITYON POINTS FOR ATMOSPHERIC FLIGHT

| $C_{L}$ | Reynolds Number | Approximate <br> $x_{1} / \bar{c}$ |
| :---: | :---: | :---: |
| 0.097 | $3.75 \times 10^{6}$ | 0.14 |
| 0.135 | 3.18 | 0.15 |
| 0.200 | 2.61 | 0.20 |
| 0.300 | 2.14 | 0.23 |
| 0.400 | 1.86 | 0.25 |
| 0.300 | 1.66 | 0.27 |
| 0.635 | 1.48 | 0.30 |
| 0.780 | 1.32 | 0.35 |
| 0.980 | 1.18 | 0.40 |

TABLE IV
the results of the theoretical calculations of the boundary layer thicknesses

| $C_{L}$ | 5* water | 5*air | $\frac{5^{*} \text { vater }}{5^{*} \text { air }}$ | 5 air | $6_{\mathbf{a i x}}^{2}$ |
| :---: | :---: | :---: | :---: | :---: | :---: |
| 0.960 | 0.00372 ft | 0.01216 ft | 0.306 | 0.0730 ft | $0.00530 \mathrm{ft}^{2}$ |
| 0.780 | 0.00382 | 0.01377 | 0.278 | 0.0825 | 0.00680 |
| 0.635 | 0.00391 | 0.01544 | 0.253 | 0.0926 | 0.00859 |
| 0.500 | 0.00399 | 0.01687 | 0.236 | 0.1011 | 0.01023 |
| 0.400 | 0.00406 | 0.01748 | 0.232 | 0.1048 | 0.01098 |
| 0.300 | 0.00412 | 0.01804 | 0.228 | 0.1081 | 0.01170 |
| 0.200 | 0.00418 | 0.01929 | 0.217 | 0.1157 | 0.01336 |
| 0.135 | 0.00423 | 0.02063 | 0.204 | 0.1239 | 0.01531 |
| 0.097 | 0.00425 | 0.02043 | 0.208 | 0.1225 | 0.01500 |

table V
THE EFFECT OF CHANGING THE POWER LAW DESCRIBING THE BOUNDARY LAYER PHOFILE

|  | $8^{*}{ }_{\text {a }}$ Ir | $8^{* * 19}$ |  |
| :---: | :---: | :---: | :---: |
|  | (where $\frac{U}{U_{\infty}}=\left(\frac{y}{b}\right)^{1 / 4}$ ) | $\text { (where } \frac{U}{U_{\infty}}=\left(\frac{y}{6}\right)^{1 / 5} \text { ) }$ | Percont Diffurence |
| 0.980 | 0.01150 ft | 0.01216 ft | 3.4 |
| 0.035 | 0.01440 | 0.01544 | 6.6 |
| 0.135 | 0.01858 | 0.02083 | 9.9 |

TABLE VI
COMPILATION OF THE THEORETICAL AND EXPERIMENTAL RESULTS

| Velocity | $\left\{\bar{n} f_{\text {watex }}\right\}^{2}$ | $8^{2}$ | $W_{V_{61}} / W_{b 1}$ | ${ }^{W}{ }_{V g^{\prime}} /{ }^{W}{ }_{W}$ |
| :---: | :---: | :---: | :---: | :---: |
| 67 fpm | 1.96 | 0.0053 ft | 2.81 | 153 |
| 74 | 2.21 | 0.0068 | 3.58 | 172 |
| 82 | 2.02 | 0.0086 | 3.66 | 183 |
| 93 | 1.77* | 0.0102 | 4.04 | 131 |
| 104 | 1.64* | 0.0110 | 3.24 | 89.5 |
| 120 | 1.81* | 0.0117 | 2.84 | 63.5 |
| 146 | 1.22* | 0.0134 | 2.44 | 37 |
| 178 | 0.84 | 0.0153 | 1.84 | 21.4 |
| 210 | 0.85 | 0.0150 | 1.83 | 11.8 |

## 1. OUTLINE

The objective of the aerodynamic experiments with the owl wing was to attempt to discover the aerodynamic mechanisms, if such exiat, which were responsible for the silent flight. A small wind turnel was deaigned and constructed specifically for testing the wing at low velocitiea. A wing from a smallow was used so that three-dimensional flow could be studied. Four types of experiments were deaigned:
(1) Visual observation of the aeroelastic motions,
(2) Flow viaualization using a moke generator,
(3) Measurement of the motions of the wing or its components by use of a strobe light,
(4) Flow studies over the wing by use of tufta and hot wire inftrumenta.

## 2. TEST FACILITY

The wind tunnel. was dealgned and conatructed from plywood. It consiated of three bailc parte:
(1) The atiliting ohamber. Thim is a rectangular box of dimensions $49^{\prime \prime} \times 30^{\prime \prime} \times 30^{\prime \prime}$. Three double thicknesses of fine meah acreen wire were used to eliminate large flow irregularities. Butween the ecreons and the blower, the bax was filled with plastic wool to reduce the effect of the blower which inducea unsteady, turbulent flow. The exhaust from the stilling chamber was a squaro rigid nozzle with a $10^{\prime \prime} \times 10^{\prime \prime}$ exhaust croms section. A mmoke generator Was ylaced upetream of the nozzle to inject the smoke prior to acceleration by the nozzie. The souice of smoke was cigars placed in the end of a tube through which air was blown. The amall plastic mouth piece on the cigar was used as a nozzle.
(2) The air flow generator was a centrifugal fan. It was powered by a $1 / 4$ hormepower electric motor through the ued of pulley drive axjangement. The rpm could be adjusted to vary the wind speed.
(3) The model mount system. Immediately downstream of the nozzle a frame outside of the alr jet was used to mount the owl wing. The nounting platform could be adjusted to vary the angle of attack, the angle of yaw and the position downstream of the nozzle.

The wind tunnel arrangement is shown in Figures 25 and 26.

## 3. ANALYSIS OF THE WIND TUNNEL

The wind tunnel experimente on the owl wing were run only at one 12 . Ner speed. This corresponded to the average flight velocity experienced in the accustic experiments. The relative velocity profiles and the turbulence data taken for calibration was at an average velocity of 1250 feet per minute. The following is the description of these results: ligure 27 shows the flow calibration. The deviations from the basic calibration speed are shown at each measurement position. A hot wire anemometer was used as the measuring instrument. The edge of the laminar section can be seen. Steady state distortion between -120 and +70 feet per minute were found. The unsteady character of the flow is shown in Figure 28. An oscilloscope trace was photographed depicting the output of the hot wire anemometer. Measurements showed that the average unsteadiness was plus or minus $1.25 \%$ in the low frequency range and plus or minus $3.9 \%$ at high fiequency.

## 4. HOT WIRE INSTRUMENT

The velocities were measured by a portable Flowtronic; Model 55日l, hot wire anemometer manufactured by the flow Corporation, Cambridge. It is linearized and indicatfs the velocity in units of feet per minute. The calibrated meadurement range was from 0 to 12,000 feet per minute.

## 5. STROBE LIGHT

Frequencies of aeroelastic motions of the wing were measured with a Strobetac, Model 631DL, manufactured by the Radio Corporation, Cambridge. Its range is from 1 Hz to 240 Hz .

## 6. TEST OBJECT

Two owl wings were used. These were both the left and right wh.ngs from the same bird. After several photographs were taken
of the live owls from the Bionics study in gliding filght, the planform geometry was established. The two small wings were then made flexible by soaking them in water. Next, they were formed into a life-like shape using the aforementioned photographs for reference. Finally, the wings were dried so that they retained the desired shape. From this configuration the basic data were obtained:

| Wing span: | $18^{\prime \prime}$ |
| :--- | :--- |
| Wing sweep: | $0^{\circ}$ |
| Maximum chord: | $4.5^{\prime \prime}$ |
| Wing area: | 72 sq. in. |

## 7. EXPERIMENTS

## a. Visual Observations

The owl wing was fixed rigidiy in the model mounting system. The wing root was held with a clamp while the outboard portion of the wing was allowed to remain flexible. Care was taken to simulate the live flight configuration. The following observations were made:
(A) The wing changed shape upon introduction of the flow velocity because of the elasticity of the feathers. The first three tip feathers curved up to some position which was strongly dependent upon the angle of attack. Figure 30 shows this condition. If the angle of attack was either very small or very large the curvature of the feathers diminished. There was a unique angle of attack where maximum curvature occurred. The tradilng edge of the wing also changed its shape in the flow. Flgure 31 shows the comparative geometries of the wing in undisturbed and disturbed air. On the leading edge at about the mid-span poaition, there are two small featherg (see Figure 33). These feathere separate from the wing at a diatance which is porportional to the angle of attack.
(B) The wing oscillates. Some parts of the wing appear to be independent of the total wing system in the manner in which they move. Oscillation is always present in the wing despite the fact that the nozzle flow $1 . s$ laminar. The amplitude of the motion depend on the angle of aitack, They achieve their maximum amplitudes at very small and very large angles of attack. There is one angle
of attack where the amplitude of oscillation reaches a minimum.

There are five regions of the wing which appear to maintain independent oscillation. These are:

1) The first three feathers of the wing $t i p$,
2) The tradiling edge,
3) The leading edge (which is extremely small in amplitude),
4) The surface region near the owl body,
5) The trailing edge near the owl body. These regions are shown in Figure 34.
(C) Behind the leading edge in region 3 of Figu re 34 the surface of the wing ds mildiy porous. This allows the air to pass through from the lower to the upper surface beneath the amall secondary feathers. This is noticed during certain movements of the feathers.

## b. Smoke Generator Experiments

The owl wing was tested in the wind tunnel using the smoke generator to study further details of the flow . Photographa were taken to allow detailed investigations of the flow. The angles of attack and position of the smoke generator were varied. Special emphasis was placed upon the flow in the outboard region. Figure 35 shows a high speed picture with an exposure time of 0.001 sec, Vibration of the smoke generator pipe caused the smoke stream to appear turbulent; however, this was not felt detrimental as the flow in the high speed section remained laminar. The exposure time was increased to approximately $1 / 60$ sec. In order to better describe the streamine flow.

Figures 36 and 37 show the flow at a moderate angle of attack and a very large angle of attack. In neither case was the flow separated on the outboard regions of the wing.

Figure 38 shows how the streamlines are shifted outboard on the upper surface. Figure 39 shows that the amoke is separated into two stream, one of which curles to form a helix. Further downstream both parts coalesce to form a large trailing vortex.

Fdgure 40 shows the wake being turned in, aft of the trailing edge. The flow defined in these pictures da outside of the boundary layer. Flow immediately on the wing surface will be described in detail in a later section.

In an earlier section of this report a flow oscillation was described. The purpose of the strobe light experiments was to analyze the frequency and the modes of these oscillations. This was accomplished as follows; By focusing the strobe light on the model ia a darkened room the model was clearly illuminated. The oscillating parts of the wing appeared blurred. The frequency of the strobe light was slowly changed until the blurring vanished on a particular element being studied. When the moving part appeared to stop the frequency of the strobe light was recorded. This was considered to be the frequency of the oscillating part. The following was discovered: Each part that was found to be in motion fell in a frequency band of between 13.3 to 15.0 Hz . The large and small feathers alike vibrated in this same way. The motion appeared to combine wing bending and torsion in such a way that the leading edge of the wing was very nearly the nodal axis. Though the frequencies were extremely close together the amplitudes and phases were different.

## d. Experiments with the Tuft Probe

Several types of probes were used to study the flow. The larger probes were made by attaching a small wool thread to the end of a metal tube (approximately 2 mm in diameter). This type of probe is sketched in Figure 41. Another prove using a smail sewing neddle was built for purposes of studying flow very near the wing surface. The lengths, $d$, of the tuft were varied from upproximately 4 mm up to approximately 5 cm to atudy various portions of the flow. A small amount of glue was touched to the end of the tube or needie, and the tuft attached in this way. The glue was of such a plastic nature that the tuft could be separated from the tube proper and be hold with the extended filament of the glue to allow extreme flexibility.

By use of a tuft 4 mm in length attached to the sharponed end of the sewing needle the flow direction of the boundary layer was studied. This study was difficult and time consuming due to the intricate mechanism of flow control over the owi wing. Only small vertical displacements of the tufts would indicate large changes in flow direction due to the presence of the complex vortex sheet distribution. For example, approximately four hours of uninterrupted probing were necessary to map the flow field on the outboard half of the wing upper surface. At one location the direction of the atreamines changed 180 degrees over a depth change of about one mlilimeter. The following conclusions were made conderning the mechandsms of the flow at two angles of attack.

If the wing was at a small angle of attack, the boundary layer performed in the following way: Laminar flow existed
over both the upper and lower surfaces everywhere. An extremely thin flow discontinuity existed behind the wing. The atreamlines over the upper surface curved slightly toward the wing root. The streamlines over the inner parts of the wing moved more in the streamwise direction. No effect concerning the leading edge comb could be recognized at this angle of attack. Such flow 1s seen in figure 39. Large vortices separated from the tip feathers and appeared in the trailing wake over the entire outboard half of the wing, The leading edge slot was only silghty open. The termination of the large tradiling vortex system in the wake occurred directly downstream of the slot.

The flow at large angles of attack corresponding to the flight condition showed an entirely different pattern. This is shown in Figure 42. The leading edge combs became very active in producing the resulting field. No deformation of the comb system occurred at any conditions tested. The leading edge slot was extended away from the wing and at the increased angles of attack. Iminediately behind the slot the flow moved inboard; however, the sense of the trailing vortex was as expected (clockwise, looking upetream). This vortex aided in moving the baundary layer flow on the aft portions of the wing outboard. The leading edge combs developed a vortex wheet which allowed the flow in the boundary layer to be initiated in the outboaid direction while the flow above the sheet moved in the chordwise direction.

Flgure 43 shows how the comb elements are twisted and tapered to provide the developmunt of the sheet. Thus, a vortex system exists from the leading edge slot to ali wing positions outboard. The tip feathers also aid in promoting fiow in the outboard direction.

Probably the most interesting and unique condition of the flow field was the development of back flow immediately aft of the leading edge comb system. This was delinitely not a separated region. The flow was attached and laminar throughout this region. The way this develops may be explained by the sketch in Figure 44. The leading edge comb immediately develops a vortex sheet promoting spanwise flow in the outboard direction on the surface. The leading edge slat being intersected with the upper surface of the wing leading edge develops a low pressure in the region immediately behind its point of intersection, which creates a localized inboard flow. This leaves a region aft of the combs to be filled with the flow containing the comb induced vorts: sheet. Somewhat further downstream of the wing the slat induced vortex begins to move the boundary layer flow back in the outboard direction. Flow along the comb induced vortex sheet fills the void that is created by these diverging flows. Upon interaction with the wing the flow moves back toward the leading edge

- and develops two counter-rotating fields. The streaminne flow from these figlds is able to eacape from this region by the strongly three-dimensional nature of the flow. The entire process occurs in a region of thickness from 4 to 5 mm . Above this (boundary flow) the streamines move in a more chordwise
direction. Thus the inlluence of the leading edge combs in combination with the slat and tip leathers produces a vortex sheot system that enables a very complex, yet entirely laminar, flow system to develop.

Another interesting observation was that this system con$t i n u e s$ to work in this same manner at extremely large angles of attack. Though the tip ieathers reach some maximum deflection position at an intermediate angle of attack the leading edge siot continues to open over the increaning angle of attack range. Laminar flow continued to exist oven at angles of attack in excess of $30^{\circ}$. Turbulent flow and separation thus are effectLvely resisted by the throe-dimensional flow created by the wing.

The final part of this study was to investigate the ow wing with the leading edge vortex sheet generator removed. The comb was simply cut off with a pair of scissors. The flow over the wing near the surface was mapped. The counterrotating areas were replaced by a single large system with the flow being directed inboard at the leading edge. Flow separation occurred immediately aft of the leading edge and reattachment was seen near the trailing edge with considerable turbulence being induced.

Figure 25. Wind Tunnel Test Facility.


Figure 26. Photograph of Wind Tunnel.


Bagic Speed 1250 土tm
" $T$ " indicates regions of flow turbulent

Figure 27. Flow Calibration.

$20 \mathrm{ft} \cdot \mathrm{pm}$

Basic Velocity $=1250$ FPM


20 ft . pm

Basic Velocity -925 FPM
Figure 28. Turbulence Records.


20 ft. pm

Basic Velocity - 950 FPM


400 ft. pm

Basic Velocity $=950$ FPM

Figure 29. Turbulence Records.


Figure 30. Wing Curvature.


Flgure 31. Wing in lhe Undisturbed Air.


Figure 32. Wing in the Flow.


Figure 33. Mid-Span Leading Edge Feathers.


Figure 34. Regions of Independent Osciliation.


Figure 35. High Speed photopraph ol Smoke Stream.


Figure 36: Flow at Moderate Angle of Attack.


Figure 37. Flow at Large Angle of Attack.


Figure 38. Flow Through the Wing.


Figure 39. Flow Above the Wing.


Figure 40. Trailing Flow.


Figure 41. Sketch of Tuft Probe.


Figure 42. Flow Pattern at Flight Angles of Attack.


Figure 43. Sketch of Flow Through Loading Edge Comb.


Figure 44. Backilow Generation.

## APDENDIX III

WATER 'TUNNEL STUDIES

A series of fluid flow tests were conducted in The University of Tennegsee Space Instituto water tunnel to evaluato concepts suggested by the combined Aerodynamic, Acoustic and Bionius studies. The objective of the experiments was to apply vailous aerodynamic quieting devices to low drag series airfoils and evaluate the resulting flow ilelds. An additional objective was to develop the quieting modifications in such a way that they could be applied to lull scale gliders. That is, when a quieting mechanism wus described from the other parts of the program, a practical analog was to be designed and testod lol. stability, boundary layer performance and wake chadactordatics. Early phases of the project pointed to the downwadd fluctuntions and boundary layer radiation as the important sources of 110 ls is to consider. Thus tho flow conditions at tho tralilng odge were considered to be the most important key to the noise problem.

## 1. THE WATER TUNNEL

The UTSI water tunnel is a olosed roturn systom driven by a 1 HP electric motor through an infinitely variable transmission system. The test section dimensions are $50 \times 12 \times 18$ inches The top speed with sufficient guality flow for these tests was one foot per second or a Reynolde Number of about 60,000 par foot. The largest airfoll model available had a chord of two feet. Thus to operate at a reasomable Reynolds Number, tho tunnel speed had to be incleased. This was accomplished by designing a test section insert which dropped the channel width from 12 inches to two Inches. This provided Reynolds Numbers lif excess of one half million whiteh was considered adequate. The top of the test section was open for easy model access.

A dye probe using potassium pormagamate crystals dissolvad in water was used for flow visualization. The dye was injected upatream of the nozzle in the low speed section. The dyeflow rate could be varled to eliminate unsteady flow resultink liom the presence of the probe.

The probe poaition could be varied vertically for studyinh different levels of stream flow above and below the model, it. could also be moved along the direction of flow into the nozale so the intenaity of the dye could be increased. Figure 45 shows a schematic of the test section arrangement. The flow was sulficiently steady and laminar to provide good visualization.

## 2. THE MODELS

The models used in these tests were three sizes of NACA 652414 airfoils (Ref., Appendix I). Their chords were 27 , 18 and $\theta$ inches. When they were instulled in the test section they were sealed to the walls all around to eliminate leakage from the lower to the upper surface. The only tests considered were two dimensional. However, some degree of three-dimensional flow could be simulated.

The angle of attack could be varied over a wide range. Excessive boundary interference was observed when the large model was installed so the 18 and 9 inch models were used for most of the experiments.

## 3. EXPERIMENTS

Two types of experiments were conducted. These were to investigate the effectiveness of vortex generators simulating leading edge combs and flexible trailing edge devioes to simulate wing compliance. The resulta of these teats are ais follows;

## (a) Vortex Generators

Several suggestions for vortex generator configurations were provided by the Aerodynamicm and Bionicm atudies. Some of these are depicted in Figure 46. Early in the experimenta iittie was known of the use of the leading odge vortex generator by the owl so nearly arbitrary solections of configurations were made. Various perturbations from these basic shapes were easily made since systems a through e were conatructed of 0.015 sh m atock and could be easily cut and installed. The trailing wake system was visually observed by use of dye injection to determine its thickness and turbulence. The gradiente in the boundary layer at the trailing edge could be obmerved by pulsing the dye stream. The vortex generator could be placed on the leading edye or any chordwise position on the upper or lower surface. In virtually all configurations temted, the remulte were momewhat the same. The boundary layer transitioned to turbulent much earlier than on the clean airfoil. The wake turbulence increased and appeared to gain higher frequency componente. The wake thickened. The maller vortex generators meemed to have lem effect on the tralling wake thickness, It appearyd that no positive mense of direction was being gained by these experiments so they were temporarily stopped.

Next, a closer look was taken at the flow mechanisms induced by vortex generatore. Tho ideal vortex motion immediutely downstream of a configuration such as in Figure 46d might appear an shown in Figure 47.

Next, consider the flow pathern il the vanos are stargerod with an alternating loft and reght angle of attack to the flow. The sheet roll-up will be of the same type, yet having only half as many elemental vortices in the wake. Neither of thase systems of vortices is etable and both very quickly dissipate into turm bulent flow.

Findily, if the vanes are all placed at the aame angle of attack to the flow as in a cascade, a vortex sheet such as shown in Figure 48 would be developed. This configuration possesses a high degree of stablifty.

After the above configurations were made concerning the action of various vortex generating devices, a new type was tried. This was merely a piece of shim stock cut normal to the leading edge at one-edghth inch intervals about one-fourth inch deep and the reaulting blades twisted to form a cascade. Ihis system was installed at the leading edge, slightly downstream of the atagnation point. No significant flow improvement was made but a rather crude crose flow on the adrfoil surince appoared. Upon close examination of the flow over the wind tunnel mounted owl wing, described in Appendix II, a similar cross flow was noted. Further inspection of the leading edge comb rovealed a twiated cascade arrangement with tapered, washed out tipa. Ihis gystem was aimulated by taking the vortex generator ahown in Figure 46c and twisting each of the elementa until the tipe had a zero dogree angle of attack with respect to the free strean. Care was taken to approximate a linoar twist from the root to the tip. Tests of this systom showed the development of a distinctiy laminai vortex sheet downstream of the comb, The threedimensional nature of the flow precluded arriving at further conm clusions threugh the une of the water tunnel because of the narrow test channel. The studies on the perfurinance of vortex Meneratore were stopped and referred to the wint tunnel for further analysis.

## b. Flexible Trailing Edge Devices

During the course of the research program the important nature of the wing compliance was discovered. The development of a mechandem to produce an optimized wing compliance was lelt to be beyond the level of thim effort as it would involve very specialized and detailed considerations. However, a mechanism was conceived which could possibly have some merit. This was the use of a flexibly mounted aplittor plate along the traling edge of the airfuil. This arrangement is shown in figure $4 \theta$. The plate was taped to the airfoil so that no gap would bo present yet a high degree of compliance would oxist at the attachment.

The exact mechaniem for noise reduction by compliance was not known; however, preliminary teate indicated the sencitivity of the point of boundary layer transition on the wing upper
surface to the position and size of the splitter plate.
The first tests that were made were to establish the plate length. This was done by cambering the plate so that it would move to a predetermined position. A range of auch positions was tried until the furthest aft point of boundary layer transition would occur yet the laminar nature of the flow on the lower surface was not destroyed. Several plate lengths were tried until an optimum was found. This corresponded to a plate length of twelve percent of the wing chord. It is expected that this particular dimension will be a function of Reynolds Number. The proof of this method of noise reduction could not be done in the water tunnel. However, the plate did oscillate at a low frequency as was predicted from the study of the Owl flight.

A brief and inconclusive look was taken at the influence of porosity on the boundary layer flow at the trailing edge. A cambered flat plate with lateral mlota was placed in the water tunnel. Flow viaualization showed that the lower surface could be made fully laminar. The wider the alots were cut, the lower the frequency of the outer boundary layer could be made. However, no concluaion wan made regarding the spectral content of the inner layere. The total boundary layer was thickened, however, posmibly implying reduced gradients and frequencies. The application of this concept to a thick airfail would be quite simple.

This concluded the water tunnel experiments.


Fifure 45. Water Tumel Test Section Set-Up

| a | b |
| :---: | :---: |
| c | $\square$ <br> d |
| mam <br> $\theta$ | SCREEN WIRE <br> $\mathbf{f}$ |
| CIRCULAR SPIKES <br> g | 0.010 WIRE <br> h |

Scale；Full

Figure 46．Vortux Generators．


Figure 47. Tip flow from Vortex Gonerator.


Figure 48. Tip Flow From Cascade.


Figure 49. Splitter Plate Attachment.

## APPENDIX IV

## ACOUSTIC MEASUREMENTS OF THE AERODYNAMIC NOISE PRODUCED BY FLYING OWLS

A series of tests was conducted at The University of Tennessee Space Institute to determine the "aerodynamic noise" produced by owls during flight, especially during gliding approaches. The owl species used in these tests is the Florida Barred Owl (strigiformes strix varia alleni).

In order to measure the total sound power produced by the flying owl and the spectral distribution of the overall noise signal the reverberation chamber method was selected for these tests. The chamber used was a $240 \mathrm{~m}^{3}$ room with reverberation times varying between approximately 0.8 sec and 0.4 sec in the frequency range 100 Hz to 10 kHz . The lower limiting frequency of this room below which statistical acoustic conditions were degraded was found to be around 100 Hz . The general procedure of obtaining the acoustic data was to record the unmodified noise signal and the overall sound pressure level of the notse produced by the owl flying in the reverberation chamber. Subsequently the noise signal record was frequency analyzed in terms of one-third octave frequency bands.

A particular requirement of these tegts was to obtain representative noise samples from only the gliding phase of the observed filghts. In all tests the owl was forced to fly from an upper perch in one corner of the rocm to a lower perch in the diagonally opposite corner. A flight generally consisted of an initial flapping phase followed by a giiding phase and a short flapping during touchdown. As a result of the owl's speed and the dimensions of the test room the observation of the gliding phase was relatively short, generally in the order of 0.6 to 0.8 seconds. By introducing a light string barrier mounted near the middle of the room the test flight configuration became very reproducible. The owl was forced to flap to the string barrier and then to a gliding filght with nearly constant speed to the lower perch. Figures 50 and 51 give more details of the filght conditions.

The resulting acoustical data which are reported here were measured with a single condenser microphone having a sensitivity of $5 \mathrm{mV} / \mathrm{microbar}$ and a flat frequency response in the range 2 Hz to 10 kHz . The microphone was positioned alightly above the flour at a sufficiently large distance from the flight path, and for all frequencies well beyond the reverberation radius of the moving noise source, During the filght tests the noise signal and the overall sound pressure level were recorded simultaneousiy by an FM magnetic tape recorder and a sound level recorder.

The measuring and analysis setup and its frequency characteaistins are shown in Figures 52 and 53. The average reverberation time of the test room is given in Figure 54. The one-third cotave band spectral analysis in the frequency range below 25 Hz has been made by application of magnetic tape frequency transformation. Samples of the resulting measurements are shown in Figures 55 through 61. On the photographs showing the unmodified noise signal as sensed by the microphone, Figures 55 and 56, a 1 kHz signal traced during part of the filght indicates the gliding phase as judged by the observer in the test room. In identifying the part of the signal produced during the gilding of the owl, evidently the observer's reaction time has to be taken into consideration. The time history record of the overall mound pressure level in decibel is a graphical logarithmic recard of the root-mean-square value of the time varying signal shown on the photograph (see Figures 57 through 39). In a followup analyais a reproduction of the noise signal only from the gilding phase of the test flight was made and a frequency analysis of this signal in terms of onemthird octave bands was obtained. The gliding noise/frequency spectra for the reported flight tests are shown in Figure 60 and 61. The spectral sound pressure level distribution as measured in the reverberation chamber is compared with the ISO, R226, threshold of hearing case for pure tones. The spectra shown are corrected for the ambient noise level observed during the testa. This was done by using the formula

$$
\begin{equation*}
0-\frac{n_{\text {ospl }}-n_{\text {back }}}{20} \tag{1}
\end{equation*}
$$

where $n_{\text {ospl }}$ js the measured overall sound pressure level in dB and nback is the background noise level in dB.

The overall sound pressure level data resulting from the reverberation chamber measurements permit the calculation of the total sound power produced by the flying owl (see for example: Kinsler, Frey, Fundamentals of Acoustics, (1967) p. 436), If the values found for the overall produced sound power are used to define a simple spherical source radiating with the same intensity in a free field, an estimate of the sound pressure levels produced by a gliding owl under dree field conditions can be obtained. Spectra resulting from such an estimate with absorption effects (air of $50 \%$ relative humidity at $20^{\circ} \mathrm{C}$ ) taken into account are shown in Figures 62 through 64 for distances equal to the reverberation radius $r_{1}, 3$ meters, and 10 meters. The reverberation radius has been estimated according to

$$
\begin{equation*}
r_{1}=\left[0.163 \mathrm{~V}(16 \pi \mathrm{~T})^{-1}\right]^{1 / 2} \tag{2}
\end{equation*}
$$

with the chamber volume $V$ measured in cubic meters and the reverberation time $T$ in seconds.

The aeroacoustic data resulting from these tests with filding owls make it quite understandable that an aural detection ol a gliding owl at distances larger than approximately three meters from a human observer is quite unlikely. From the spectra shown in Figures 62 through 64 it can be seen that beyond a propagation distance of three meters the sound pressure levels in all frequency bands up to at least 10 kHz has dropped below the threshold of hearing curve. The produced total overall sound pressure level ( 40 dB to 50 dB ) is relatively high. The relative quietness of the owl flight appears to be rather a result of an appropriate noise energy distribution over the frequency spectrum rather than an overall low noise level. A comparison with a similar noise spectrum from a sailplane overflight in a compaiable distance, for example, shows the fundamental difference in the shape of the spectra, see Figure 65. Evidently the overall noige level time hiatory during a teat flight, is a periodic aignal component which is found to occur in all gliding phases of the analyzed test filghts. This periodic component has a dominant frequency of 15 Hz well in the aubaudible range of the human ear. The following facts about this periodic component are to be mentioned. The 15 Hz component has never been observed, except during the owl filghts where it dominates the noise gignal during nll gilding phases. Further, the signal in a number of tests exhibits an amplitude modulation as e.g. in the case of Figure 55, with first increasing and then decreasing amplitude. The 15 kiz coincide with the lowest fundamental mode of the reverberant chamber. The measured amplitude is therefore not directily comparable with the amplitudes of the high frequency components. This ia due to the fact that at low frequencios pronounced room resonanceg occur which lead to a non-uniform frequency reaponse of the room. The reverberation time for this frequency range depends strongly on the frequency and a meaningful average reverberation time cannot be defined. At higher frequencies when the sound field becomes "diffuse" the frequency response of the room is more uniform and average reverberation $t$ imes may be used.

In preliminary wind tunnel tests flutter frequencies of about the same frequency have been observed (see Appendix II).


Figure 50. The Owl Approaching the Lower Perch in a Gilding Descend After Pameing the String Barrier.


Fiyure 51. Double Exposure of the Owl During a Gilding phase. The Measured Filight Spood in this Case is $20.6 \mathrm{ft} / \mathrm{s}$.

## Recording:



Reproducing:


Figure 52. Block Diagram of Tost and Analysia Instrumentation.


Figure 53. System Frequency Response.


Figure 54. Average Reverberation Time of Test Roolin.

Figure 55. Typical Recordings of Linear Sound Signals.



Flight No. 12:
Leading edge removed

Flight No. 14:
Trailing edge removed

Flight No. 17:
Top of wing modified

Figure 56, Linear Sound Signale of the Modified Owl.


Figure 87. Typical Recordinge of Overall Sound Preasure Level. (A, Filght No. 102; B, Flight No. 109)

Time, $\Delta t=t-t_{\text {start }}$ gliding
Figure 58. Sound Pressure Level of a Gliding owl.

Figure 59. Sound Pressure Level of a Gliding Modified Owl xधqо.


Figure 61. Frequency Analysis of the Sound Pressure Level of a Gliding Modified Owl
Frequency, Hz

Frequency Analysis of Flight No. 7, Series II, for Several Distances of Obeervation:


F'igure 62 Distance of Observation: 1 meter.


Figure 63. Distance of Observation: 3 meters.


Figure 64. Distance of Observation: 10 meters.


Figure 65. Comparison of Spectrum Shape of Aerodynamic Noise Produced by Owl and Sailplane

TABLE VII
TEST SERIES I: ACOUSTIC MEASUREMENTS OF OWLS DFCEMBER 13, 1970

| Flight No. | Bird | Flight Quantity | Comments |
| :---: | :---: | :---: | :---: |
| 100 | Ow 1 Rebel | nu gilding phave | bird didn't land |
| 101 | Owl Rebel | groodgliding phame |  |
| 102 | Ow 1 Rebel | grood gliding phase |  |
| 103 | both owls | no gliding phame | bl.t didn't land |
| 104 | Owl Rebel | good gliding phame |  |
| 105 | Owl Rebel | goodgliding phame | one flap during gliding phase |
| 106 | Owl Rebel | no gliding phase | bird didn't land |
| 107 | Owl Rebel | good gliding phame |  |
| 108 | Owl Rebel | no gliding phase | bird didn't land |
| 109 | OwI Rebel | goodmlidink phame |  |
| 110 | Ow1 Rebel | no gldiding phase | bird didn't land |
| 111 | Owl Rebel | no gliding phase | bird didn't land |
| 112 | Ow1 Rebel | no gliding phawe | bird didn't land |
| 113 | Owl Rebel | no gliding phase | bird didn't land |
| 114 | Ow1 Rebel | no gliding phase | bird didn't land |
| 116 | Owl Rebel | no gliding phase | bird didn't land |

## TABLE VIII

TEST SERIES II: ACOUSTIC MEASUREMENTS OF OWLS JANUARY 10, 1971

| Flight No. | Bird | Flight Quantity | Comments |
| :---: | :---: | :---: | :---: |
| 1 | Owl Rebel | good gliding phase | no aine-signal during gliding phase |
| 2 | Owl Rebel | goodgliding phase |  |
| 3 | Owl Rebel | goodgliding phase |  |
| 4 | Owl Rebel | goodgliding phase | wing touched wall |
| 5 | Owl Rebel | good gliding phase |  |
| 6 | Owl Rebel | grood gliding phase | $\begin{aligned} & \text { leas gliding than } \\ & \text { before } \end{aligned}$ |
| 7 | Owl Rebel | goodgliding phase |  |
| 8 | Owl Rebel | no gliding phase | bird lande like a parachute |
| 9 | Owl Rebel | goodgliding phase | wing touched wall |
| 10 | Owl Rebol | goodgliding phase | gine-slgnal was too short |
| 11 | Owl Rebel | goodgliding phase | wing touched wall |
| 12 | Owl Rebel | goodgliding phase | wing touched wall |
| 13 | Owl Rebol | goodgliding phase | sine-signal was too long |
| 14 | Owl Rebel | goodgliding phame |  |
| 15 | Owl Rebol | no gliding phase | bird lands like a parachute |

TABLE IX
TEST SERIES III: ACOUSTIC MEASUREMEATS OF OWLS - MARCH 4, 1971

| Flight No. | Bird | Flight Qunntity | Status of Bird Modification | Comments |
| :---: | :---: | :---: | :---: | :---: |
| 1 | Rebe 1 | no gliding phase | unmodified | bird lands like a parachute |
| 2 | Bebel | gliding | unmodified | Low swoop |
| 3 | Rebel | good gliding phase | uniodified |  |
| 4 | Rebel | gliding | unmodified | 10w swoop |
| 5 | Rebel | good gliding phase | unmodified |  |
| 6 | Rebe 1 | good gliding phase | unmodified |  |
| 7 | Rebel | no gliding phase | leading edge removed | bird lands like a parachute |
| 8 | Hebel | good gliding phase | leading edge removed |  |
| 9 | Bebel | good gliding phase | leading edge removed |  |
| 10 | Rebel | bad glidinf phase | leading edge removed |  |
| 11 | Rebel | bad gliding phase | leading edge removed |  |
| 12 | Pebel | good gliding phase | leading edge remored |  |
| 13 | Rebel | good gliding phase | trailing edge removed |  |

Table IX continued

| 14 | Rebel | good gliding phase | trailing edge removed |  |
| :---: | :---: | :---: | :---: | :---: |
| 15 | Rebel | gliding | trailing edge removed | 10w swoop |
| 16 | Eebel | good gliding phase | trailing edge removed |  |
| 17 | Bebel | good gliding phase | top of wing removed |  |
| 18 | Bebel | gliding | top of ring remored | low swoop |
| 19 | Rebel | good gliding phase | top of wing removed |  |
| 20 | Rebel | gliding | top of wing resoved | 10w swoop |
| 21 | Rebe 1 | good gliding | top of wing removed |  |

TABLE X
DATA OF THE TESTED OWL

| Weight | 1.52 lb |
| :---: | :---: |
| Wing area*) | $344.8 \mathrm{in}^{2}$ |
| Wing span | 38.1 in |
| Wing eweep angle | 32.7 deg. forward |
| Chord at half span | 11.0 in |
| Wing loading | $4.41 \cdot 10^{-3} 1 \mathrm{~b} / 1 \mathrm{n}^{2}=.635 \mathrm{lb} / \mathrm{ft}^{2}$ |
| Wing aspect ratio | 4.21 |
| ${ }^{*}$ ) Wing area includes the body area intercepted by the wing. |  |

## APPENDIX V

## BIONIC STUDY OF SILENT FLIGHT

## 1. INTRODUCTION

Owls hunt their prey during low levels of ambient light and location of their prey is by means of aural detection and penetrating vision. A frequent aource of food for the owl in the wood moume which has large ear lobes and reaponds quickly to any detectable noise, especially in higher frequencien. This information combined with the knowledge that the owl has a low attack speed implies that the owl must be capable of noiselems flight, This capability is needed not only to prevent his prey from becoming aiarmed, but almo, to hear the faint mounds of his prey, Upon these observations, ornithologists have reported the owl to be silent for many years.
R. R Graham [ 1] was the first to draw attention to the owl'meilent flight. He identified three pooularities common to the owl'milent filight. Uming a combination of thewe three features Graham proposed a hypothesis on the silent filight, However this hypothesim has not been verified nor proven invalid. Speculatione by others as to the reason for quiut flight have followed Graham' early euggentione.

The main questions which arime are

1. In the owl's flight unumually quiet?
2. If so, what can be profited by this information?
3. What charactoristica allow this wilent flying?
4. How can this information be tranalated into technology'f

Theme and aimilar questions are investigated in the research described in this atudy.

## 2. THE BIONIC-CYBRRNETIC APPROACH

The paet half century has been oharacterized by mcientific permonnel becoming more and more mpecialized in their area of work. It is not clearly understood how such highly epecialized knowlenge by itself can be productive to human society. On the other hand a too euperficial knowledge of many subjectm can produce ifttie or no practical applications. This indicatias that true progress can be accompliahed only with a composite approach.

Bionics is a new way of analyzing problems of living systems and machines by the pooling of the work of the biologist, the psychologist, the mathematician and the engineer. When Steele [31] introduced the word bionios in 1958 he gave it this definition:

It is the science of systems whose function is based on living systeing, or which have characteriatics of living aygtema, or which resemble these.

Gerardin [32] has expanded upon this definition to show the active role it must play in the future of mankind when he mald.

> Bionica is the result of bringing together the analytical activity of biologiats and the synthetic activity of the engineers. It is asystematic study of the behaviour of living mechanisms so that the principles discovered may be adapted for use in manmade aystems.

A well phrased 111 utration of the engineer's dependence upon his own knowledge and his concept of an innate ability to creato is given by Hertel [33]
Since Nature has 'unlimited time and reatourcea'
and has been 'extravagant and wasteful' in build-
ing the presont, it in not therelore, murprising
that Nat.ir has developed myetems and components
which aro incomparably more advanced and superior to
all th: l Homo tapiens ham conceived and Homo Fabor
ham deviaed and bulit, It appears to me as an over
abundance of preaumption and a lack of reverence for
oreation that we human beluge neglect or even refuae
to admit the inferiority of our methode to the ways
of Nature.

This strong emphasis on the systems approach requires that cybernetics be applied whenever it contributes to comprehension. Where cybornetice is,

The science which utudies the communication and the processes of control in living organisma and machines
as defined by Wiener [34].
A more recent definition that has been $\leq 1 v e n$ by Helvey [35] 1m mimply,

Cybernetics is the mcience of interactions.
To emphanize the need for a mymems approach to problem solving, the following is quoted from the program of the Communiet Party of the Soviet Union [36]
... cybernetice and electronic computer and controd
instruments will receive wide employment in the production processes of industry, construction, and transportation, in scientific research, in planning and designing calculations, and in the sphere of metering and control.

In order to transfer bionic information from a living system to mechanical technology it is necessary to have as much knowledge of the living model as possible. Otherwise the model and the prototype will not be isomorph and the results of one could not be expected to follow from the other. Because of the desire to transfer those characteristics of the owl which contribute significantly to its blient filght into airpiane technology it is necessary to be familiar with the filght dynamics of the owl as well as the "design characteristics" of its wings and body.

## 3. BIOLOGICAL STUDY OF THE OWL'S FLIGHT MECHANISMS

Every motion of the bird in flight, every change in shape and position of the feathers, is designed to extract energy from the air and use it to eatablish the desired flight. The wing and feathers of the bird in flight use the eame principles and mimliar mechanism, such as: wings, propellers, ste日ring and high-lift devicem in much the mame way an an airpiane.
a. Biologioal Characteristios of Bird Locomotion

To appreciate the analogy between bird and airplane flight it is important to know the basic construction of the wing of a bird, Figure 66. From Figure 66 we can aee that the wing lis of the same basic construction am the human arm, the leathers being attached to the "forearm" and the "hand, with the small "alula," or "bastard wing" constructed as a "thumb." The feathers on the hand are known as "primaries," and thome on the forearm are "ecocondarien." The "tertiary" feather" grow from the "elbow." The "scapulars" grow from the acapular membrane, which reaches from the "shoulder" out to the elbow. A strong elamtic membrane of two layerm of ekin, the "patagium," etretches between the ghoulder and the "wrist." The feathere along the front edge of the wing grow from this membrane, which holde them in position to give the wing a straight, otreamined leading edye, even when the elbow if partially bent.

The bird'e wing acte not as a angle piece of flying equipment but an a composite, each part having a different function or movement. The inner portion of the wing, from shoulder to wrist, corresponds to the wing of an arplane. it moven comparatively little during flight but, like the wing of an airplane, suppiles the required lift. The outer section of a
bird's wing, starting at the wrist, constitutes the propeller and is also used for control surfaces.

Bird flight can be divided into two madn seotions, glidink flight and flapping flight. The later division covering normal "powered" flight, take-off and "aerobatics," which are umually apecial maneuvera used in catching prey or in combat. For comm parison with alrplane flight flapping filght can be diaregarded.


#### Abstract

With only slight muscular activity the bird can change the angle of attack of his wing in gilding filght. With equal ease it can vary the camber, the span and the sweep angle thus affecting the lifting gurface, the lift-to-drag ratio and the control aurfaces.


The shape of filght feathers (remiges) in action is the rew ault of many different factors. The design of the "vane", Figure 67, and the way it responds to alir pressure; the design of the "barbs" which make up the vane and affect its elasticity and shape; and the design of the "ghaft." The vane, made up of parallel row of barbs, stands out on oach side of the "rachis" cloping toward the tip. From each side of the barbs rows of "barbules" slope out toward the feather edge.

Bird feathers have a remarkable number of quality design standards. They are lightwejght which is in part duo to the hollow or "foam" filled shaft and barbas. Through the use of an extremely thin-walled construction with tho lightweight "foum" for aupport, the feathers show a high load-bearing capacity, Optimum shape, power cross-sectional dimensions and excelient materials resulta in a high degree of atifiness, alastiaity and flexibility. They reaint damage because of the many divisions of pliant parts, Each barb may completaly separate from the one next to it and be joined back by morely making contact agaln.

It is especially interesting to note how Nature has handied, In the flight feathors, the aeroelastic problem of "induced airfoil oscillations," The followhag filght characteristics in birds combine to produce a "critical velocity" higher than the maximumfilght velocity. The oritical velocity is that velocity at which oscillations are generated which may rupture vibrating parts. The features which have beon perfected in the bird to obviate flutter are: good resistance to feather torifon and bending due to the hollow cross section of the calamus; the ultra-ijift construction of the vanea to ensure minimum moment of mass about the rachis axis; the form of the remigea, with narrower anterior vane sections and broader posterior vane sections permite the resultant of the aerodynamio forces to lie behind the shaft. Consequently, the aerodynamic axis (the center-ofpressure locus) lies behind the centroidal axis which in turn falls behind the torsion axis.
b. Unique Features of the Owl's Flight Structure

With the above knowledge of the nerodynamic role of feathers and the winge it enables us to analyze the unusual characteristics of the owl which allow it to fly allently. Following is the list of the three unique characteristics of the owl's wing which were listed by Graham [1].

1. The. Leading Edge Comb. There is a remarkably otifi, comb-like fringe on the front margin of evexy feather that functions as a leading edge of the wing. The teeth of this comb are extensions of the barbs of the anterior part of the vane.
2. The Downy Upper Surface. Both the anterior and the posterior parts of the upper aurface of the feathers are covered with a thick, flufiy down-1ike covering.
3. The Trailing Edge Fringe. Along the trailing edge of the wing and of each primary feather, there is a fringe with ragyed outilne. The fibers of which it is formed are extensions of the barbs that make up the posterior part of the vane.
c. Suggested Biological Explanation of the Owl's silent Flight

In all previous attempts to explain the silent flight of the owl it has been stressed that the "hooked comb" on the leading edge of the leading remegis functions somehow to suppress noise. This emphasis on the leading edge comb has resulted from the lack of same on the fishing owl of Asia (Ketupa flovipes). This owl differs from the others in being a fish preying bird and therefore not requiring any ailencing mechanism because its prey in submerged and unable to hear the owl.

Graham [1] fet the pace on emphasizing the assumed importance of the leading comb as the prinary apparatus in reducing the radiated noise. His conclusion was that the leading edge comb served as a false leading edge of the wing and performed the task of gradually slowing down the velocity of the air which flowed through the comb and over the top of the wing. The benefit of thia, he concluded, was to amooth out the presbure gradient and thereby reduce any associated sound generated. Concerning the trailing edge firinge Graham postulated that the fringe allowed a mixing of the upper and lower stioamilnes in such a way that the noise-producing vortices did not form. 'The down-like texture of the feathers allowed them to move relative to one another without generating mechanical noise, Graham concluded.
A. Rapset [37] whu had done much research in experimental bird aerodymamics, formed an analogy between we leading edge comb and a swinging wile. He noted that two pieces of wire twisted together and swung through the air will generate noiae at a much lower intensity and frequency than a single piece of wire. He suggested that the toothed leading edge acts in the same way, i.e., in a manner to reduce the vortex noise emitted by the flow over the wing.

## 4. EXPERIMENTAL AERODYNAMIC INVESTIGATION OF THE GLIDING OWL

Free-flying birds are capable of a variety of steady and unsteady modes of flight. Some birds have exceptional abilities in flapping flight during takemoff, other birds are able to soar for long periods of time apparently with little effort yet others have outstanding maneuvering abilities.

## a. Types of Gliding Flight

Knowing the practical limitations of tactica' aircraft and being cognizant of the definition of bionics this project was linited to the investigation of gilding flight. Gifding flight is that steady state motion in which no propulsive forcea are supplied by the birds muscles or the aircrait's propulsion unit. There are three ways [38] by which an air vehicle may fly in the gliding mode. First the vehicle may maintain or gain altitude by gilding in an air mass that has an upward velocity component equal to or greater than its sinking speed, $V_{g}$. The second condition of gliding flight is described by the air vehicle flying in an air mass that has a horizontal velocity. In this condition the vehicle maintains its altitude if the horizontal velocity is sufficient to make the following equation an equality

$$
L_{1}=W=1 / 2 \rho S V^{2} C_{L}
$$

where $L$ is the iift force, $W$ is the weight of the vehicle, $\rho$ is the air density, $S$ is the appropriate wing area, $V$ is the gilding velcicity and $C_{L}$ is the nondinensional iff coefficient which is a function of the airfoil shape and the angle of the airfoil to the glide path.

The first and second cases of gliding filight are dependent upon the aforementioned dynamic atmospheric conditions. In the third case the gliding flight is restricted to flight in motionless, air masses, but is influenced by the initial gilde conditions, gravitional forces and the vehicle's aerodynamic
characteristics. In this gliding mode the path of flight is inclined downward with the angle $\varnothing$. The gliding velocity and the angle of inclination are the main factors for the determination of the aerodynamic characteristics of the gliding vehicle. In this respect the acceleration forces are negligible (see Reference 39).

This third type of gliding filght can be investigated by aliling the vehicle in the open atmosphere when vertical or horizantal air currents are negligible. This approach was used by Raspet [37] and Farrar and Farrel [40] in the early mornings or late afternoons to study the gliding characteristice of the buzzard. This condition can be simulated by investigating the gitding vehicie in a wind tunnel. This procedure has long been used to study mechanical vehiciea but only recentily living bircls [39, 41] have been trained to glide free in a wind tunnel.

## b. Bionic Need for Aexodynamic Characteristics

Several atudies ( 42 through 44) have been executed to determine those aerodynamic properties, which, when changed, result in a different level of noise and its frequency produced by the flying aircraft. The apparent parameters include the general contour of the vehicle such as cavities, protrusions and aimilar variations in body geometry which may generate sound during flight. However, often the mechanism of sound generation is not well undergtood. The dependence of sound production on aircraft speed is recognized but the relationship describing the dependence is still being investigated. Researchers have suggested that the noise varies according in some function of the change in wing loading, aspect ratio, drag, and Reynolds number, Investigation of this function or scaling parameter was not included in the scope of this study therefore no conclusions are drawn. The aerodynamic parameters in this study were established to provide a foundation from which the acoustic data through a bionic approach can be transiated in the correct scaling factors for sailplanes.

## c. Experimental Aerodynamic Procedure and Data Reduction

In the gliding filight of the owl the necessary acoustic data were recorded in open atmospheric conditions instead of in a wind tunnel. This decision was influenced by the readily available roon in whinh the owls could fily freely. This decision was also encouraged by the long training which would be required to attain succesaful filghts in a wind tunnel. Also the necessary low turbulence wind tunnel was not available, which would be required to obtain valid acouatic data.

In order to utilize the living owls as instruments to
establish experimenta aerodynamic and acoustic data extreme difficulties had to be overcome. The objective was to have the owls fly a repeatable course and have a maximum distance in a gliding mode. The measurements were made in the room illustrared in Figures 68 and 69. After prolonged training of the owls the flight wan standardized by forcing the owls to fly from a high perch over a gitring barricade to a low perch. The altitude difference between the low and high perches was 5 feet and the horizontal distance was 38 feet,

It was not possible to train the owls to leave the high perch and gilde the angle of 7.5 degrees to the low perch because the owls after leaving the high perch flapped their wings until they were very near the low perch and they took a variety of routes to the lower perch. Furthermore, they landed on every possible protrusion in the room. Therefore, every object was removed or covered in a manner to encourage the owls to land only on the two perches. The low perch was consistently used as a feeding place. Much experimentation was necessary to estabilsh an environment which forced the owle to $f 1 y$ the same route every time with minimal deviation. To achieve this a barricade was erected over which the owls had to fly to reach the low perch (see Figures 68 and 69). It was necessary that the birds see the lower perch at the time of departure from the high perch in order to determine the flight direction. Therefore, no solid barrier like a sheat of cloth or even transparent plagtic was satisfactory. Consequently, a loose white cotton atring barriuade was erected by hanging six feet long streamers approximately one foot apart from a horizontal string. It occurred very infrequently that the owls, whose wing span about three feet, tried to pass through the vertical strings. To achieve the maximum gilding distance the string barricade was systematically varied in position, both in the vertical and horizontal location.

In the dimly ift room the owls took off from the high perch in flapping filght to gain velocity, then pasising over the barrier they went into the gliding mode toward the low perch. About three feet before arriving to the low perch they changen the angle of attack of their wings and further slowed down their flight by using their tail and wing tip feathers as flaps and high drag devices.

The feeding of the owls was done once a day approximately at the same time. The main food was chicken gizzards and small balls made of the bone dust accumulated from sawing up ireah beef bones so that it provided, besides the bone marrow, the necessary calcium for the owla. Regular attontion had to be given to the water supply because the owle consumed much water. There were no difficulties during the nine months of working with the owls with parasites or diseases.

During the rcoustic experimentation with the owls the room was well lighted, but during the rest periods the light was
subdued. The windows were covered with dark paper to prevent the birds from flying against the glass.

The two Florida Baxred owls (strix varia alleni) were purchased from an animal dealer as adult birds. The experimental tests were conducted with only one of the owls. The physical properties of this owl are listed in Table XI.

Although a cage was available for the owls it was determined to be advantageous to leave the owls free in the room where they could use the two perches. This had the disadvantage that other birds, especially if they were smaller than the Barred owls, could not be used for experimentation because the Barred owls, being birds of prey, kilied them.

It was tried in a few caser to work also with hawks pretreined by their owners. However, they would not fly a course from which repeatable and meaningful flight dynamic data could not be obtained.

The Barred owls never became fully domesticated although sometimes they took food from the trainer'a hand.

Photographs of the gliding owl were taken in order to obtain the filght velocity, $V$, and the gliding angle, $\phi$. One method was through the use of the grid pattern on the wall and floor of the experiment room, established by eighth inch wide black tape. Furthermore, the locations of the camera and lights relative to the floor and wall were known. The position of the flying owl at two times in each flight was estabilshed by sequential triggering two electronic light flam units mounted on the ceiling of the test room. They provided, through capacitor discharge, sufficient light, of $1 / 2000$ second duration, to illuminate the owl and the turntable which was used for timing (Figure 70). The atrategic location of the flash units permitted shadows to be cast on the grid on the floor and wall. Difiticulty was encountered in developing the photographic system since the location of the bird at the flash times was determined from the shadows cast on the wall and floor. The positioning of these shadows was controlled by the location of the electronic flash units, the position of the camera, filght path of the bird and time of flash triggering. The triggering of the two flash units was accomplished via individual remote switches operated by an observer. To use the entire gilde time for determining the velocity the flash operator antlcipated the beginning of the glide phase and triggered the first flash. The mecond flash unit was triggered just as the owl broke the gilde mode, but before it began to flap it's winge for landing. Many flighte were made to adapt the observer to trigger the flash units at the proper times. The time interval ( $\Delta t$ ) was determined by reading the percent of a revolution that the constant speed turntable made between dual flamhes which were made visible by a dark line on the white turntable. The apeed of rotation of the turntable was standardized prior to each test.

The vertical, longitudinal and lateral positions were then measured on the photograph (Figure 70) to finally estabilah by triangulation the distance traveled along the glide path. The numerical procedure for determing the total distance traversed in time $\Delta t$ was obtained from the following relationships where the letters $X, Y$ and $Z$ define the apparent position as determined from the photographs, whereas, the starred letters denote the actual coordinates of the bird. The other alphanumerice refer to physical dimensions of the room and equipment locations; (see F'gures 71 and 72).

$$
\begin{gather*}
\frac{X^{*}}{X}-\frac{R-Z^{*}}{R}  \tag{1}\\
\frac{X^{*}-C}{Y-C}=\frac{R-Z^{*}}{R}  \tag{2}\\
\frac{3-Z^{*}}{3}=\frac{H-Y^{*}}{H} \tag{3}
\end{gather*}
$$

Solving from $Z^{*}$ from Equation (3) we find

$$
\begin{equation*}
Z^{*}=3-\frac{\left(H-Y^{*}\right)(3-Z)}{H} \tag{4}
\end{equation*}
$$

where, from Equation (2)

$$
\begin{equation*}
Y^{*}=C+\frac{\left(R-Z^{*}\right)(Y-C)}{R} \tag{5}
\end{equation*}
$$

mubitituting Equation (B) in Equation (4) and solving for $Z^{*}$ given

$$
Z^{*}=\frac{R[H Z+X(3-Z)]}{R I I+(3-Z)(Y-C)}
$$

thus knowing $Z^{*}$ the expressions for $X^{*}$ and $Y^{*}$ can be obtained from Equations (1) and (2) rompeotively in terme of physical dimenelons, measured apparent valuas and known actual coordinate valuea.

Conditionm at the first and second flashem are denoted by
the subscripts (1) and (2) respectively, By taking the distance between $X_{1}^{*}$ and $X_{2}^{*}$, defined as $D X^{*}$ and similarly for $D Y^{*}$ and $D Z^{*}$ the actual flight distance, $\Delta t$, in time $\wedge t$ is:

$$
\Delta D=\left(D X^{*}\right)^{2}+\left(D Y^{*}\right)^{2}+(D Z *)^{2}
$$

Therefore knowing $\Delta D$ and $\Delta t$ the flight velocity, $V$, which is aspumed to be constant, is oalculated as

$$
V=\Delta D / \Delta t
$$

The gilde angle, $\sigma$, is ohtained from the following relationahip:

$$
\phi=\arctan \left(D Y^{*} / D X^{*}\right)
$$

The glide angle and the gliding flight velooity are tabulated in Table XIII.

Another method of obtaining the flight velocity and the glide angle was through the use of high apeed movie camera (48 frames/eeo). From this film the position of the owl relative to the wall grid, was recorded (ees Figure 73) and plotted. For that portion of glide which was along a stralght path the glide angle and the velocity were determined. Extracting data from the movie film was difficult because of inadequate ifghting arrangements.

The velocity was calculated by multiplying the movio frame rate times the ratio of the dimtance travereed to the number of frames during the straight gliding rogion of the ilight. The following velocity calculation example using data from Figure 73 illustratem the method:
(48 framen/mec) (3.4 ft/11 framen) = $23.6 \mathrm{ft} / \mathrm{mec}$

The nominal location where the owl departed from straiglat gliding flight path wan emtablished from the movie data relativo to the wall grid. It was noted that the mesond flanh of the dual flamh temt wam frequently made when the owl war paret the end of the etraight flight. From the flightw where data were recorded almultancousiy on dual flam photographe and on movie ilim it
was determined that the data from the dual flash photographs ahould be adjusted to account for the delayed second flash of the dual flash test. The velocity from the dual flash experiment has been increased by 49 percent to make it consistent with tho movie data. The aerodynamic data from the movie film experiment is presented in Table XIV.
d. Discuasion of Experimental Aerodynamic Resulte

Aerodynamic forcen are deacribed in the conventional manner as ilft, $L$, and drag, $D$, which are respectively perpendicular and parallel to the bird'a flight path through the air. The following amemptions are made relative to their gilding filght:

1. The gliding ie, due to compenation by the bird, at conetant velocity, therefore no acceleration consideration is required.
2. The flight path is a etraight line. At these condition the ifft and drag forcea are balanced by weight, W, of the bird (seo Figure 74.)

If $\sigma$ is the angle of the flight path to the horizontal (the glide angle) then,

| and | $L=W \cos \alpha$ |
| :--- | :--- |
| where | $D=W \ln \phi$ |
|  | $\phi=\tan ^{-1} \frac{D Y^{*}}{D X^{\top}}$ |

Lift and drag forces are related to non-dimenaional coefficiente by the equations
and

$$
c_{L}=2 W /\left(p s v^{2}\right)
$$

$$
C_{D}=2 D /\left(\rho 8 V^{2}\right)
$$

where $C_{L}$ and $C_{D}$ ure the ifit and drag coefficiente respectively.
The bamio performance of an airoraft during equilibrium cilding in etilimir is demoribed by theme parameterm; air gpeed, glide angle, madminking mpeed relative to the air. The minking speed equation is:

```
Sinking speed mV/(L/D)
```

An important parameter in the description of gliding per. formance is Reynolde number, Re, because it is functionally related to the lift and drag coefficienta.

The Reynolds number it defined as:

$$
\operatorname{Re}=\frac{V C}{V}
$$

Where $c$ ds characteribtic length amsociated to the airfoil. The length of the wing chord at half apan was choosen for $c$. The kinematic vigcomity, $v$, of air was calculated at ambient conditions.

The aerodynamic remulte are mumarized and tabulated in Tables XIII and XIV. The mean value of the remults from the
 The etandard doviation, $\sigma$, which im based on the arithmetic mean is dofined in the following equation [45]

$$
0\left[\frac{1}{N-1} \sum_{1=1}^{n} x_{i}^{2}-\frac{1}{N(n-1)} \sum_{i=1}^{n} x_{i}^{2}\right]^{1 / 2}
$$

From these meari values the following parameters are calculated,

$$
\begin{aligned}
& \text { Re }=1.31 \times 10^{5} \\
& L / D=2.25
\end{aligned}
$$

The mean value of the aerodynamic data may be used in any future ecaling of the data but the large deviationm in the measured parameters should be taken into consideration. These data show that the owl in certainly a "low performance" ilyer. This in especially noted in the gliding ilight when compared to a gliding black buzard (Coraryps atratus) which has a lift-tom drag ratio range from 15.0 to 20.0 over its normal filght velocity range \{37\}. From the eame reference the sinking apeed apeed of the buzeard in its normal flight velocity range ia roughly $3.0 \mathrm{ft} /$ eec whereas for the owl the mean sinking epeed is
$10.94 \mathrm{ft} / \mathrm{sec}$. This comparison shows that the owl is not as well adapted to high performance gliding filght as in the black buzzard.

## 5. OWL MODIFICATIONS FOR ACOUSTIC TESTS

After studying the movie and still photographa and coneidering the potential quieting mechanimm of the gilding owl a methodology was developed for the purpome of determining the effect of variou physical features of the owl which enhance or damp the radiated nolse. Numeroun acountic tests of the gliding owl were made with the bird in an unmoditied condition. These data were obtained on three different occasions and on the lant teat yetematic modification of the owl's phyuical characteristice were made.

Figure 75 how the owl wing in an unmodified conaition. The firet modification (Figure 76) war uade to the wing by romoving the leading odge comb am shown in Figure 77. Following this modification the noime produced during gliding flight was recorded. Next, major portion of the trailing edge (approximately $3 / 4$ inches) was removed to provide a conventional atraight trailing edge (Figure 78). Finally, upon completion of the acoldetic tent following the trailing edge modification a large feather panel waw removed (Figure 79) from the upper murface of each wing. It was suspected that these contour feathera gave the owl the ability to reduce the ncountic output by virtue of its compliance. Acountic recordinge were made following thia adjurtment completing the biolica experimental effort uming live birde.

## 6. CONCLUSION OF BIONICS STUDY

a. A kreat effort wam required to train the owls to perform deaired program repeatedly, Only after acclimating to the experiment room and the asmociated equipment would they fly a reproducenble couree. Short term attempte with hawk were unrewarding.
b. Variations in the configuration characterietion of the owl from flight to flight remulted in mignifioant variation of the aerodynamic flight performance factors.
c. There are a variety of featurea ameociated with the owl feathere which indicate the importance of compliance in acoustic quieting of the wing.
d. The physical modifictions of the owl'm wing did not
appreciably affect his performance characterlstics.
e. Application of information gained from the owl studius to airplanes can only be made when the results of all phases of thim projeot are considered.

Only a brief discumaion of the Bionios input to the total program were included in this Appendix. The Bionics approach is implicit throughout other mections of this report.

Figure 66. Basic Construction of a Bird Wing.


Figurc bit Paらiot Festures of a Featie:



Figure 69. Procile View of Experiment Room.


Figure 70. Dual Flash Photograph of a Gilding Owl.


Figure 71. Sketch Looking Into Flight Path of Owl at Flash Number 1.


Figure 72. Sketch Looking Down on Flight path of Owl.

Figure 73. Typical plot of the Movie Data Results.


Figure 74. Forces on a Bird in Equilibriun Gliding.


Figure 73. Photograph of an Unmodifiod Owl Wing.


Figure 76. Photograph of the Removal of the Leading Edge Comb from an Owl Wing.


Figure 77. Photograph of an Owl Wing with the Leading Edye Comb Removed.


Figure 78. Photograph of an Owl Wing with the Trailing Edge Fringe Removed (Top View).


Figure 79. Photograph of an Owl Wing with the Upper Contour Feathers Removed.

TABLE XI
PHYSICAL PROPERTIES OF TESTED OWL

| Parameter | Value |
| :--- | :---: |
| Weight, lb | 1.52 |
| Wing area*, in. ${ }^{2}$ | 344.8 |
| Wing apan, in. | 38.1 |
| Wing aweop angle, deg | 32.7 forward |
| Chord at half apan, in. | 11.0 |
| Wing loading, lb/ft ${ }^{2}$ | 0.635 |
| Wing aspect ratio |  |
| *Wing area includen the body area intercepted |  |
| by the wing |  |

## TABLE XII

## MEAN VALUE AND STANRARD DEVIATION OF SEVERAL AERODYNAMIC PARAMETERS OF THE OWL

| Parameter | Mean Value | Standard Deviation |
| :--- | :---: | :---: |
| Velocity, ft/sec | 24.04 | $\pm 4.26$ |
| Glide angle, deg | 25.86 | $\pm 3.99$ |
| Lift coefficient | 0.027 |  |
| Drag coofficient | 0.012 |  |
| Sinking epeed, ft/aec | 10.94 |  |

TABLE XIII
arrodynamic data of gliding owl
FROM DUAL FLASH PHOTOGRAPAS

| $\begin{aligned} & \text { Date } \\ & 1970 \end{aligned}$ | Flight Nuaber | Velocity, ft/sec | $\begin{gathered} \text { Glide Angle, } \\ \text { deg } \end{gathered}$ | $\begin{aligned} & \text { Reynolds } \\ & \text { Nuaber } \\ & \times 10^{-6} \\ & \hline \end{aligned}$ | Lift-to- <br> Drag Ratio | Lift <br> Coef | Drag Coef | Sinkin Speed, ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 11/22 | 2 | 25.25 | 19.09 | 0.136 | 2.888 | 0.025 | 0.008 | 8.74 |
| 11/22 | 5 | 22.26 | 23.19 | 0. 20 | 2.333 | 0.031 | 0.013 | 9.54 |
| 11/22 | 3 | 20.04 | 22.84 | 0.108 | 2.373 | 0.039 | 0.016 | 8.44 |
| 11/22 | 6 | 22.44 | 21.09 | 0. 121 | 2.592 | 0.031 | 0.012 | 8.65 |
| 11/22 | 7 | 23.52 | 22.07 | 0.127 | 2.465 | 0.028 | 0.011 | 9.54 |
| 11/22 | 9 | 21.86 | 26.57 | 0.118 | 1.999 | 0.032 | 0.016 | 10.93 |
| 11/22 | 10 | 21.60 | 26.60 | 0.117 | 1.996 | 0.032 | 0.016 | 10.81 |
| 11/22 | 11 | 21.99 | 25.23 | 0.119 | 2.121 | 0.031 | 0.015 | 10.36 |
| 11/28 | 3 | 22.55 | 25.27 | 0. 122 | 2.118 | 0.030 | 0.014 | 10.65 |
| 11/28 | 4 | 20.61 | 24.70 | 0.111 | 2.173 | 0.036 | 0.016 | 9.48 |
| 11/28 | 5 | 21.08 | 23.19 | 0.114 | 2.333 | 0.035 | 0.015 | 9.03 |

TABLE XIII (continued)

| $\begin{aligned} & \text { Date } \\ & 1970 \end{aligned}$ | Flight Nuaber | Velocity, ft/sec | ```Glide Angle, deg``` | $\begin{aligned} & \text { Reynolds } \\ & \text { Number } \\ & \times 10^{-6} \\ & \hline \end{aligned}$ | Lift-toDrag Ratio | Lift Coef | Drag Coef | Sinking <br> Speed <br> ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 11/28 | 6 | 22.24 | 22.71 | 0.120 | 2.388 | 0.031 | 0.013 | 9.31 |
| 11/28 | 7 | 21.83 | 25.40 | 0.118 | 2.105 | 0.032 | 0.015 | 10.36 |
| 11/28 | 13 | 21.96 | 25.20 | 0.119 | 2.124 | 0.032 | 0.015 | 10.33 |
| 11/28 | 14 | 20.55 | 24.37 | 0.111 | 2.206 | 0.036 | 0.016 | 9.31 |
| 11/28 | 15 | 19.86 | 23.74 | 0.107 | 2.273 | 0.039 | 0.017 | 8.73 |
| 11/28 | 16 | 21.12 | 22.93 | 0.114 | 2.363 | 0.035 | 0.014 | 8.93 |
| 11/28 | 17 | 21.27 | 23.95 | 0.115 | 2.250 | 0.034 | 0.015 | 9.45 |
| 12/5 | 2 | 21.58 | 25.34 | 0.117 | 2.110 | 0.033 | 0.015 | 10.22 |
| 12/5 | 3 | 24.27 | 24.17 | 0.131 | 2.228 | 0.026 | 0.011 | 10.89 |
| 12/5 | 4 | 25.27 | 26.17 | 0.137 | 2.034 | 0.023 | 0.011 | 12.42 |
| 12/5 | 6 | 26.74 | 26.15 | 0.144 | 2.036 | 0.021 | 0.010 | 13.13 |

TABLE XIII (continued)

| $\begin{aligned} & \text { Date } \\ & 1970 \end{aligned}$ | Flight Number | Veiocity, ft/sec | $\begin{gathered} \text { Glide Angle, } \\ \text { deg } \end{gathered}$ | $\begin{aligned} & \text { Reynolds } \\ & \text { Number } \\ & \times 10^{-6} \\ & \hline \end{aligned}$ | $\begin{aligned} & \text { Lift-to- } \\ & \text { Drag Ratio } \end{aligned}$ | Lift Coef | Drag <br> Coef | Sinkin Speed ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 12/19 | 5 | 25.30 | 21.67 | 0.137 | 2.516 | 0.024 | 0.009 | 10.05 |
| 12/19 | 6 | 25.54 | 22.63 | 0.138 | 2.397 | 0.023 | 0.009 | 10.65 |
| 12/19 | 7 | 25.68 | 2\%.2E | 0.139 | 2.119 | 0.023 | 0.010 | 12.12 |
| 12/19 | 8 | 25.11 | 22.22 | 0.136 | 2.447 | 0.025 | 0.010 | $10.2 \overline{6}$ |
| 12/19 | 12 | 19.22 | 26.93 | 0.164 | 1.967 | 0.040 | 0.020 | 9.77 |
| 12/19 | 13 | 22.59 | 21.81 | 0. 122 | 2.498 | 0.031 | 0.012 | 9.04 |
| 12/19 | 15 | 24.22 | 22.52 | 0.131 | 2.411 | 0.026 | 0.011 | 10.04 |
| 12/19 | 17 | 23.63 | 22.29 | 0.128 | 2.439 | 0.028 | 0.011 | 9.68 |
| 12/19 | 14 | 24.94 | 23.42 | 0. 135 | 2.307 | 0.024 | 0.010 | 10.81 |

TABLE XIV
AERODYNAMIC DATA OF GLIDING OUL
PROM MOVIE FILM

| $\begin{aligned} & \text { Date } \\ & 1979 \end{aligned}$ | Flight Number | Velocity ft/sec | $\begin{gathered} \text { Glide Angle, } \\ \text { deg } \end{gathered}$ | Peynolds Furiber $\times 10^{-6}$ | Lift-toDrag Ratio | Lift Coef | Drag <br> Coef | Sinking <br> Speed <br> ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 12/19 | 7 | 25.85 | 25.00 | 0.140 | 2.144 | 0.023 | 0.c20 | 12.05 |
| 12/19 | 10 | 22.15 | 26.09 | 0.120 | 2.041 | 0.031 | 0.015 | 10.85 |
| 12/19 | 11 | 22.70 | 23.50 | 0.123 | 2.299 | 0.030 | 0.013 | 9.87 |
| 12/19 | 12 | 21.82 | 28.29 | 0.118 | 1.857 | 0.031 | 0.017 | 11.74 |
| 12/19 | 13 | 21.62 | 29.39 | 0.117 | 1.774 | 0.031 | 0.017 | 12.18 |
| 12/19 | 14 | 24.00 | 26.00 | 0.130 | 2.050 | 0.026 | 0.013 | 11.70 |
| 12/19 | 15 | 24.00 | 27.69 | 0.130 | 1.904 | 0.026 | 0.013 | 12.60 |
| 12/19 | 17 | 23.41 | 27.79 | 0.126 | 1.896 | 0.027 | 0.014 | 12.34 |
| 12/19 | 18 | 23.53 | 26.50 | 0.127 | 2.005 | 0.027 | 0.013 | 11.73 |
| 12/19 | 20 | 23.61 | 26.59 | 0.127 | 1.996 | 0.027 | 0.013 | 11.82 |
| 12/19 | 21 | 24.00 | 24.50 | 0.130 | 2:194 | 0.027 | 0.012 | 10.93 |

TABLE XIV (continued)

| Date 1970 | Flight Number | $\begin{aligned} & \text { Velocity, } \\ & \text { ft/sec } \end{aligned}$ | $\begin{gathered} \text { Glide Angla, } \\ \text { deg } \end{gathered}$ | Reynolds Nulber $\times 10^{-6}$ | Lift-to- <br> Drag Ratio | Lift Coef | Drag Coef | Sinking Speed ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 12/19 | 22 | 25.45 | 23.39 | 0.137 | 2.310 | 0.024 | 0.010 | 11.0i |
| 12;19 | 23 | 21.82 | 28.19 | 0.118 | 1.865 | 0.031 | 0.016 | 11.69 |
| 12/19 | 24 | 24.20 | 25.00 | 0.131 | 2.144 | 0.026 | 0.012 | 11.28 |
| 12/19 | 25 | 23.66 | 24.39 | 0.128 | 2.204 | 0.027 | 0.012 | 10.73 |
| 12/19 | 26 | 23.66 | 24.29 | 0.128 | 2.214 | 0.027 | 0.012 | 10.68 |
| 12/19 | 31 | 25.04 | 27.29 | 0.135 | 1.937 | 0.024 | 0.012 | 12.82 |
| 12/19 | 34 | 23.66 | 23.79 | c. 128 | 2.267 | 0.027 | 0.012 | 10.43 |
| 12/19 | 37 | 24.53 | 23.29 | 0.132 | 2.322 | 0.026 | 0.011 | 10.56 |
| 12/19 | 39 | 23.04 | 23.29 | 0.124 | 2.322 | 0.029 | 0.012 | 9.92 |
| 12/20 | 2 | 22.68 | 25.19 | 0.122 | 2.125 | 0.030 | 0.014 | 10.67 |
| 12/20 | 4 | 24.00 | 22.00 | 0.130 | 2.475 | 0.027 | 0.011 | 9.69 |

TABLE XIV (continued)

| $\begin{aligned} & \text { Date } \\ & 1970 \end{aligned}$ | Flight Number | $\begin{aligned} & \text { Velocity, } \\ & \text { ft/sec } \end{aligned}$ | Glide Angle, deg | Reynolds Number $x 10^{-6}$ | Lift-toDrag Ratio | Lift <br> Coef | Drag Coef | Sinking <br> Speed <br> ft/sec |
| :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: | :---: |
| 12/20 | 6 | 24.41 | 26.50 | 0.132 | 2.005 | 0.025 | 0.012 | 12.17 |
| 12/20 | 7 | 23.23 | 26.89 | 0.125 | 1.971 | 0.028 | 0.014 | 11.78 |
| 12/20 | 8 | 26.67 | 25.39 | 0.144 | 2.106 | 0.021 | 0.010 | 12.66 |
| 12/20 | 9 | 21.82 | 25.59 | 0.118 | 2.087 | 0.032 | 0.015 | 10.45 |
| 12/20 | 10 | 24.49 | 26.09 | 0.132 | 2.041 | 0.025 | 0.012 | 11.99 |
| 12/20 | 15 | 24.71 | 20.59 | 0.133 | 2.660 | 0.026 | 0.009 | 9.28 |
| 12/20 | 21 | 24.89 | 23.89 | 0.134 | 2.256 | 0.025 | 0.011 | 11.02 |

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[^0]:    $l_{\text {All sound presente levels (SPL) in this report are referenced }}$ to 0,0002 microbar.

[^1]:    *For smali angles of attack only.

