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NAI-55-857

RESEARCH AND REPORTS ON LAMINAR FLOW BOUNDARY LAYER CONTROL SYSTEMS

CONTRACT AF33(616)-3168
TASK NO. 13618

PROGRESS REPORT FOR PERIOD
15 SEPTEMBER 1955 THRU 31 OCTOBER 1955

REPORT TO WRIGHT AIR DEVELOPMENT CENTER

NORTHROP AIRCRAFT, INC.

HAWTHORNE, CALIFORNIA

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NORTHROP AIRCRAFT, INC.

PROGRESS REPORT ON CONTRACT AF33(616)-3168

RESEARCH AND REPORTS ON LAMINAR FLOW BOUNDARY LAYER CONTROL SYSTEMS

I. BASIC CONTRACT PURPOSES

To supply the necessary personnel, services, and facilities to investigate laminar boundary layer control on wings and bodies through suction and to develop methods for the design and construction of a laminar boundary layer control airplane.

II. FACILITIES AND EQUIPMENT

There has been no change in the facility and equipment situation during the period covered by this report.

III. ORGANIZATION AND PERSONNEL

A. Organization

There have been no organizational changes during this period.

B. Personnel

The staff now engaged full time on this contract consists of the following:

Engineering

Supervision	2
Clerical and Secretarial	2
Direct Charging	27
	31

Shop

Supervision	1
Clerical	1
Direct Charging	16
	18

Flight Test Department

Direct Charging (2 shifts)	7

IV. VISITORS

Dr. Max G. Scherberg, Asst. Chief Scientist, Wright Air Development Center, Wright-Patterson Air Force Base

Mr. E. C. Luthy, Aircraft Gas Turbine Specialist, General Electric Company, Los Angeles, California

Mr. Wm. B. Blaufuss, Field Installation Engineer, Pratt and Whitney Aircraft Division, Beverly Hills, California

Mr. C. A. MacGregor, Engineering Supervisor, Propulsion Research Corporation, Santa Monica, California

Mr. F. E. Hoffman, Senior Engineer, Propulsion Research Corporation, Santa Monica, California

Col. D. H. Heaton, Hq. USAF, Washington, D. C.

Lt. Col. W. P. Maiersperger, Hq. USAF, Washington, D. C.

Mr. B. A. Hohmann, Wright Air Development Center, Dayton, Ohio

V. TECHNICAL PROGRESS

A. Theoretical Investigations of the Laminar Boundary Layer on a Swept Wing

1. Numerical Integration of Boundary Layer Equations on Swept Wings

The incompressible integration on one of the best pressure and suction distributions encountered on the F-94 flight test glove was resumed.

2. Numerical Integration of the Compressible Laminar Boundary Layer Equations on Swept Wings

The compressible integration for a typical pressure and suction distribution on the lower surface of an untapered wing swept 35 degrees and at 0.9 Mach number (insulated surface) was completed. Summary data, for analysis of the boundary layer development and later incorporation in a report, are being prepared from these results.

3. Calculation of the Stability of the Laminar Boundary Layer on a Swept Laminar Suction Wing

Work is being continued on the Blasius profile with the new and exact boundary conditions. Up to R = 500, as far as the calculations have been completed, the results agree very well with the data of Schubauer and Skramstad and somewhat better than the results using the approximate boundary conditions.

In the case of the rotating disk, a few calculations are being made with the amplification parameter c_i having values other than zero to test Stuart's suggestion that the traces observed on the china clay may be due to the most unstable disturbances of linearized instability theory. Two profiles are being examined, the critical and supercritical*, and the phase velocity c_r is being kept at zero, i.e., these wave components are stationary relative to the surface.

B. Basic Laminar Suction and Transition Investigations

1. Laminar Suction Experiments in 2-Inch Tube with Suction through 80 Rows of Holes

Work on these experiments was halted last month due to breakdown of the compressor, but the compressor has been repaired and the experiments are again underway. For the runs that have been completed, the drag coefficients (including suction power) of the "equivalent" airfoils have been calculated. These drag coefficients are approximately 12 to 100% (depending on the pressure rise) greater than the corresponding drag coefficient of a laminar flat plate of the same Reynolds number.

*The critical velocity profile is that profile where the point inflection is on the x-axis, i.e., c = 0 when w" = 0. The plane of this profile makes an angle with the radial plane of about 13°10'. The profile designated supercritical is the one in which c = 0 corresponds to the minimum, or critical Reynolds number in the neutral stability curve. The angle with the radial plane is about 9°59'.

2. Laminar Suction Experiments in an 8-Inch Low Speed Tube with Suction through Holes

Fabrication of the 7-row-of-holes test section has been completed and this section has been installed on the 8-inch low speed tube. The experiments with this section should begin next month.

3. Experiments in the 2-Inch Multisection Tube

Purpose: Investigation of suction through holes at various chordwise stations. The experimental setup is now complete and the first group of experiments has been started. The tube is designed with joints at various points along its length, so that an additional short section can be inserted in the tube at each of the stations where there is a joint. The short section contains the hole configuration and appropriate suction chamber for which the experiments are being conducted. Measurements of the critical suction quantities have been made for four different hole configurations. Each configuration consisted of a single row of holes going completely around the circumference of the tube, and included 80, 90, 100, and 110 holes per row. These experiments are a continuation of previous experiments, but they cover a wider Reynolds number range than formerly. It is hoped that the results of this program will help to make it possible to present the critical suction quantities for suction through rows of holes in the form of dimensionless parameters which can be used for design purposes. These experiments are also prerequisite for the experiments with finite length rows of holes.

C. Wind Tunnel Tests on Laminar Suction Swept Wing Model

Purpose: Preliminary investigation of a swept laminar suction wing model at moderately high Reynolds numbers. Swept laminar suction wings would be desirable from the standpoint of high subsonic cruising speeds provided laminar flow can be maintained back to the trailing edge. Theory indicates that somewhat increased suction quantities (as compared with an equivalent straight wing) should enable 100% laminar flow on swept wings, with but a minor sacrifice in wing profile drag and range. These theoretical expectations should be verified by experiment.

The design of a swept laminar suction wing model (to be mounted in the 8-foot by 11-foot NAI tunnel and in the Michigan tunnel) is now in progress. The panel under consideration has a thickness ratio of 12%, a 7-foot chord, and a wing sweep of 30°. A large number of fine suction slots will be installed in the region from 0.25 c to 0.95 c. Relatively weak suction will be applied in the region of slightly accelerated flow from 0.25 c to 0.6 c. Stronger suction will be necessary in the region of the rear pressure rise. A design wing chord Reynolds number of 7×10^6 and 10^7 was assumed for the experiments in the Northrop tunnel and Michigan tunnel, respectively. The suction quantities and the dimensions of the suction slots and holes located underneath the slots have been determined.

D. Bodies of Revolution

1. 96-Inch Model Without Suction Slots

The previous transition experiments on the 96-inch body of revolution indicate that disturbances in the front part of such a body are amplified at a faster rate than on a wing, under otherwise the same conditions. In order to increase the stability of laminar flow, the front part of the 96-inch body is being modified by the installation of a large number of fine slots between 5% and 21% length. A calculation of the boundary layer development in the front part of the body is being conducted to determine the suction quantities needed for those boundary layer profiles to remain at the theoretical stability limit at reasonably high length Reynolds numbers. The redesign of the model and the theoretical study are in progress.

2. <u>142-Inch Ellipsoid With Suction</u>

Installation of the instrumentation and assembly of the model are continuing. The front part of the body will be modified in the same manner as the 96-inch body of revolution.

3. <u>Preliminary Experiments on Wing-Body Interference</u>

No further progress has been achieved during the report period.

E. <u>Aerodynamic Investigations of Suction Ducting Systems</u>

The inlet flow measuring nozzles for the vee-inlet suction duct have been completed, and further investigation of this duct will be resumed during the later part of October, 1955.

F. <u>F-94 Flight Laminar Suction Experiments</u>

The flight program was interrupted for about four weeks due to inspection of the airplane, repair of the compressor turbine system by AiResearch, and replacement of some parts in the afterburner system.

Further evaluations of the flights for maintaining laminar flow with local supersonic fields along the glove proved that 100% laminar flow and reasonably low drag coefficients were obtained at a flight Mach number of M = .727 at 30,000 feet altitude. The maximum local Mach number at the glove surface was $M_{local} = 1.08$. The flow returned to subsonic speeds steadily, and the measured pressure distribution along the surface did not indicate the occurrence of a compression shock. It is not certain, however, whether or not the transition from supersonic to subsonic flow occurred through a series of multiple weak shocks.

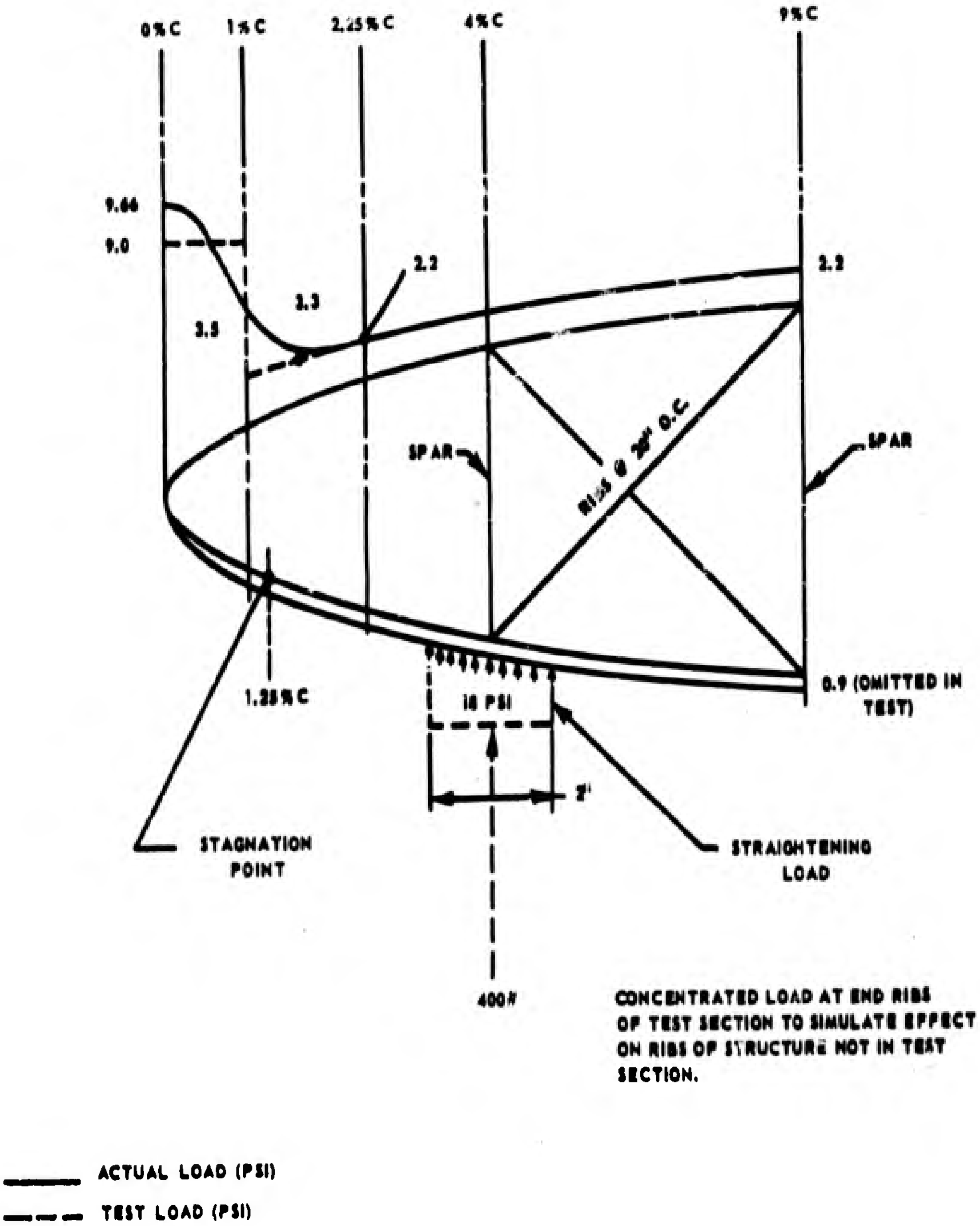
It is intended to extend the flight program by investigating the case of an aileron deflection in laminar flow. Modification of the trailing edge of the F-94 glove (.92 c to 1.00 c) is in progress for the purpose of simulating a flap or aileron configuration for small angles of deflection. Design layouts have been completed and details were released to the shop for fabrication.

G. Structural Investigations

1. Full-Scale Wing Segment

Calculations were made of camber distortion of a thin, cambered, highly stressed wing. It was found that camber straightening effected by wing bending imposes high vertical loads on the forward and aft section of the wing structure.

a. Design of the structure aft 0.6 chord as reported last month is now in the detail drawing stage.


0% C
1% C
2.25% C
4% C
9% C
9.66
9.0
3.5
3.3
2.2
2.2
SPAR
SPAR
RIBS @ 20" O.C.
1.25% C
18 PSI
2'
0.9 (OMITTED IN TEST)
STAGNATION POINT
STRAIGHTENING LOAD
400#
CONCENTRATED LOAD AT END RIBS OF TEST SECTION TO SIMULATE EFFECT ON RIBS OF STRUCTURE NOT IN TEST SECTION.
ACTUAL LOAD (PSI)
TEST LOAD (PSI)

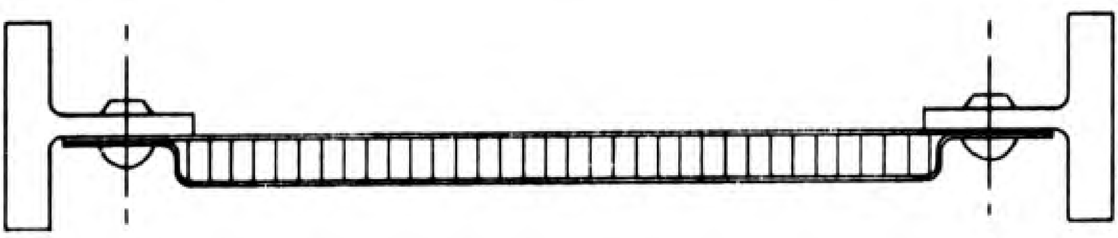
b. Design of a leading edge box beam is under consideration for the nose section.

A tension pad whiffletree is being designed for the nose section test specimen. A 3 g loading condition is being simulated normal to the upper surface; the ultimate synclastic or straightening loads have been evaluated and are applied to the lower surface at an arbitrary point.

Skin waviness is to be checked at 1.25 g.*

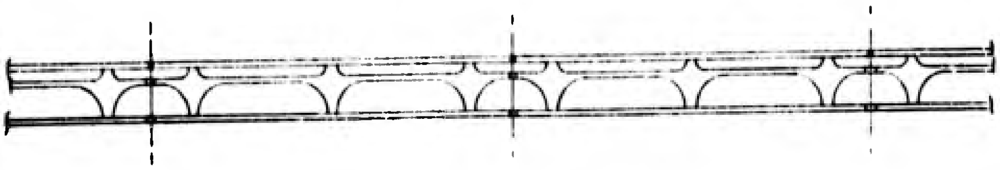
Test specimens for panel compression, simple support on four sides, and bending have been fabricated.

Aluminum honeycomb sandwich panels to be used as diagonals in the Warren trusses of the 60% c - 90% c section of the wing have been completed.



One type of sandwich skin under consideration for the suction area of the wing consists of .016 75ST faces bonded to a chemically milled 75ST core. The bond is Minnesota Mining AF 13 cured at 325° for 35 minutes under 12 psi vacuum pressure. The required holes in the core and lower face are drilled prior to bonding; the slots in the upper face are cut after bonding.

*The lower surface air load is being omitted in the test.



The calculated critical compression stress for this type of panel is determined by analogy with a corrugated sandwich.

$$f_{F_{eR}} = \frac{\pi^2 D_F E_F}{a^2 H}$$ (Ref: ANC 23, 2d Edition 1955, p. 96)

The transverse stiffness, a mechanical property necessary to evaluate "K", is computed from simple beam deflections with the plane of bending parallel to the slots.

The eccentric connections are to facilitate wing assembly. The illustrated specimens shall be tested as columns; calculated Euler loads, 4450 pounds for Specimen 1 and 2100 pounds for Specimen 2, are based on a 0.1-inch eccentricity. If erratic and generally unpredictable failures occur locally (in the vicinity of the neck of the connection) the design will be modified to eliminate the eccentricity.

The trailing edge, 60% c - 90% c, Warren truss rib is 3 inches wide and 19 inches long. The compression chord is essentially a .15 Plate (75ST) bonded between .016 skins. Diagonals (see sketch) are bonded to the upper and lower chords. A proposed cantilever test of the truss rib and adjacent spanwise structure will put an ultimate compression load of 5000 pounds in the forward diagonal. The skins adjacent to the upper or compression chord alternate honeycomb and chemically milled cores at about three-inch intervals in the forward-aft direction. The shear lag distribution in this type of orthotropic skin is of some concern and will be measured with strain gages.

The action and effect of the bonded truss joints on skin smoothness will be checked.

Report No. BLC-81, "Structural Design Considerations for Low Drag Boundary Layer Control," is enclosed as an appendix to this progress report.

H. Design Studies of a Hypothetical Long Range Laminar Suction Airplane for Very High Altitudes

1. A long range subsonic-supersonic bomber was studied. The purpose of this investigation was to study the feasibility of such a design under favorable conditions. Optimistic assumptions were made concerning drag, engines, weight, etc. The conclusion reached in the preliminary design studies of a subsonic-supersonic bomber is that a 100,000-pound or more gross weight airplane utilizing laminar suction for the subsonic cruise could deliver a 7000-pound bomb load over a 10,000-mile range mission including 1000 miles range at supersonic speeds. The configuration selected has a 1700-square-foot strut-braced wing with the 25% chord line swept back 25°. The mean wing thickness ratio is 3.5% and the aspect ratio is 7.5. Laminar flow was assumed over wing, tail, and strut, and over a large part of the fuselage. In the subsonic range the most critical aeroelastic problem is the forward shift of the flexible wing center of pressure, and the most critical supersonic problem is aileron rolling moment effectiveness. The configuration as designed by strength requirements has adequate aeroelastic

properties up to a dynamic pressure of 1 psi at a Mach number of 0.9, and 3.5 psi at Mach number of 2.0. The average subsonic lift to drag ratio of the airplane is 52 and the supersonic lift to drag ratio is 6.7. Two outboard wing-mounted J79 turbojet engines power the airplane for subsonic flight. Basically, the supersonic cruise requirements could be met with lightweight turbojet engines, designed for the low density at high altitudes and relatively short engine life and high cycle temperatures. By designing a low pressure ratio turbojet engine stressed to operate only at altitudes above 50,000 feet and at Mach numbers between 0.9 and 2.0, engine studies show that a specific engine weight of 1/4 the weight of a J79, and a specific fuel consumption 1.2 times that of a J79, might be possible. For transition and supersonic flight, eight turbojet engines with afterburners (yielding a total of 10,200 pounds thrust without afterburners at M = 2.0 at 65,000 feet) are mounted in two wing pods. Forward and aft subsonic fairings on these engine pods contain some of the fuel used for the subsonic cruise-out. At the beginning of transition to supersonic flight, these fairings are jettisoned and the auxiliary engines and afterburners are started. The afterburners would be used for acceleration to supersonic speed, but not during the supersonic cruise. At the end of the supersonic portion of the flight the engine pods are jettisoned.

The following table gives the configuration weight breakdown:

WEIGHT BREAKDOWN

Structure		
Wing at 7.2 lbs/ft^2	12,250	
Horizontal Tail	1,050	
Vertical Tail	700	
Fuselage	4,500	
Nacelles (2)	400	
Landing gear	3,000	
		21,900
Military Load		
Crew (3)	750	
Furnishings and Equipment	650	
Instrumentation and Radio	800	
Bombing and Navigation Equipment	2,700	
Bomb Load	7,000	
		11,900
Engines		
Engines (2 - J79's)	7,200	
Surface Control, and Hydraulic and Electrical Controls	3,000	
Fuel System	1,300	
Temporary Nacelles (2)	3,000	
Temporary Engines (8)	7,040	
Temporary Engine Afterburners (8)	2,360	
		23,900
Fuel (conventional)		
Climb	2,500	
Cruise-out	13,200	
Transition	2,000	
Supersonic	15,600	
Cruise-back	7,000	
Reserve	2,000	
		42,300
		100,000

2. Work on the strut-braced model for the tests at the Wright Field transonic tunnel is continuing. Unexpected difficulties were experienced during the machining of the cutouts at the lower wing surface in the region of the struts.

3. Design Studies on Wing-Strut Interference

In order to determine the minimum interference drag at a wing-strut intersection and the effect of flow over concave surfaces, design studies of a wind tunnel model of a wing-strut intersection have started.

I. Power Plant Studies

The work done during this period has consisted primarily of the following:

1. Collection and Evaluation of Data on Existing Power Plants and Their Components

Meetings have been held with representatives of Pratt and Whitney, General Electric, Westinghouse, Aerojet General, Bendix Aviation, Propulsion Research and the Rex Division of the Garrett Corporation. Revised performance specifications for the J79 turbojet engine have been received and are being evaluated. The performance specifications for the Westinghouse PD33 and the Pratt and Whitney J52 turbojet engines are also being evaluated for System I, described below, in the first laminar suction airplane. This has not been done before because the development of these latter engines appeared too uncertain. A design proposal for System I with the J79 engine was presented by the Propulsion Research Corporation and is being evaluated. The Rex series of engines proposed by the Garrett Corporation will be evaluated as soon as the necessary data becomes available.

Because there are several features in common between the Rex engines and the closed cycle engines under consideration, it appears most promising to study these systems as a group and consider possible variations in cycles and configurations which may lead to improved propulsion systems for aircraft with boundary layer control.

2. Subsonic Flight Performance Studies of Turbojet Engines Driving Suction Compressors through Special Turbine Stages (This type of power plant is referred to as System I.)

The effects of suction system pressure losses have been studied in detail for the cruise conditions being considered for a possible laminar suction airplane. Other flight conditions will be covered later as needed.

The performance of a power plant, consisting of a J79 turbojet engine driving a suction compressor with a bleed and burn system, has been estimated for the range of cruise conditions being considered for the first laminar suction airplane. Due to the mismatching of main engine components, the specific fuel consumption of this system is from 6 to 10% higher than that for System I without bleed system pressure drop, and 6 to 20% with maximum specified pressure drop based on engine rpm from 100% to 80%. However, it is to be noted that an engine specifically designed for a bleed and burn system could have similar performance to System I, and may therefore be advantageous in special future applications since it allows the suction compressors to be placed at stations other than the main engines.

3. Preliminary Design and Development of Suction System Components

(a) A basic design study of a suction compressor and a turbine for System I, meeting the requirements of a possible first laminar suction airplane and matching the J79 engine, has been nearly completed. This design is being used as a basis for selection of a suitable control system.

(b) A design study of a ram turbine-driven auxiliary suction compressor for boosting the low pressure suction air to the main suction chamber pressure has similarly been completed and compared with an auxiliary turbine driven by exhaust bleed.

(c) For a high altitude laminar suction airplane, the blade chord Reynolds numbers for the suction compressors are relatively low. The problem is then to design bladings which maintain a high stage efficiency and high lift to drag ratio down to low chord Reynolds number. In order to investigate such a compressor stage, a single stage (rotor plus stator) axial flow blower is being designed.

The experimental setup will be an open-circuit type, with radial inlet, upstream drive, and downstream throttling. Principal characteristics of the experimental compressor stage will be:

Tip diameter = 20 inches

Hub diameter = 12 inches

Number of rotor blades = 12

Pressure Coefficient $= \dfrac{\Delta P_{compressor}}{(\rho/2)\, U^2_{tip}} = .36$

Flow Coefficient $\phi_{rotor\ tip} = \dfrac{\text{axial velocity}}{U_{tip}} = .50$

Rotor tip Reynolds No. range = 30,000 to 300,000

A special high-sensitivity torque measuring device is being designed for these compressor experiments.

(d) Preliminary low Reynolds number experiments were conducted with a single airfoil of 5% thickness and approximately 5% camber.

The following preliminary results (subject to slight corrections pending final evaluation) are as follows:

At $\alpha = 4^{\circ}$ and $R_c = 60000$, the wing profile drag was relatively high as a result of rear laminar separation ($c_{D_\infty} = .031$). The profile drag was considerably reduced by artifically forcing turbulent reattachment in the rear part of the upper surface of the model. Five masking strips of 0.18-inch width were taped from 0.27 c to 0.8 c (c = 5 inches wing chord). A free vortex layer then formed downstream of each strip, causing increased instability of the laminar boundary layer, earlier transition, and turbulent reattachment ahead of the wing trailing edge. The table below gives a summary of preliminary drag results with these strips.

R_c	60000	52000	41500	33300	25600	20300
c_L	.8					
c_{D_∞}	.016	.0165	.0185	.0215	.0265	.044
L/D_∞	50	48.5	43	37	30	18

These results indicate that reasonably high lift to drag ratios are feasible at quite low Reynolds numbers by artificially forcing rear turbulent reattachment.

4. <u>Preliminary Studies of Gas Turbine Power Plants Using Combinations of Closed and Open Cycles</u>

These studies have progressed to the point of determining the effect of the number of intercoolers, regenerators, reheaters, pressure ratios, and burner temperatures for the basic systems considered.

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NORTHROP AIRCRAFT, INC.

HAWTHORNE, CALIFORNIA

NAI-55-946

REPORT NO. BLC-82

NOTE ON THE TURBULENCE LEVEL OF THE NORTHROP WIND TUNNEL

October 1955

PREPARED BY:

O. E. Sipe, Jr.

O. E. Sipe, Jr.

APPROVED BY

W. Pfenninger

W. Pfenninger

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Drag and transition investigations of a 96-inch modified ellipsoid in the 12-Ft. Ames Low Turbulence Wind Tunnel and the Northrop Wind Tunnel showed low transition Reynolds numbers nearly equal in value. This suggested that the Northrop Wind Tunnel had a relatively low turbulence level, thus permitting future tests of laminar suction bodies of revolution and low drag suction wings at moderately high Reynolds numbers.

Transition experiments were conducted in the Northrop Wind Tunnel on a 96-inch chord flat plate to investigate the turbulence level, and also to better understand the reason for the low transition Reynolds numbers observed on the 96-inch body. Waviness of the plate was less than .0005 inch. The leading edge was contoured as an ellipse of high fineness ratio in an attempt to minimize peaks in the pressure distribution at the leading edge. Provision was made for small variations of angle of attack.

The pressure distribution over the plate (Figure 1) was measured by means of static tubes. The transition point was determined by means of a stethoscope attached to a total head tube. The transition point was defined as the occurrence of one turbulent burst per second (Figures 2 and 3). A check on the location of the transition point was made by measuring the surface total pressure and observing the minimum reading, usually defined as the beginning of transition (Fig. 4).

The most suitable pressure distribution was obtained for a negative angle of attack of 0.4°, with only a slight pressure peak near the leading edge and a slightly favorable pressure gradient along the plate. A maximum transition Reynolds number of 3.75×10^{6} was measured at a plate length Reynolds number of 4.5×10^6 .

Increased angle of attack ($\alpha = -1^\circ$) increased the favorable pressure gradient and raised the value of the maximum transition Reynolds number to 5.19×10^6 .

The value of 3.75×10^6 transition Reynolds number for practically zero pressure gradient compares reasonably well with the transition Reynolds number of 4.5×10^6 observed on a flat plate with a circular arc leading edge in the 12-ft low turbulence tunnel at the NACA Ames Laboratory. Transition experiments in the small R.A.E. low turbulence tunnel gave a somewhat higher value of 6×10^6 transition Reynolds number on a flat plate.

The relatively low turbulence level of the NAI Wind Tunnel will permit many preliminary laminar suction experiments to be conducted in this tunnel, for example, swept laminar suction wings, laminar suction bodies and wing root junctures.

Further transition experiments in the Northrop tunnel are being prepared on a cylinder of constant diameter with zero pressure gradient so that a comparison with a flat plate may be obtained. The results of these experiments will help provide an explanation of the flow over an ellipsoid.

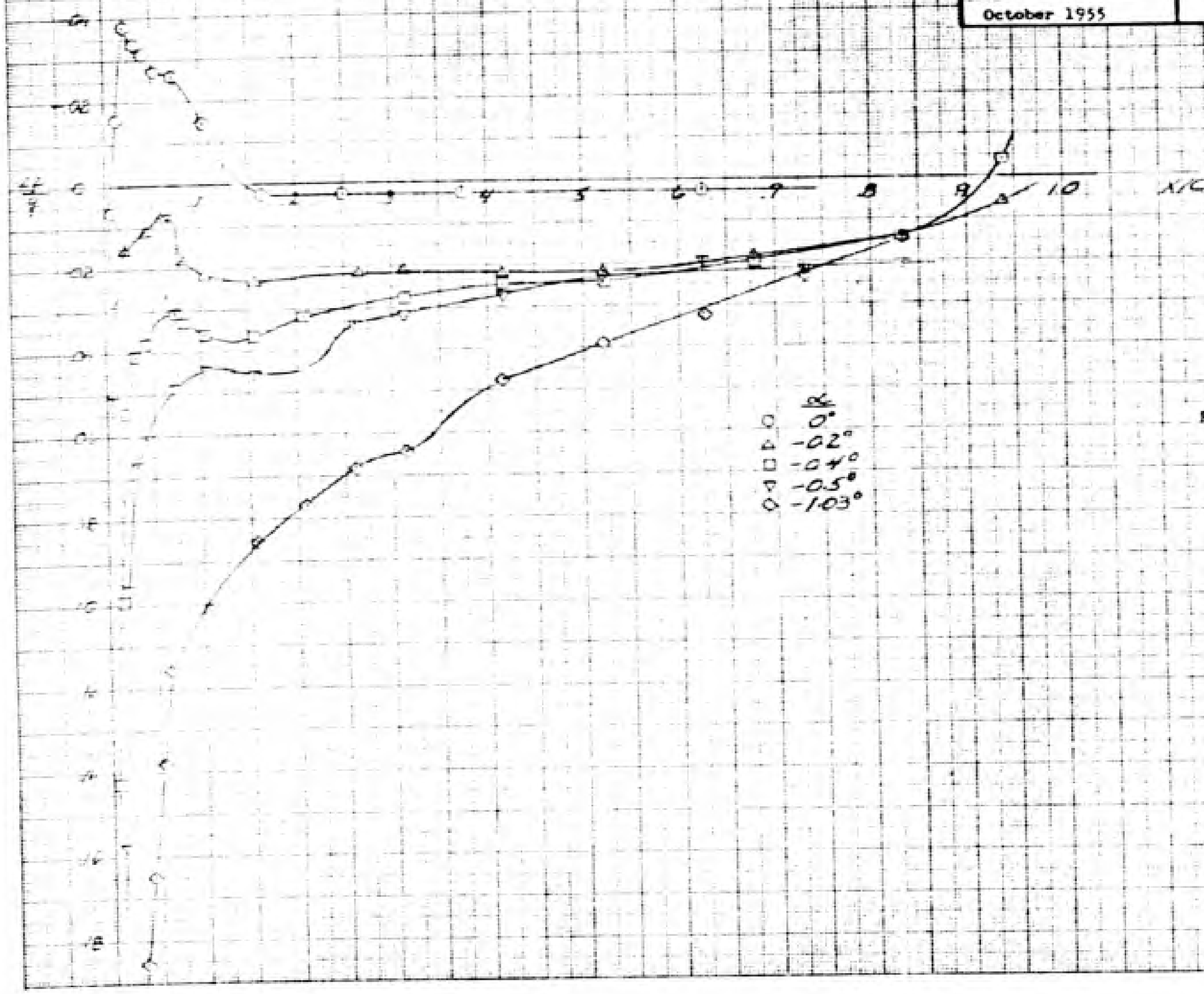

FIGURE 1

PRESSURE DISTRIBUTION OVER 96" FLAT PLATE

(VARIABLE ANGLE OF ATTACK, α)

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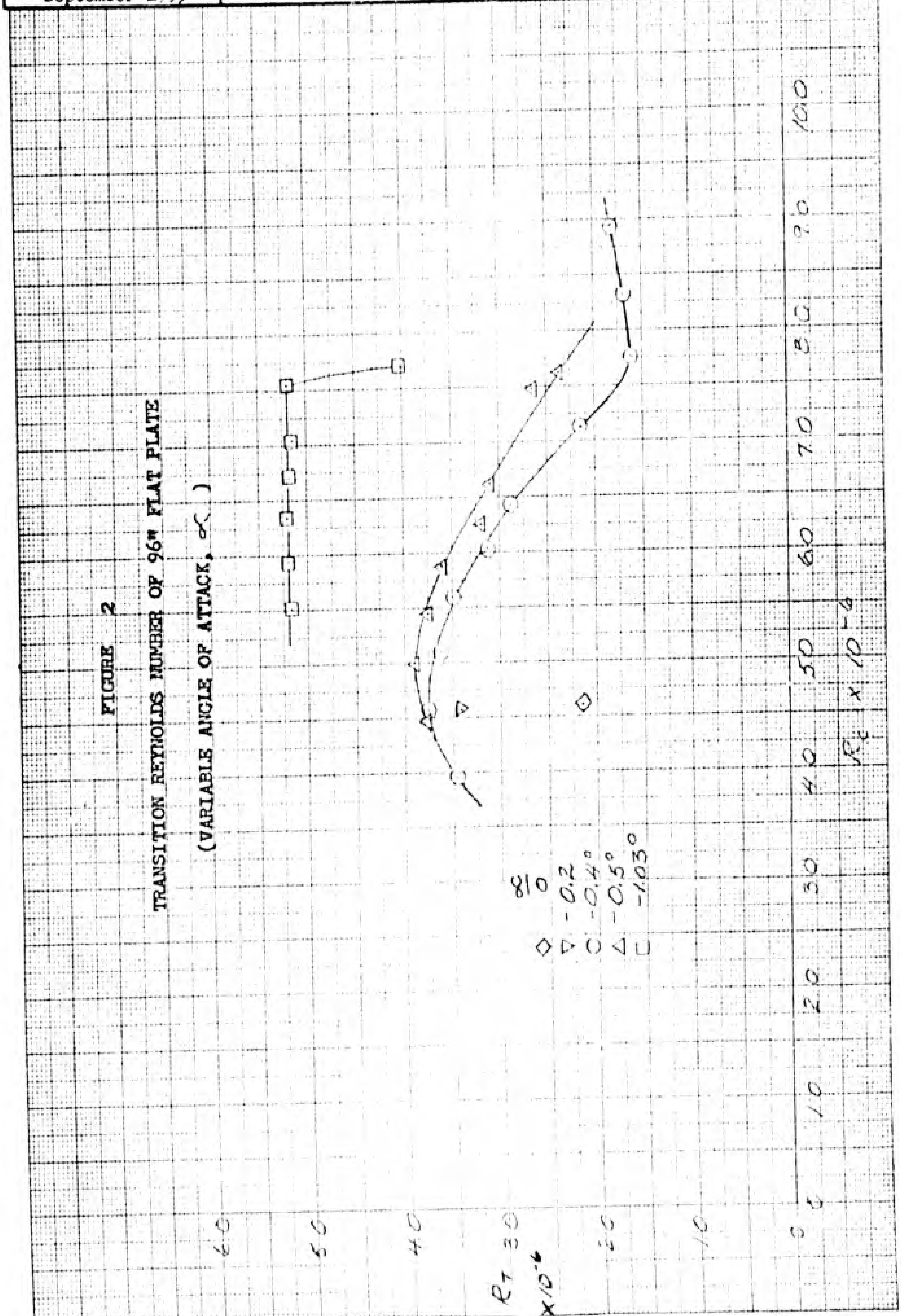

FIGURE 2

TRANSITION REYNOLDS NUMBER OF 96" FLAT PLATE

(VARIABLE ANGLE OF ATTACK, α)

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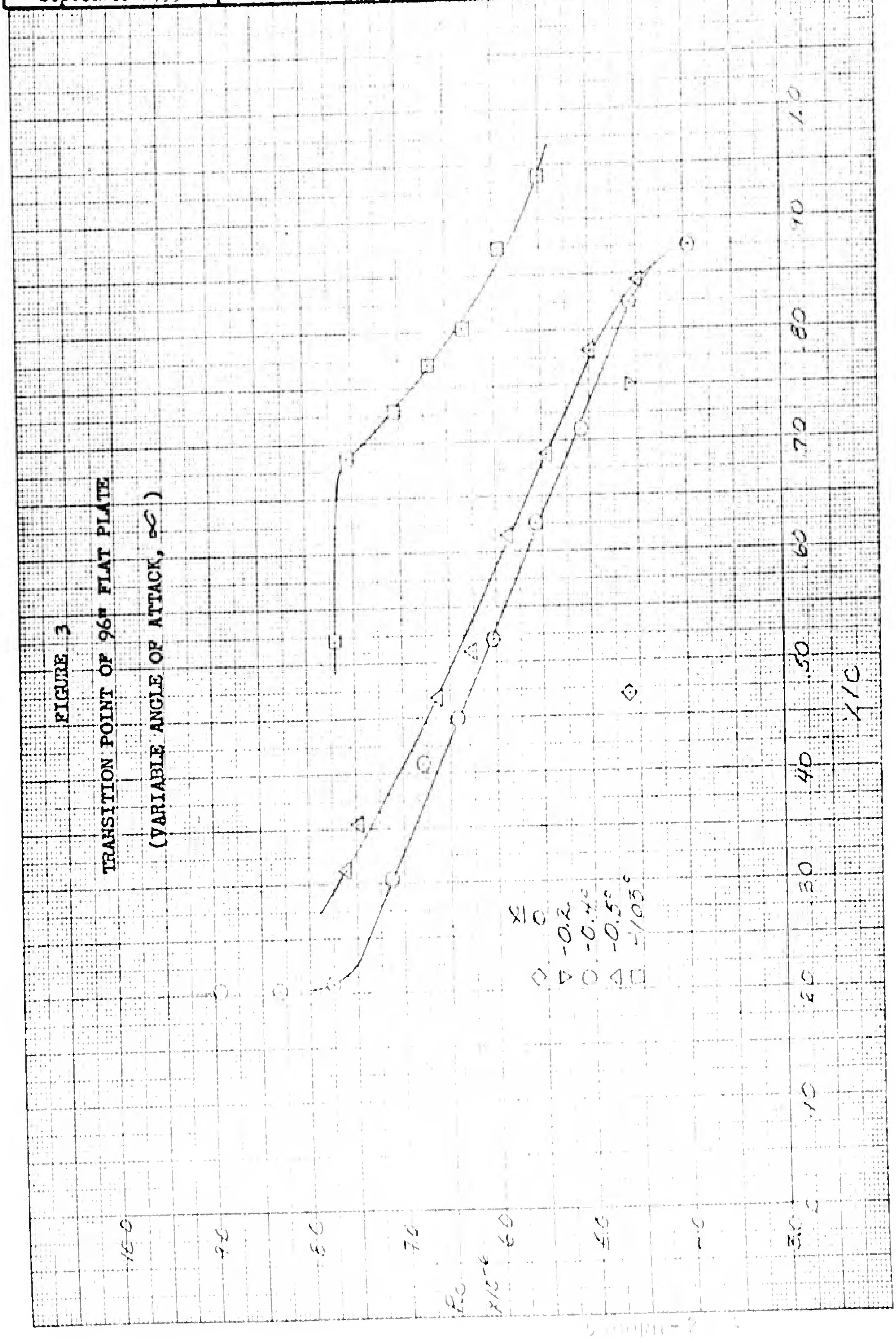

FIGURE 3

TRANSITION POINT OF 96" FLAT PLATE

(VARIABLE ANGLE OF ATTACK, α)

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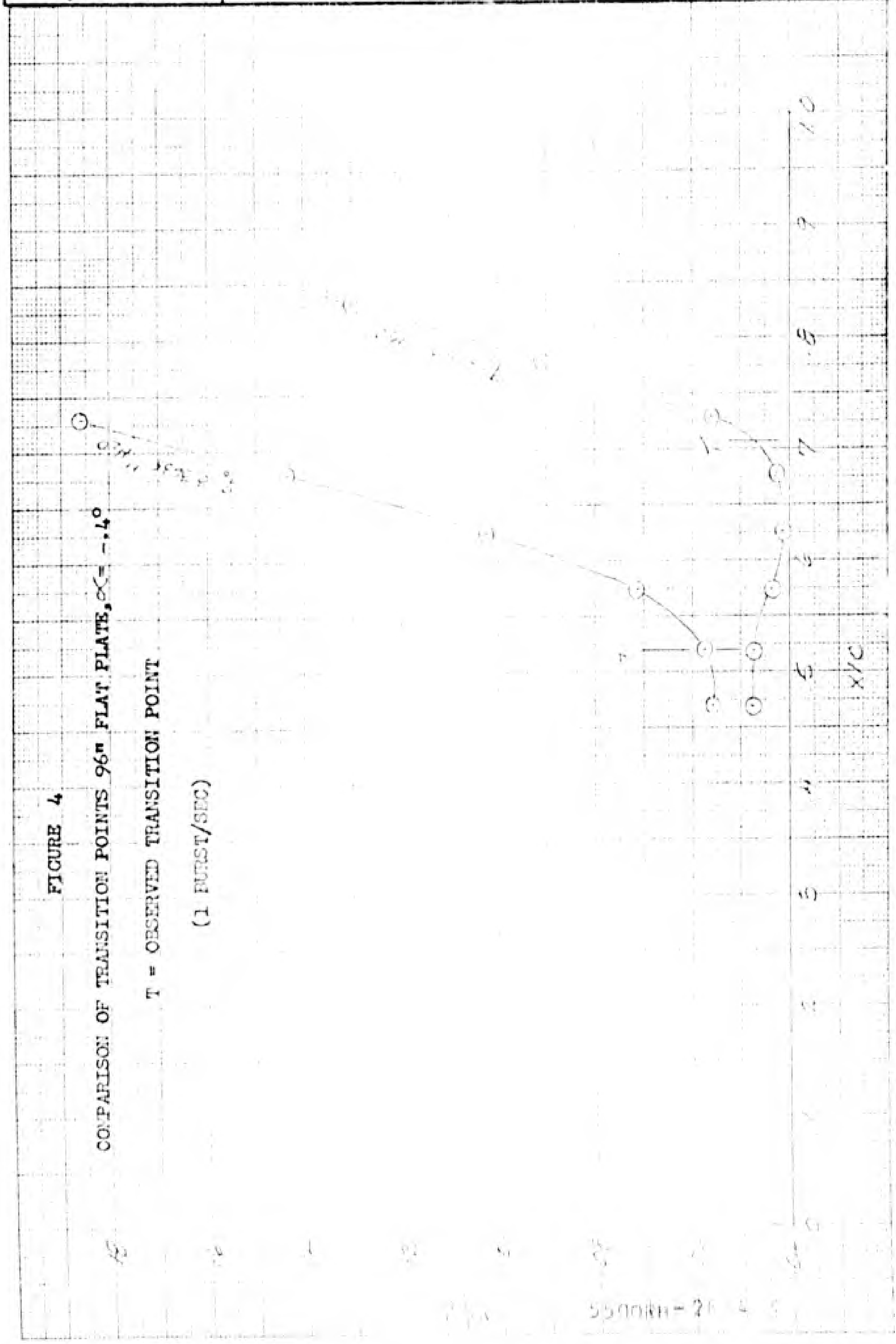

FIGURE 4

COMPARISON OF TRANSITION POINTS 96" FLAT PLATE, $\alpha = -.4^{\circ}$

T = OBSERVED TRANSITION POINT

(1 BURST/SEC)

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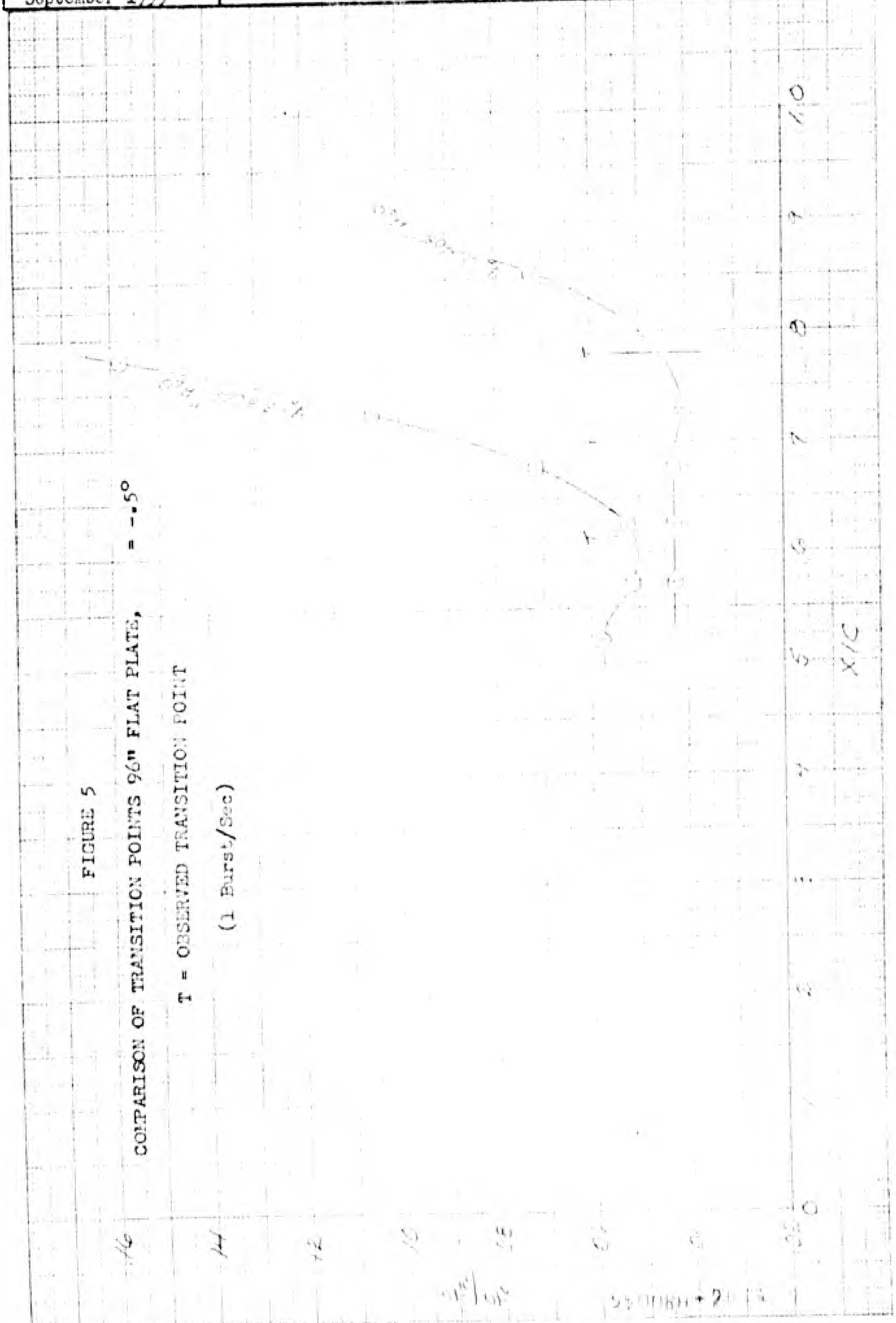

FIGURE 5

COMPARISON OF TRANSITION POINTS 96" FLAT PLATE, = -.5°

T = OBSERVED TRANSITION POINT

(1 Burst/Sec)

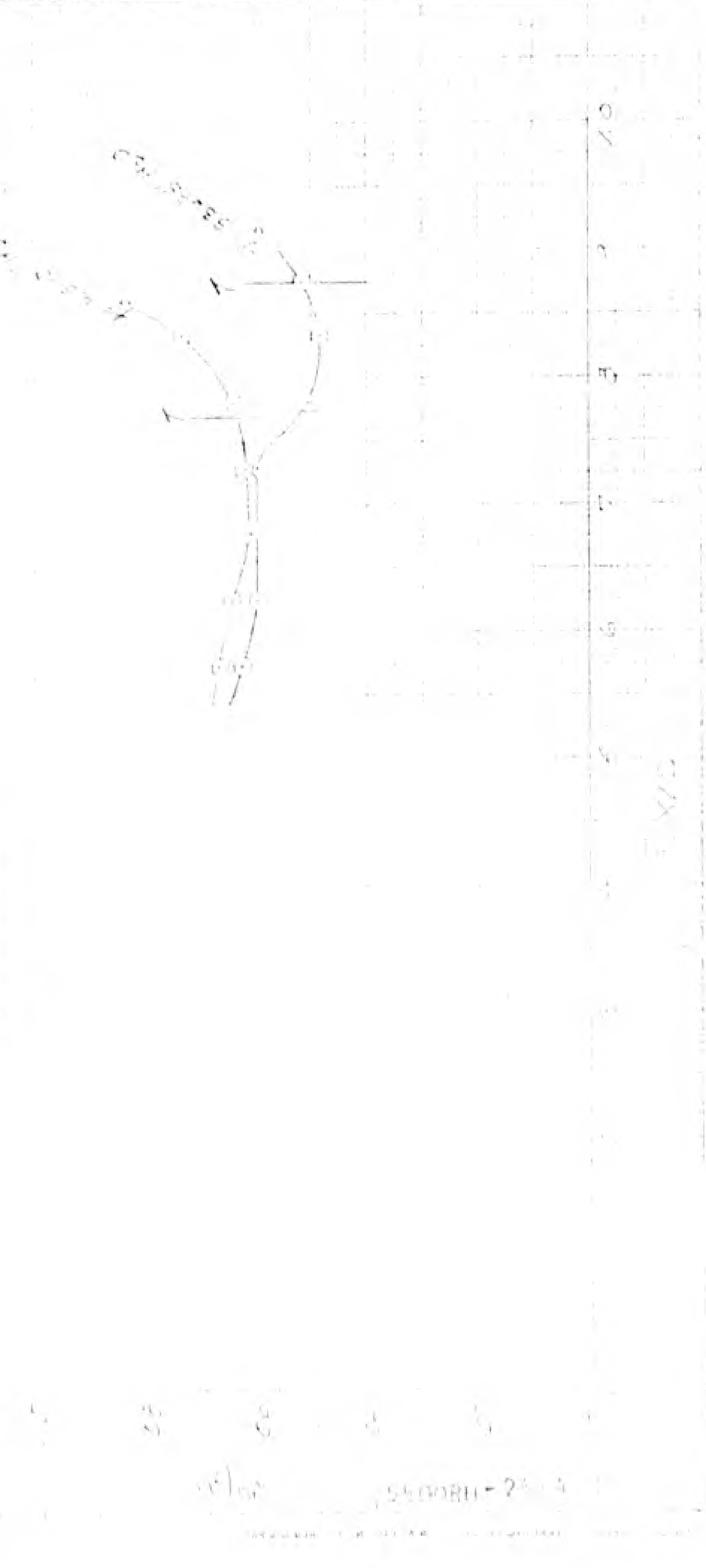

FIGURE 6

COMPARISON OF TRANSITION POINTS 96" FLAT PLATE, $\alpha = -1.03^\circ$

T = OBSERVED TRANSITION POINT

(1 Burst/Sec)

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NORTHROP AIRCRAFT, INC.

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STRUCTURAL DESIGN CONSIDERATIONS FOR LOW DRAG BOUNDARY LAYER CONTROL

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I. INTRODUCTION

Very low profile drags have been obtained by keeping 100% laminar flow at high Reynolds numbers by means of boundary layer suction in wind tunnels and in flight (References 12 and 13).

These drag reductions have profound effects on range, altitude, payload, fuel consumption, power plant requirements and other performance parameters. However, obtaining the maximum benefit from laminar flow requires the complete integration of the aerodynamic, propulsive and structural systems.

The structural designer is faced with a most exacting task. The purpose of this paper is to familiarize him with the principles that are unique to the design of a wing incorporating low drag boundary layer control. Many of the requirements of such a wing are so unusual that some general discussion of methods and reasons should be greatly expeditious.

What are believed to be the most important deviations from conventional design considered here are:

(1) Control of the physical condition of the outer skin surface

(2) Control of the distortions of the airfoil section

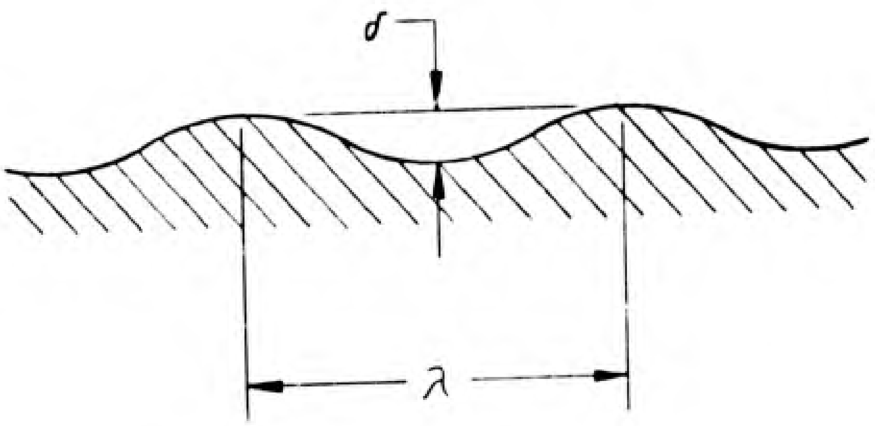
(3) Incorporation of internal ducting with a minimum weight penalty

(4) Aerodynamically and structurally efficient methods for conducting air through the outer skin surfaces of the aircraft.

II. SURFACE CONTROL

A. Surface Waviness

The definition of waviness as used in this report is given by (Figure 1 and equation 1)

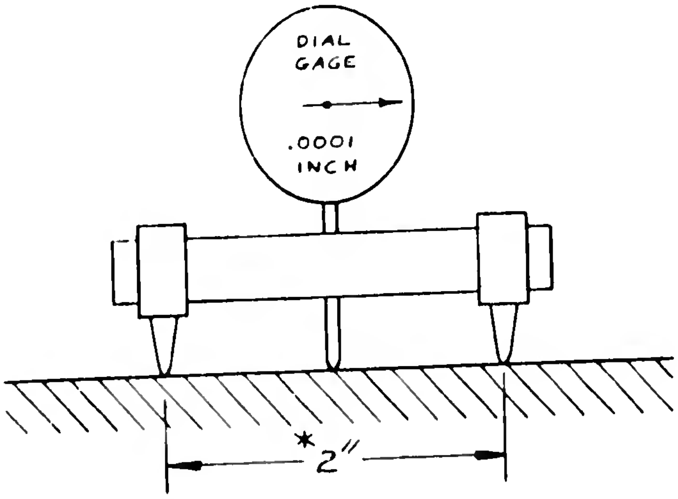


$$\text{WAVINESS} = \delta/\lambda \tag{1}$$

FIGURE 1

The surface waviness requirement is established by the aerodynamicist and this requirement has a strong influence on the structural design, particularly at high Reynolds numbers where the maximum allowable waviness for the wing is on the order of one mil per inch or less. This represents total allowable waviness and includes both manufacturing or built-in waviness and deflection-induced waviness.

The actual measurement of waviness is an operation requiring a high degree of precision. For determination of short length waves, the instrument in Figure 2 has been found satisfactory while for those of greater length, comparison with full size templates is acceptable.



* F-94 EXPERIMENT REF.13

FIGURE 2

By the term "manufacturing waviness" is meant that waviness resulting from all processes leading up to the completed surface. Machining, forming, riveting, and handling; all leave their effects on the completed surface. As received from the mill, incidentally, the surface is quite flat, having a waviness, generally, of less than 1/10 mil per inch.

To determine what might constitute a high but reasonable standard of waviness attainable by the ordinary airframe manufacturer, a survey was made of several high performance aircraft on which special care had been exercised in producing a smooth wing surface.

The Douglas X-3 wing, a polished surface, had a maximum waviness of one mil per inch except in areas near cutouts and control surfaces. This maximum occurred at only one location, the next highest value being four-tenths of a mil per inch and the remainder of the surface less than this value.

The Douglas 558 Phase II wing was a filled and painted surface and had a thinner gage skin than the X-3. Waviness values varied from seven to nine-tenths of a mil per inch. No special precautions had been taken in obtaining smoothness other than filling where indicated by visual inspection of templates placed on the wing.

The Bell X-1 maximum waviness was 1.25 mils per inch although this may be misleading since the wing had a fairly rough paint on it and was not in a flying condition at the time it was measured.

Northrop N-69 and F-89 wings were as smooth in most portions as the X-1 and D-558 wings.

As might be expected, spanwise joints usually showed increased waviness using standard production methods. Of course, it would be most desirable to eliminate all spanwise splices and recent production methods indicate that this may be accomplished.

Methods of forming the contour of the wing skin were studied with the result that no particular method showed a decisive smoothness advantage. The methods considered were braking, rolling, spring quenching, shot-peening and draping, of which the draping was the best. However, the draping of thick skin over the usual wing contour would result in excessive built-in stresses and probably objectionable effects on wing camber unless the upper and lower skin gages are nearly the same. Since, in general, this is the case for highly stressed wings, it may be possible to compensate for the small amount of unbalance by means of appropriate rib spacing. (See Reference 6.)

A convenient rule of thumb to determine built in stress is

bending stress in the skin due to draping =

$$f = \frac{4\Delta Et}{L^2} \qquad (2)$$

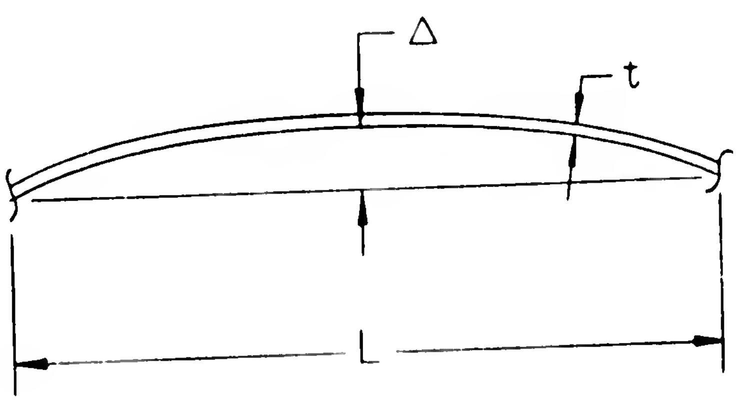


FIGURE 3

Consideration of the results of this survey might allow the optimistic assumption that with some extra care in fabrication and possibly some hand finishing, surface waviness could be held to .5 mils per inch maximum (Ref. 6).

Also assuming, and conservatively this time, that load-induced waviness adds directly to built-in waviness, the designer is left with the arithmetical difference between the total allowable waviness and .5 mils per inch as his maximum allowable.

Since turbulent flow with its accompanying drag rise will occur with an increase in waviness beyond the maximum allowable or beyond a certain angle of attack, it is obviously pointless to design for laminar flow smoothness requirements above a fairly low load factor. Under these conditions (for example, 1.5 g load factor) the stress level is quite low and the task of holding the load-induced waviness with the required values is less formidable than it would at first appear.

The major waviness or deflection-inducing loads are:

(1) Primary wing loads

(2) Crushing load due to flexure

(3) External and internal air loads

(4) Magnification of deflections by bending stresses

(5) Thermal stresses

1. Primary Wing Loads

The entire outside surface must be at less than its critical stress within the required load factor range and, in addition, must not retain permanent waviness after return from limit loads. This is not difficult in heavy skin areas and, in fact, is almost automatic, but in those areas where other requirements allow thinner gages, the skins or shear webs, etc., must be stabilized, for example, by means of sandwich construction or similar methods.

Compression tests on panels stabilized with closely-spaced webs, Refs. 9, 10, show that waviness of the skin under load varies directly as some function of the rigidity of the stabilizing members and of the load intensity in the skin. Permanent waviness caused by loading to limit loads does not generally exceed one-half mil per inch, and a fortunate circumstance noted in test is that permanent load-induced waviness does not increase with repeated cycles to limit load.

This, of course, means that the process of maintaining a smooth surface on the wing will not be an endless hand-finishing process but one which, once completed, will require only careful handling, routine cleaning, and possibly, occasional repairs of the inevitable nicks and gouges.

Internal members such as spars and ribs must also be non-buckling to limit load since they are responsible for the support and smoothness of the outer surface. Spar webs in a multi-spar wing, for instance, have quite low shears and since only a very thin gage is required for strength, some stabilization is required. Methods of accomplishing this include closely-spaced vertical stiffeners, horizontal stiffeners, a combination of the two, or double skin construction, such as honeycomb or chemical-milled sheet. Figure 4, illustrating these methods, is for a hypothetical wing and is merely for the purpose of aiding visualization.

The use of vertical stiffeners would be restricted to the spar webs in the portion of the wing not used as ducting since the restriction afforded by the outstanding leg of the stiffener would create excessive duct losses.

Horizontal or spanwise stiffeners alone, on the other hand, afford an aerodynamically acceptable web but are quite inefficient in stabilizing against crushing loads.

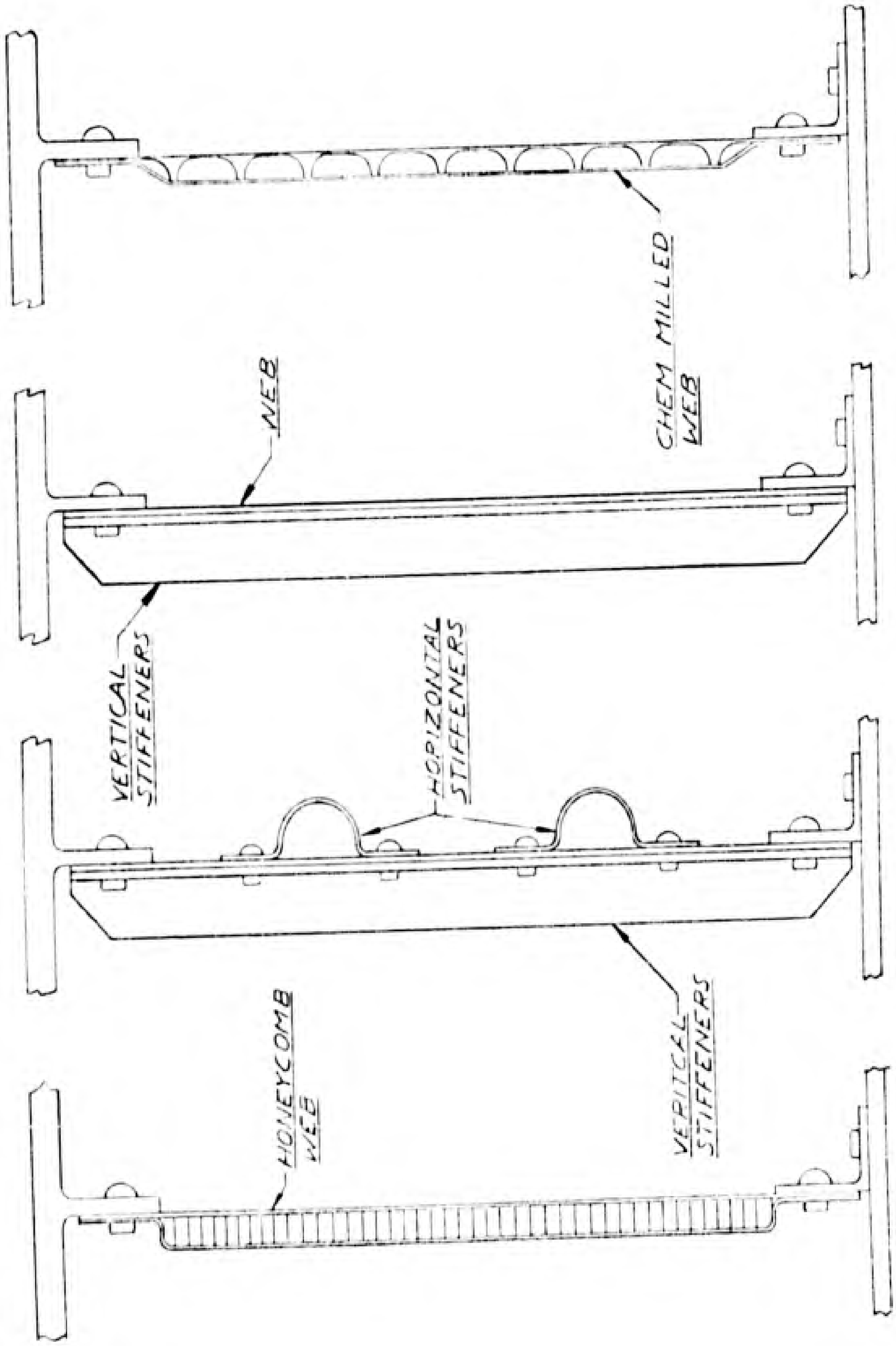


FIGURE 4

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It is apparent then, that in the ducted portions of the wing the sandwich webs are ideal from structural and aerodynamic considerations, and, as established by quite extensive investigations, from the weight standpoint as well.

2. Crushing Loads

A general expression for crushing load normal to the skin in terms of running load and radius of curvature of the wing due to bending is

$$p = \frac{N}{R} \tag{3}$$

where
- p = crushing load in psi
- N = bending load per inch of structural chord in lbs/in.
- R = radius of curvature of the elastic axis of the wing in inches

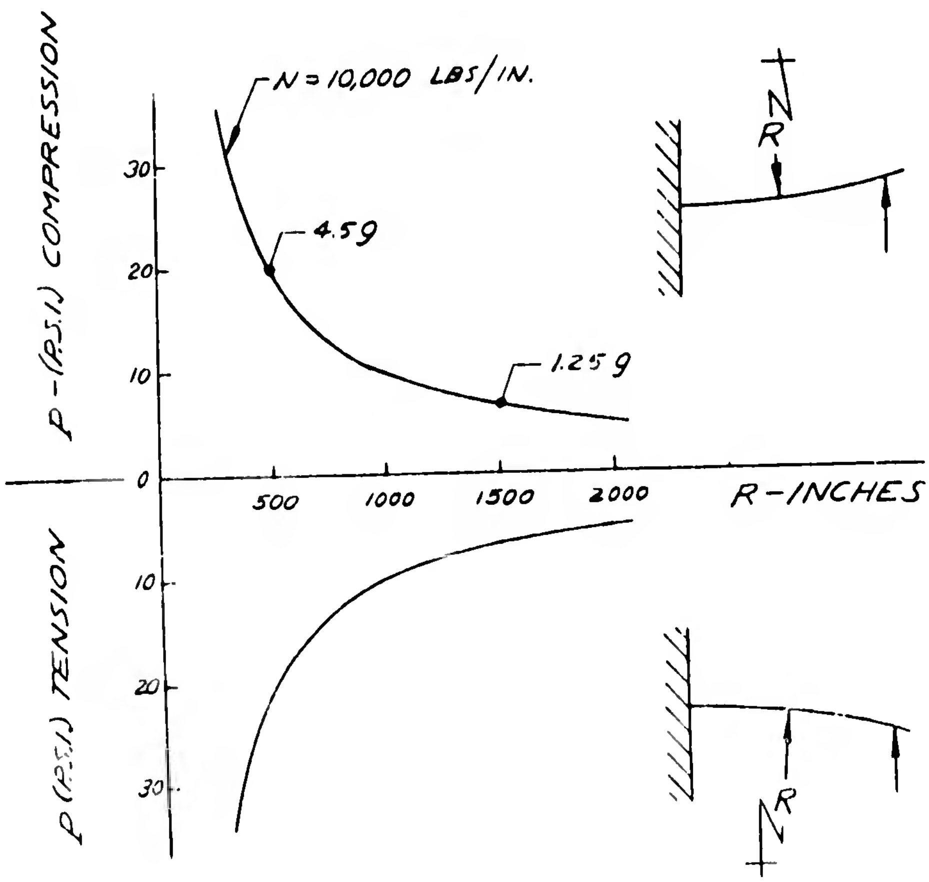

FIGURE 5

When a curvature is not built into the wing, a convenient expression for crushing load is

$$p = \frac{\sigma^2 t}{Ey} \qquad (4)$$

where

p is crushing load normal to the skin in lbs/in^2 of surface

σ = flexure stress in skin from normal wing or fuselage bending in psi

t = skin thickness in inches

E = Modulus of Elasticity

y = distance from the neutral axis to skin in inches.

This represents the crushing load of the skin only and does not include that due to spar caps or any other longitudinal members. The flexural stress, σ that interests from the waviness-standpoint is that existing within the load factor range where transition is to be avoided, and since this σ term is squared, it is obviously important to set these load factors as low as is consistent with the purpose of the aircraft.

For a multi-spar wing, the crushing load would be carried between spars by the skin in bending about its own minor axis as in Figure 6.

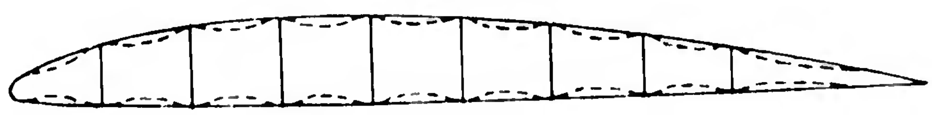

FIGURE 6

Figure 7 is shown as an example of the skin thickness required to resist crushing loads, with a given spar spacing, using the conventional deflection formula, $\delta = \frac{w L^4}{384 EI}$. This would apply to interior bays where the skin is continuous in the chordwise direction

δ = deflection at point midway between spars

w = crushing pressure plus interior pressure

L = distance between spars

I = moment of inertia = $\frac{t^3}{12}$ unit width

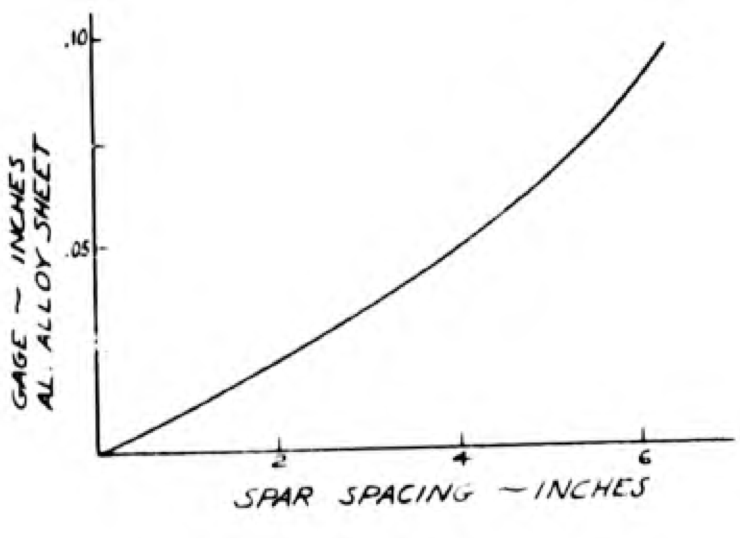

FIGURE 7

If a built-in wing droop is used to compensate for wing deflection, the resultant crushing load is, of course, only that caused by the actual curvature.

3. Air Loads

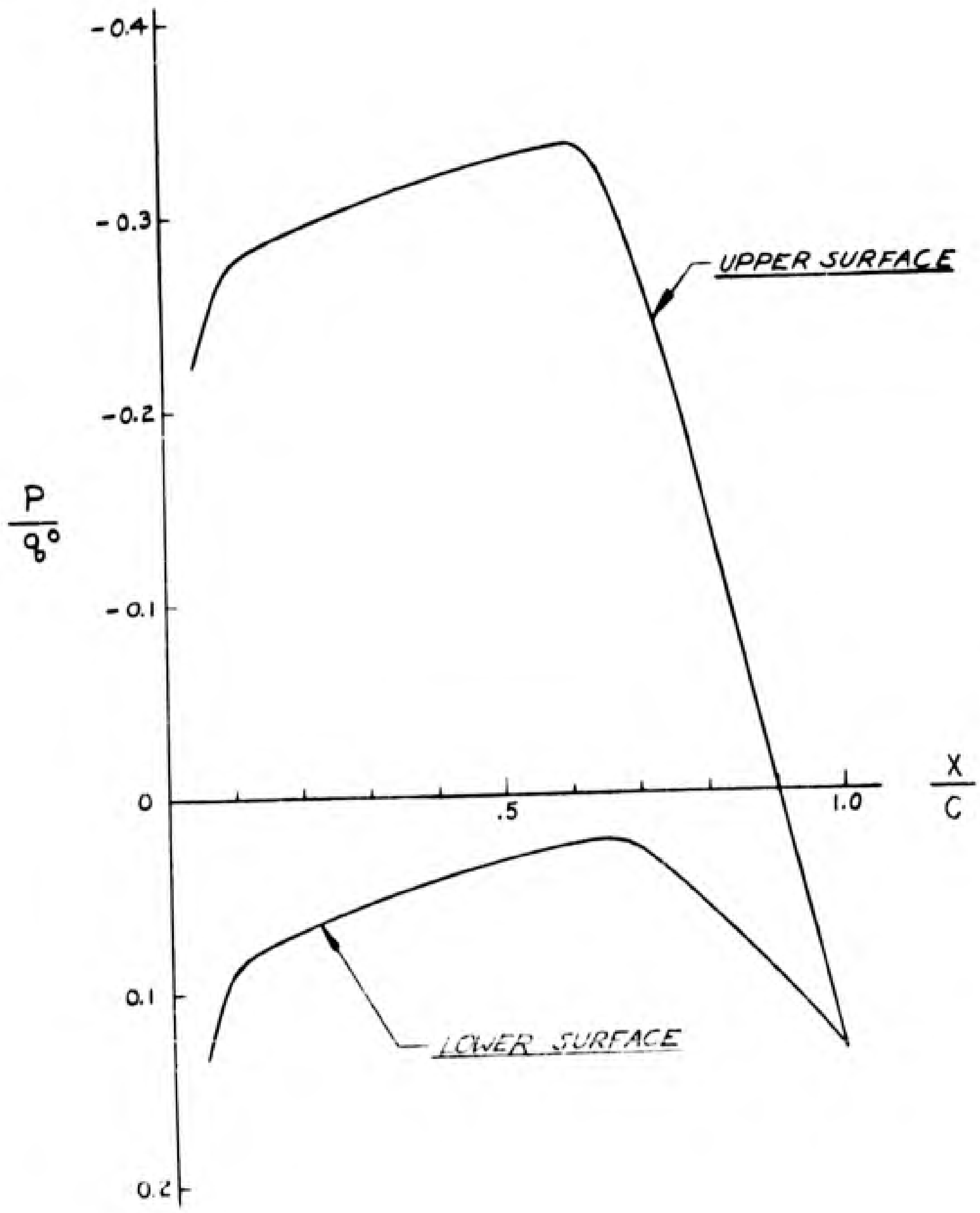

FIGURE 8

External air loads are generally subtractive from the crushing loads over a large portion of the wing upper surface. Figure 8 is a typical thin wing pressure distribution. Pressures inside the wing in the suction ducts, however, act in the same direction as crushing loads, i.e., inward, and must be added thereto.

4. Magnification of Deflections

The effect on smoothness of an axial load applied to a plate initially bent to a curvature, such as wing curvature, has been treated in Ref. 1. However, as in the case of a column where the applied load is much less than the critical load, this effect is small at the low loads existing at 1.25 to 1.50 g and ordinarily may be neglected.

5. Thermal Stresses

Stresses produced by a temperature gradient could certainly produce surface waviness and this should be considered in those areas having such a gradient. These effects might become important at supersonic speeds, but probably not up to $M = 1$.

B. Surface Roughness

Roughness elements consist of inherent and acquired surface blemishes such as those due to fabrication, handling, and acquired surface blemishes such as those due to fabrication, handling, and fly and dust accretion.

An estimate of the allowable surface roughness in flight at high Reynolds numbers can be made from the F94 flight low drag suction experiments (Ref. 13). Fluorene particles were sprayed on the glove surface to observe the state of the boundary layer. At $U_0/\nu = 3.2 \times 10^6$/ft unit Reynolds number for flight number 68, the average fluorene particle size was .0003 to .0004 inch, with 100% laminar flow, and the maximum particle size was 0.001 inch to 0.0015 inch. The glove chord Reynolds number was 24×10^6 .

Recent roughness experiments by Von Doenhoff showed critical Reynolds numbers of 300 based on the height of the roughness and the local velocity in the boundary layer at the top of the roughness element.

Assuming a critical surface roughness of 0.001 inch at $U_o/\gamma =$ 3.2×10^6/ft, corresponding to the F94 glove at 24×10^6 , the critical surface roughness for a high altitude laminar suction airplane, cruising at $M = 0.9$ would be as follows:*

H = 55,000 ft	Maximum permissible height of roughness	.0027 in.
60,000 ft		.0033 in.
65,000 ft		.0042 in.

C. Outflow

Leakage can cause transition and must therefore be prevented by sealing of all joints and rivet and bolt heads. Bonded structure works well in this respect and, of course, integrally stiffened skin is ideal.

III. CONTROL OF DISTORTIONS OF THE AIRFOIL SECTION

Distortion of the airfoil section as a whole from the theoretical section results in a change in the pressure distribution possibly requiring a readjustment

* The permissible roughness is inversely proportional to the unit Reynolds number U/γ , under otherwise the same condition.

of the suction quantities. Provisions may be made to make a reasonable amount of this readjustment in flight.

Another approach might be to build the airfoil with the proper amount of counter distortion so that at a 1 g flight condition the airfoil returns to the original theoretical shape.

Camber changes result from:

(A) Chordwise bending

(B) Anticlastic bending

(C) Camber straightening

A. Chordwise Bending

Since the contribution to wing bending stiffness of the aft section of the airfoil is a smaller percentage than the percentage of total airload it carries, a certain portion of airload must be carried chordwise, rather than spanwise, and the effect is as though the aft section were cantilevered from the forward section resulting in a vertical deflection of the trailing edge with respect to the structural box.

This is true also of the leading edge but the resulting chordwise bending is less severe because of the shorter overhang and thicker section.

B. Anticlastic Bending

in Reference 8, Fung and Wittrick show that when the spanwise curvature is small, the curvature produced in the chordwise direction in a tapered plate is the same as simple bending theory gives for a flat plate

$$\frac{d^2\zeta}{dy^2} = -\frac{\mu}{R} \tag{5}$$

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where $\frac{d^2\zeta}{dy^2}$ = chordwise curvature

μ = Poisson's ratio

R = longitudinal curvature

The longitudinal curvature is considered small when it satisfies the inequality:

$$\frac{b^2}{Rt_o} \ll 1$$

where b = semiwidth of plate (1/2 spar web spacing)

t_o = average thickness

However, inspection of the relationship

$$R = \frac{Ec}{\sigma}$$

where E = Young's modulus

c = distance from extreme fiber to neutral axis

σ = bending stress

shows that the longitudinal curvature may not be considered small for the usual range of wing working stress.

Actual calculations of the chordwise deflections have not been made for an airfoil section; however, as an example, from data in Ref. 8 it may be shown that, for a solid wedge-shaped airfoil section of 1.5% camber having a higher bending curvature than an airfoil would normally be expected to have at 1 g , the change of camber would be 0.24% .

It would be reasonable to assume that the change in pressure distribution occasioned by a camber change of this order could be adjusted in flight.

C. Camber Straightening

A thin cambered airfoil may be structurally unstable in bending exactly as a flexible steel tape measure having an initial lateral curvature. As the wing or tape is bent, the cross-section progressively flattens and its moment of inertia therefore decreases as the bending moment increases.

An approach to this problem is suggested by Ref. 11 from which Figure 9 is taken.

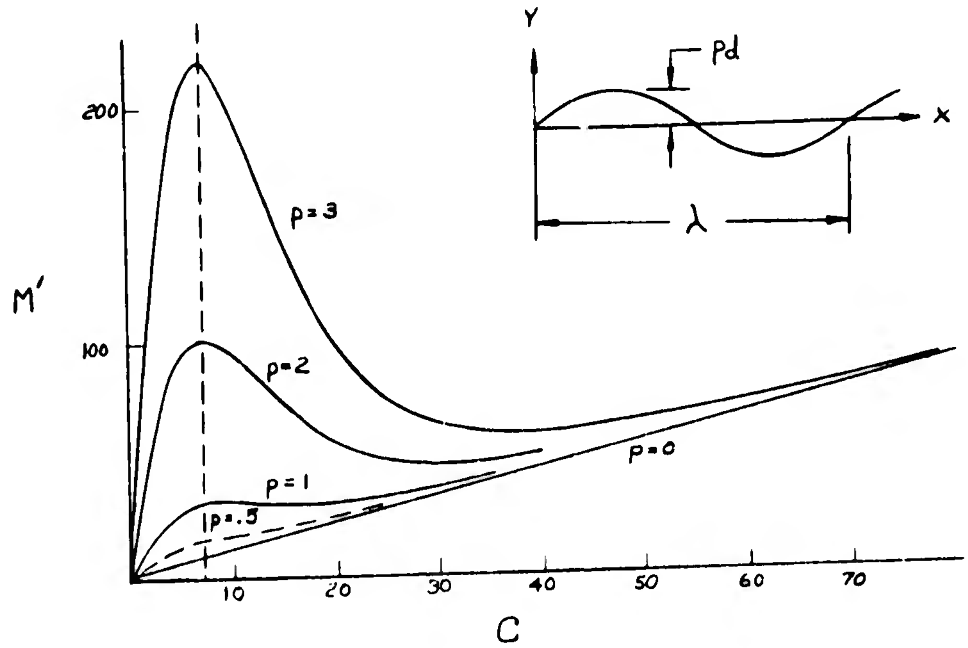

FIGURE 9

where

p = numerical coefficient, see figure.

d = thickness of plate - inches

R = radius of curvature - inches

$$C = \frac{\lambda^2}{Rd} \tag{6}$$

$$M' = \frac{12M\lambda}{Ed^4} \tag{7}$$

As an example, consider a cambered plate of 3% thickness and 1.5% camber as in Figure 10.

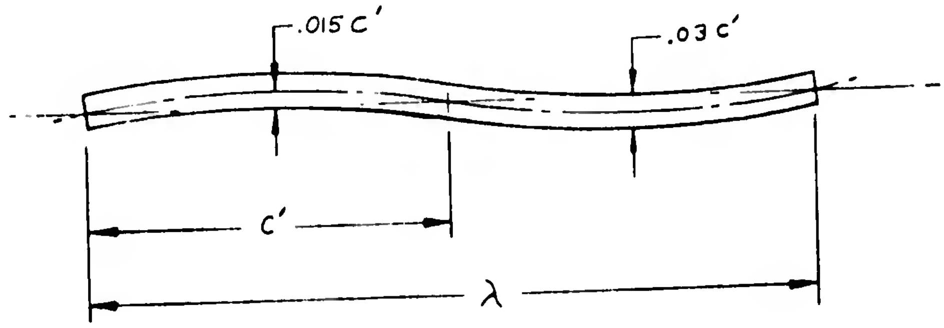

FIGURE 10

Assume $R = 5c'$ (an arbitrary choice)

then $$C = \lambda^2/Rd = \frac{4(c')^2}{5c' \times .03c'} = 26.6$$

This value of C puts the plate well past the critical value of 7.5 in Fig. 10 but it may be noted that for p = .5 the slope of the moment curve never becomes negative. In other words, a solid equivalent airfoil of the dimensions as in example, would not be subject to a snap-buckling type of instability. The correlation of this to a low density airfoil has yet to be evaluated.

1. Ailerons and Flaps

Laminar flow should be maintained on the control surface for small control surface deflections, say, in the order of $\pm 3^{\circ}$. This is possible if the usual waviness requirements are kept and if no large steps or gaps develop. Figure 11 shows the permissible magnitude of such steps for the maintenance of laminar flow.

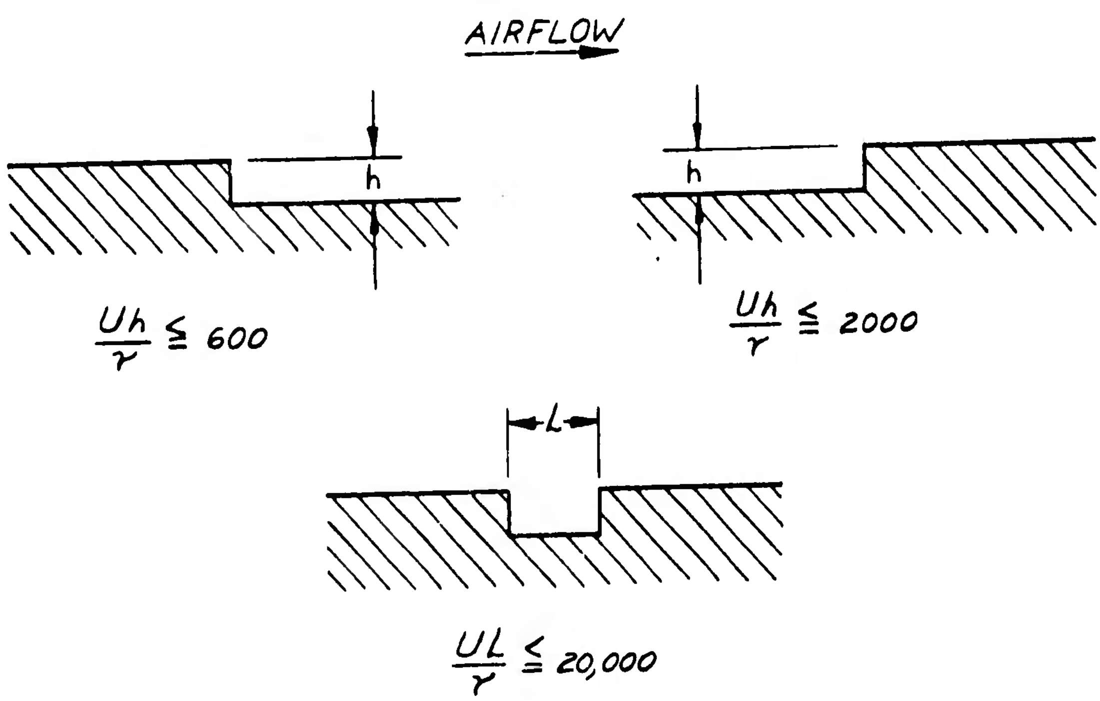

FIGURE 11

These requirements can probably be met best with a continuously-hinged surface or with closely spaced hinges. A complication develops from this design since the hinge line becomes curved as the wing is bent and the induced hinge loads and moments are of appreciable magnitude. (See References 4, 7, and 9).

Suction requirements in the control surface, if necessary at all, are met with methods similar to those used on the wing, although spanwise ducting of air is more restricted because of the smaller available cross-sectional area. Chordwise transfer of suction air out of the aileron can be accomplished through the aileron closing spar, which can be built, for example, as a trussed spar construction.

IV. INTERNAL DUCTING

At moderately high Reynolds numbers, low drag suction may be required only in regions of the adverse pressure in the rear part of the wing from 50% or 60% chord* to the trailing edge. Internal ducting will then be confined to the rear part of the wing (Ref. 5).

This means that the portion of the wing forward of approximately 50% chord would have no unique requirements other than that of smoothness and leakage requirements to distinguish it from a conventional wing. Other considerations such as high lift devices, de-icing, or ducts for other purposes would, of course, modify this conclusion.

Duct losses should be minimized in order to insure uniform suction along the span and to improve the performance of a laminar suction airplane. The design of suction ducting systems for low drag suction airplanes has been discussed in References 5 and 15. Generally, adequate duct cross-sectional area is available except in the rearmost part of a suction wing. The suction air for these rearmost parts can be ducted forward, for example, through chordwise ducts located close to the lower wing surface (Ref. 5), see Figure 12.

* On very thin wings or on swept wings, particularly at higher wing chord Reynolds numbers, additional relatively weak suction may be required in the forward part of the wing.

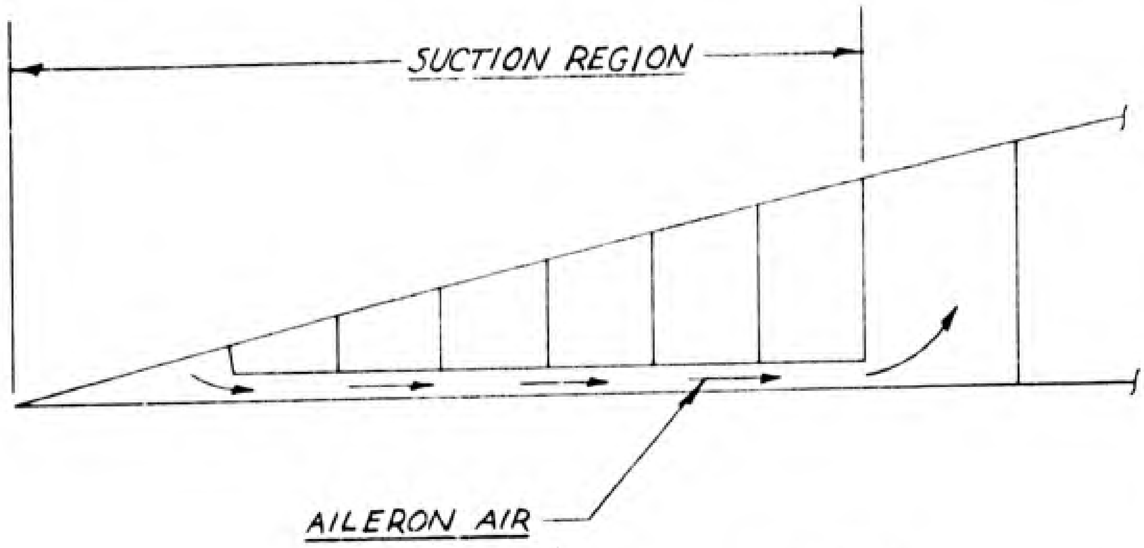

FIGURE 12

Smooth ducts are best obtained by the use of shear webs of types shown in Figures 4a and 4d without vertical stiffeners.

Chordwise stiffening (ribs with truss diagonals, Fig.15, or Vierendeel truss) of the rear section is required to maintain small chordwise deflections. The joints between the rib diagonals and the wing skin must be carefully designed and the truss diagonals must be streamlined in order to minimize blockage of the suction ducts. A Vierendeel truss-type structure (Figure 13) avoids this problem. In this case the wing skin forms the chord of the truss and the spar webs the vertical

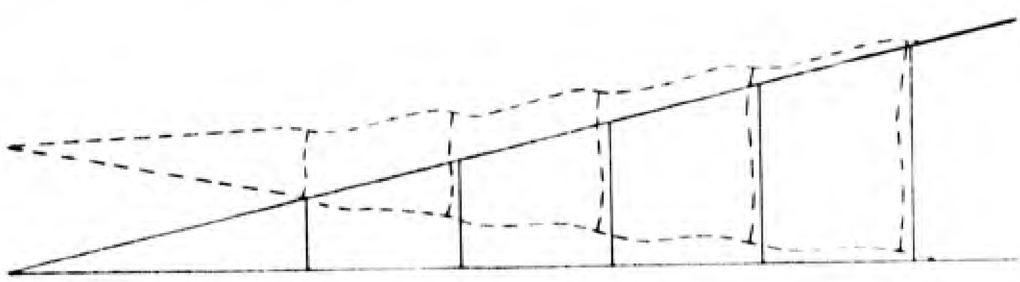

FIGURE 13

members. Calculated deflection of such a structure in a particular case was satisfactory with a nominal 70 pounds per square foot wing loading, producing a deflection at the trailing edge of .08 inch or about .07% chord. Values of joint rigidities were assumed at 100%.*

Several types of joints were tried for this structure. These are presented in Figure 14.

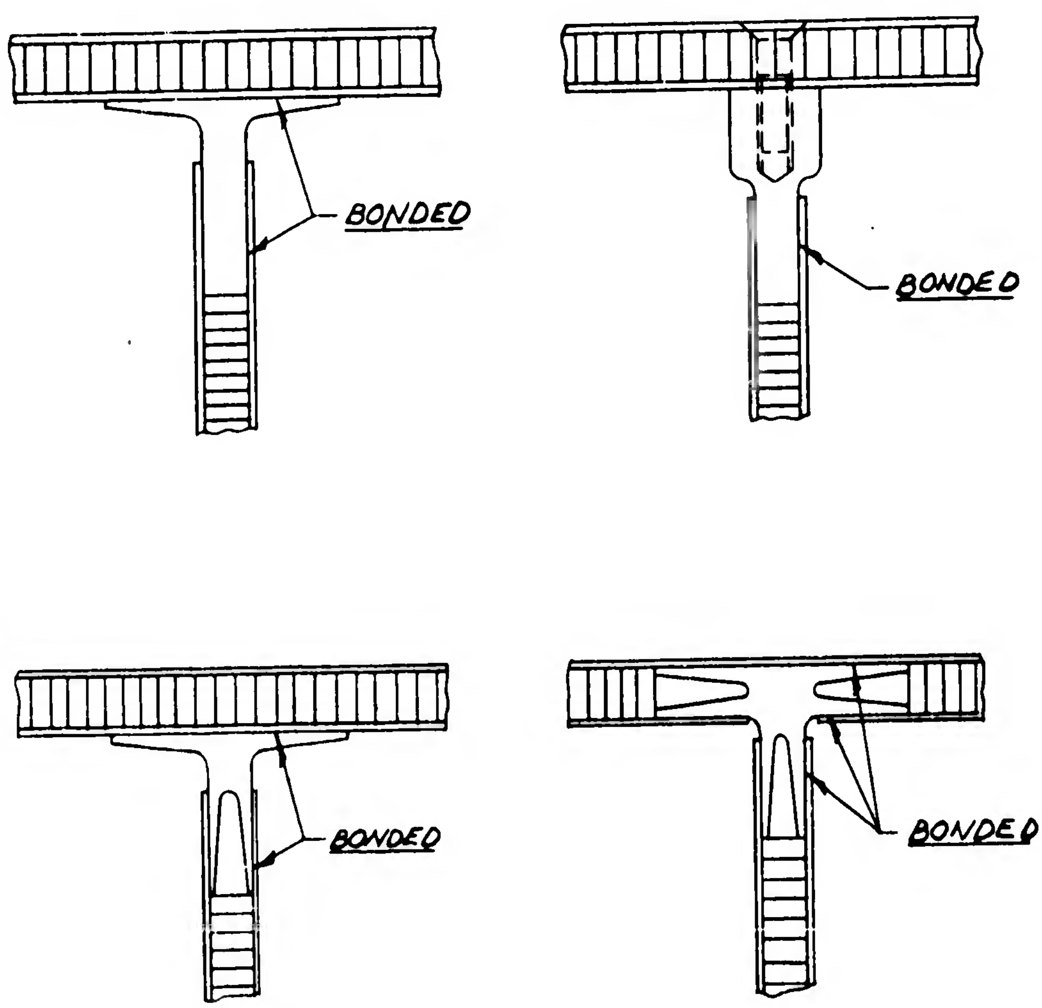

FIGURE 14

* Rigidity tests showed joints to be in the order of 80% rigid (conducted by J. E. Wieder; report pending also deals with creep).

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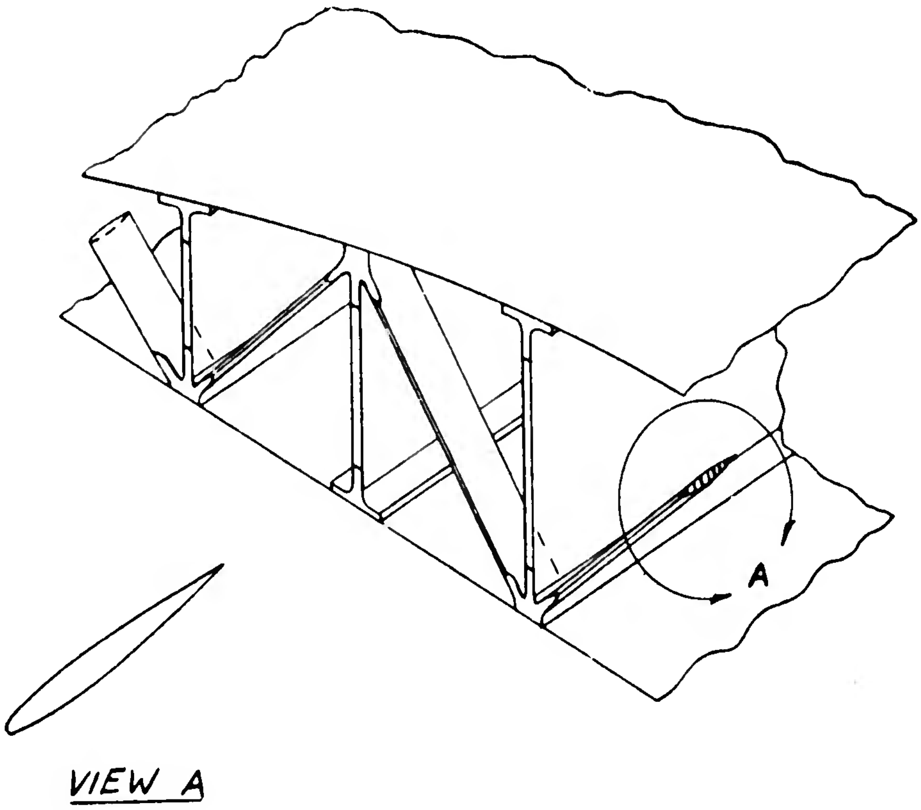

FIGURE 15

V. INFLOW

Continuous laminar suction has been successfully approached by means of suction through many fine slots or rows of closely spaced holes. Low drag suction through few slots, shaped as diffusers, is inferior to suction through many fine slots, from an aerodynamic, structural, and manufacturing standpoint. For this reason, low drag suction through a large number of fine slots or rows of holes is considered in this section.

Reference 5 presents several suggested designs for obtaining airflow through a load-carrying skin. Subsequent study and experiment have modified these designs and evolved others so that a complete, up-to-date summary will be given here. In addition, some of the more salient points in their fabrication will be stated for the designer's benefit. First, it is important to note that since the suction will generally be applied mostly in the rear portion of the wing, aft of the maximum thickness position, the structure in this area is working in bending at a reduced efficiency, being nearer the wing neutral axis, and therefore must be kept to a minimum weight.

The solid skin-type slotted skin of Figure 16 is attractive for wings of higher wing loading. Relatively few unknown problems are encountered with this type of construction.

The order of fabrication starts with machining and forming of the heavy skin, including drilling the holes. Then follows the application of the outer sheet by means of an adhesive and finally cutting the slots. Both weight and slotting considerations dictate a thin outer sheet and a very small amount of finishing or sanding will be permitted after assembly; therefore, the bonding

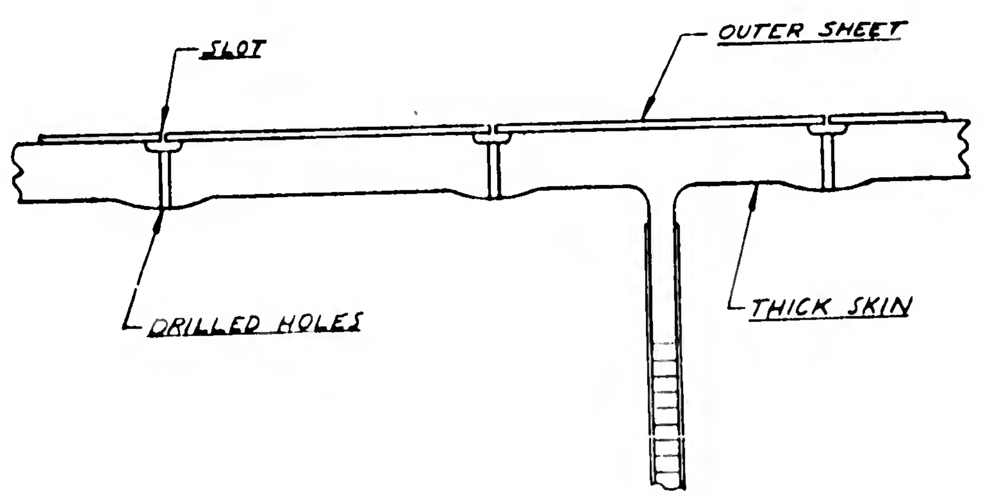

FIGURE 16

operation should be accomplished with the outside surface in contact with a negative die of approximately the same waviness that will be required of the final surface.

To allow the use of large panels and eliminate the need for frequent chordwise and spanwise joints, a portable slotting saw has been developed (Ref. 14). It is powered by a small drill motor and runs on a light track, temporarily affixed to the skin.

Cutting the slot is the last operation on the skin panel, with the exception of final hand finishing. Thus, the edges of the slots will be better protected against damage during fabrication.

Thickness of the assembled skin is dictated by waviness and stability criteria as discussed in the section on crushing loads.

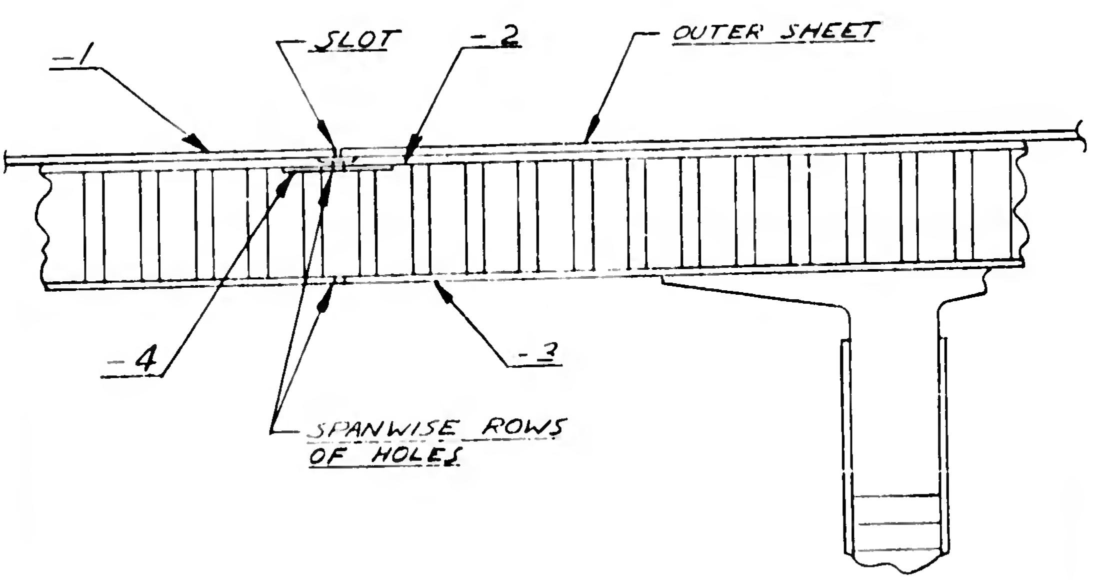

FIGURE 17

For lower wing loadings a solid skin in the rear suction region will be relatively heavy. In this case, a double skin construction (sandwich, Fig. 17, or chemically milled, Fig. 18) has less weight than a solid wing skin. Fabrication procedure begins with the machining of grooves and lands in the dash 2 skin and machining of the honeycomb to the proper thickness, including cutouts for the lands if they are to be used. Members dash 2 and dash 4 are bonded together and members dash 2 and dash 3 are then bonded to the core using the final die as a form. Next, the holes are drilled, and this must be done after bonding to insure

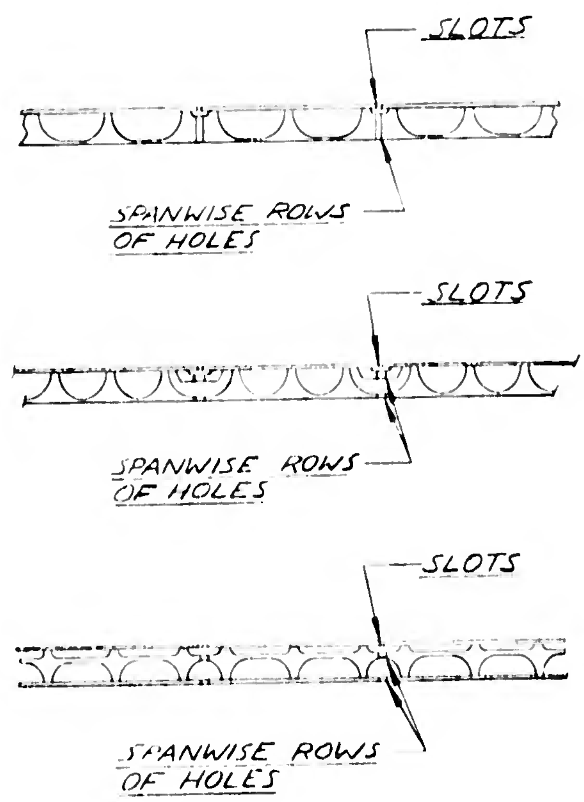

FIGURE 18

a clear passageway entirely through the assembly. Sheet dash 1 is applied using the close tolerance female die as in the solid skin method and for the same reasons. Finally, the slots are cut and the hand-finishing done.

This last operation may, of course, be delayed for safety's sake until wing assembly operations are completed, since the slotting operation is performed by portable equipment. Figures 19 through 23 show other schemes:

Figure 19 shows a honeycomb sandwich assembly with plastic overlays forming the slots and plenum chambers (see Fig. 19). Because of the softness of the plastic, great difficulty was experienced in holding the required surface smoothness and protecting from subsequent damage.

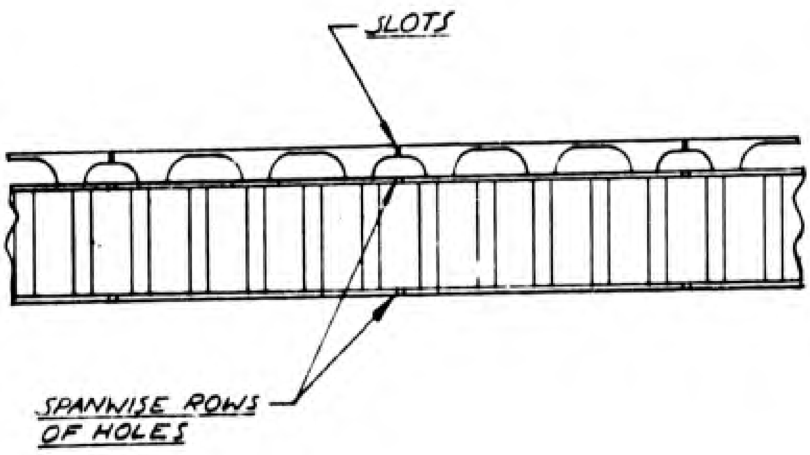

FIGURE 19

A solution similar to that of Fig. 19 except for the substitution of aluminum for plastic proved difficult in holding proper slot tolerances during bonding of the overlay to the sandwich.

Figures 20 and 21 illustrate two methods whose greatest drawbacks are the problems of holding slot tolerances and of providing a close tolerance adhesive thickness.

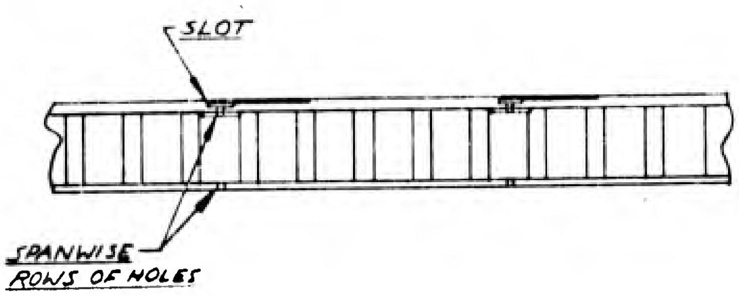

FIGURE 20

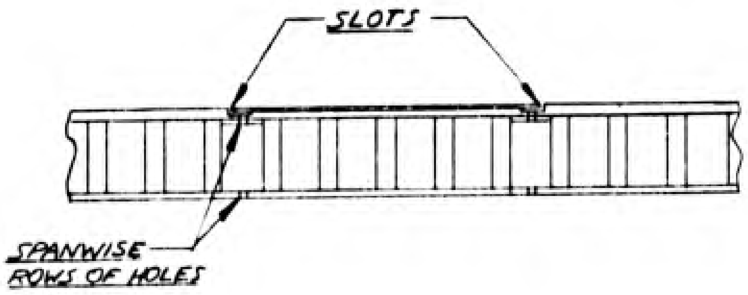

FIGURE 21

A variation of Figure 17 is shown in Figures 22a and 22b.

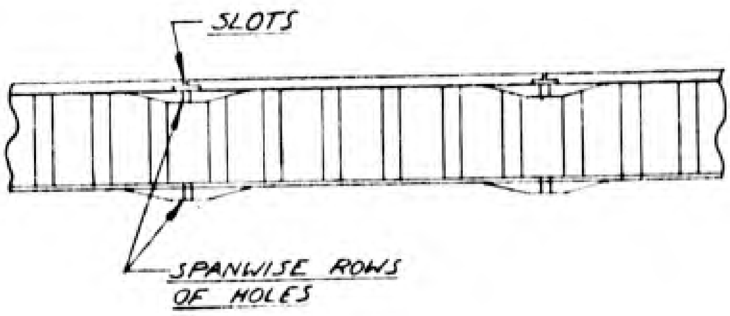

FIGURE 22a

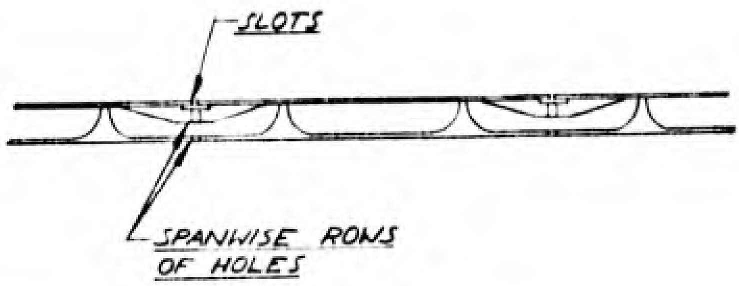

FIGURE 22b

The chemically milled panel of Figure 22b is light and structurally efficient.

Figures 23a and 23b show suction through a larger number of rows of closely-spaced small holes in a double-skin chemically milled sandwich.

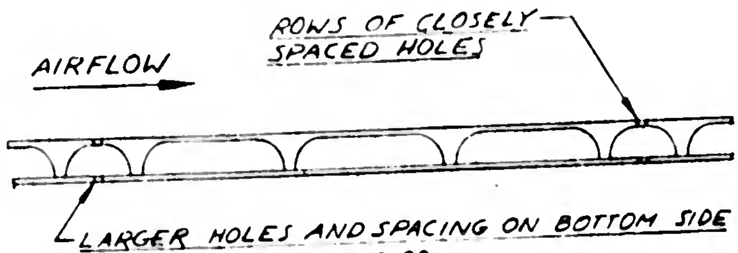

FIGURE 23a

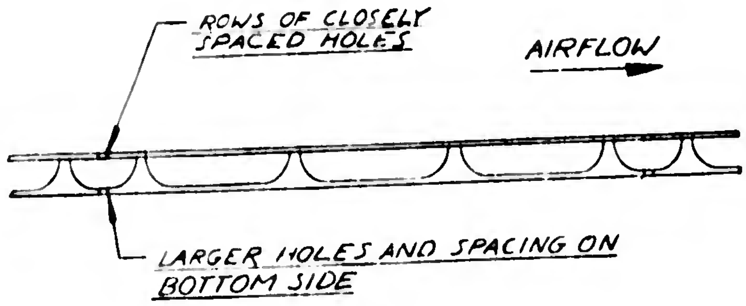

FIGURE 23b

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