Experimental Aerodynamics

B. Tech VI semester (Autonomous IARE R-16)

BY

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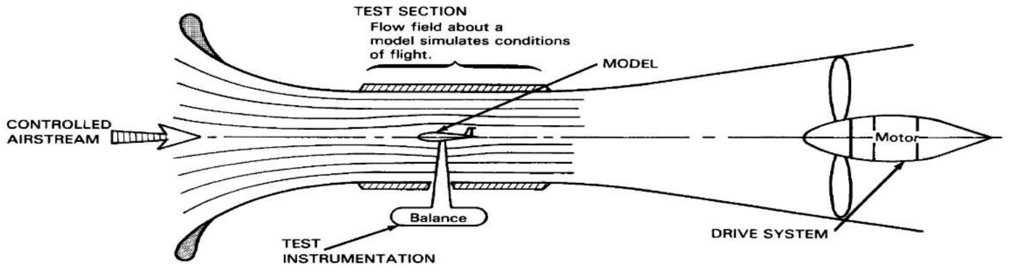
Unit I
FUNDAMENTALS OF EXPERIMENTS IN AERODYNAMICS

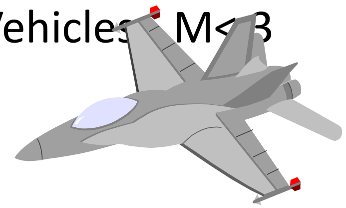
- Objective
  - Accurately simulate the fluid flow about atmospheric vehicles
  - Measure -Forces, moments, pressure, shear stress, heat transfer, flowfield (velocity, pressure, vorticity, temperature)



Low Speed Vehicles $M<.3$

U_∞

Gallilean Transformation



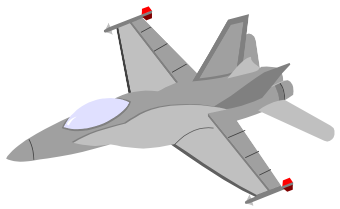
Flight in atmosphere

Scale =L

Wind Tunnel - Model Scale = ℓ

Stationary Walls

U_∞



Issues

Flow Quality - Uniformity and Turbulence Level

Wind Tunnel Wall Interference

Reynolds Number Simulation

$$\mathrm{Re} = \frac{\rho U_\infty L}{\mu} \neq \frac{\rho U_\infty \ell}{\mu}$$

Reynolds Number Scaling

- Most important on vehicles with partial laminar flow. The transition is very sensitive to Reynolds Number
- Use “trip strips”or roughness to cause boundary layer transition on the model at the same location as on the full scale vehicle

Transonic Regime .7<M<1.2

- Must Match Reynolds Number and Mach Number

$$\mathrm{Re} = \frac{\rho U_{\infty} L}{\mu}$$

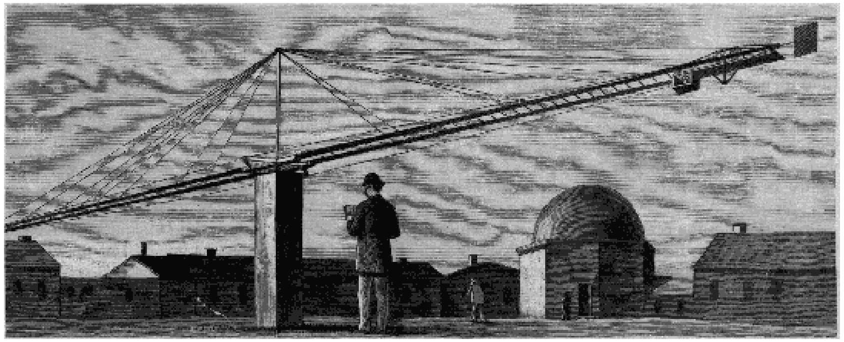
$$M = \frac{U_{\infty}}{c}$$

Must change fluid density and viscosity to match Re and M
Cryogenic Wind Tunnels are designed for this reason

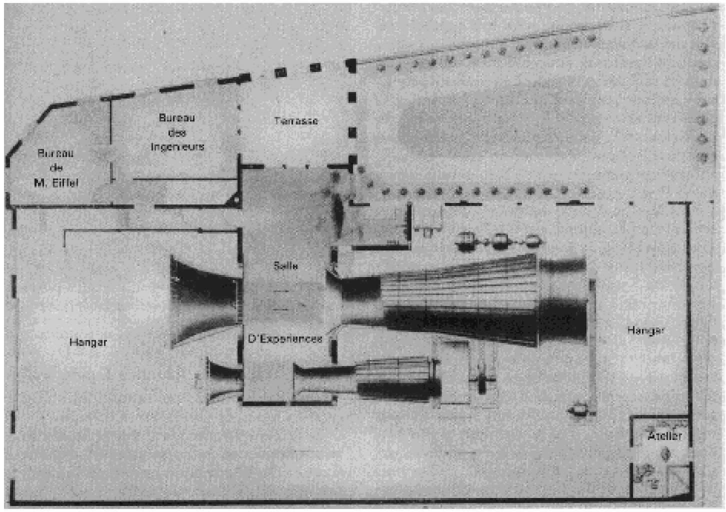
Unit-II

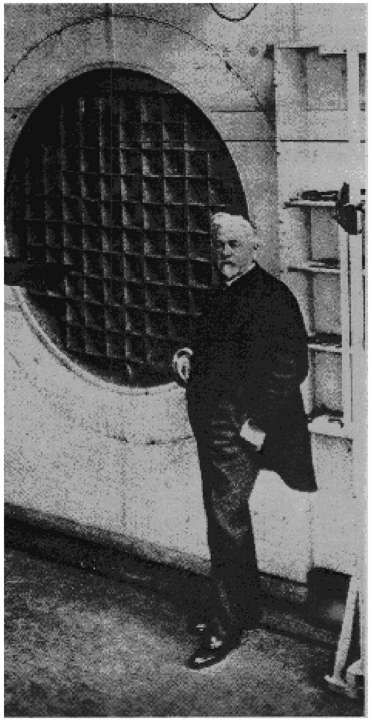
WIND TUNNEL EXPERIMENTATION CONSIDERATIONS

History Whirling Arm

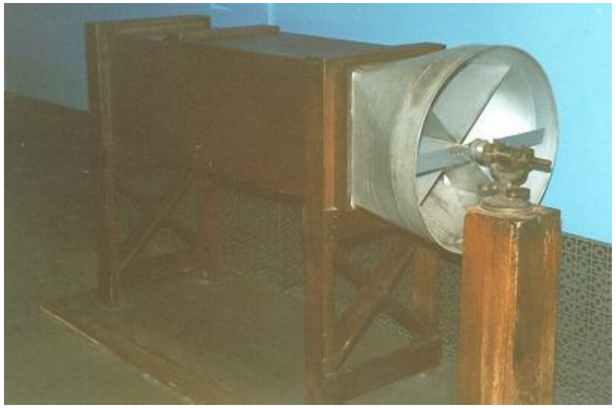


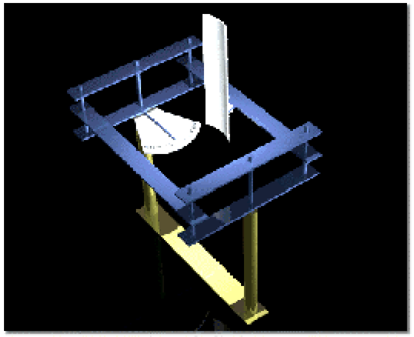
Eiffel Tunnel



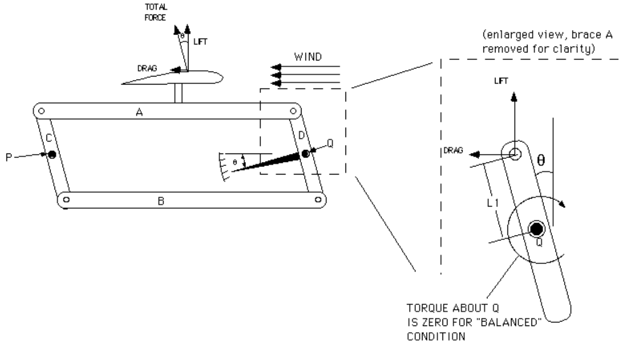


Wright Brothers

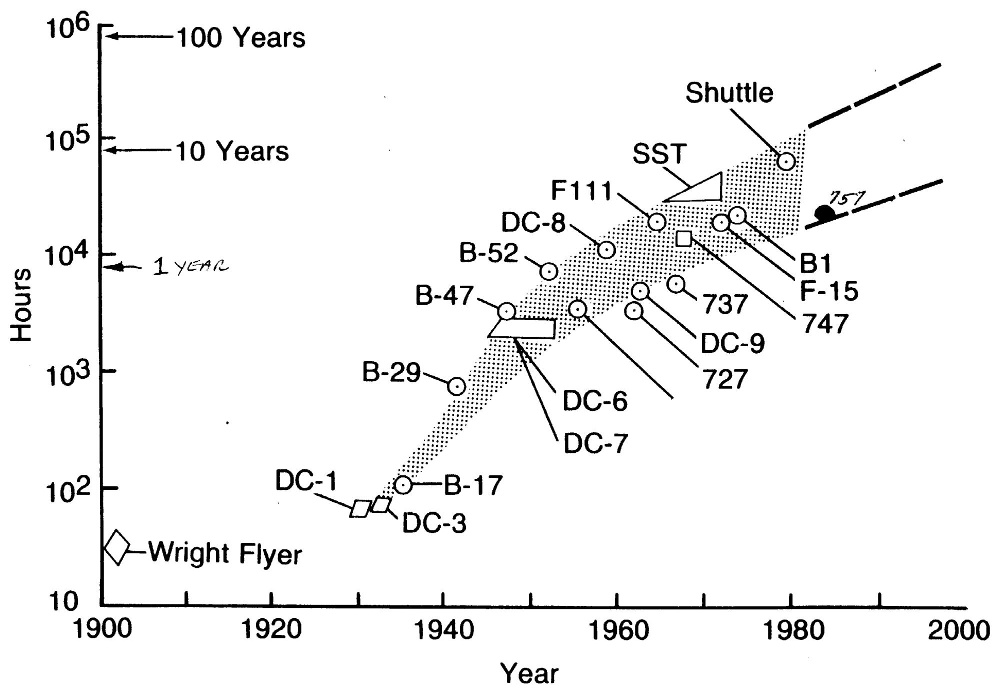




The Wright Brother's "Drift" Balance
(top view)



Wind Tunnel Test Trend

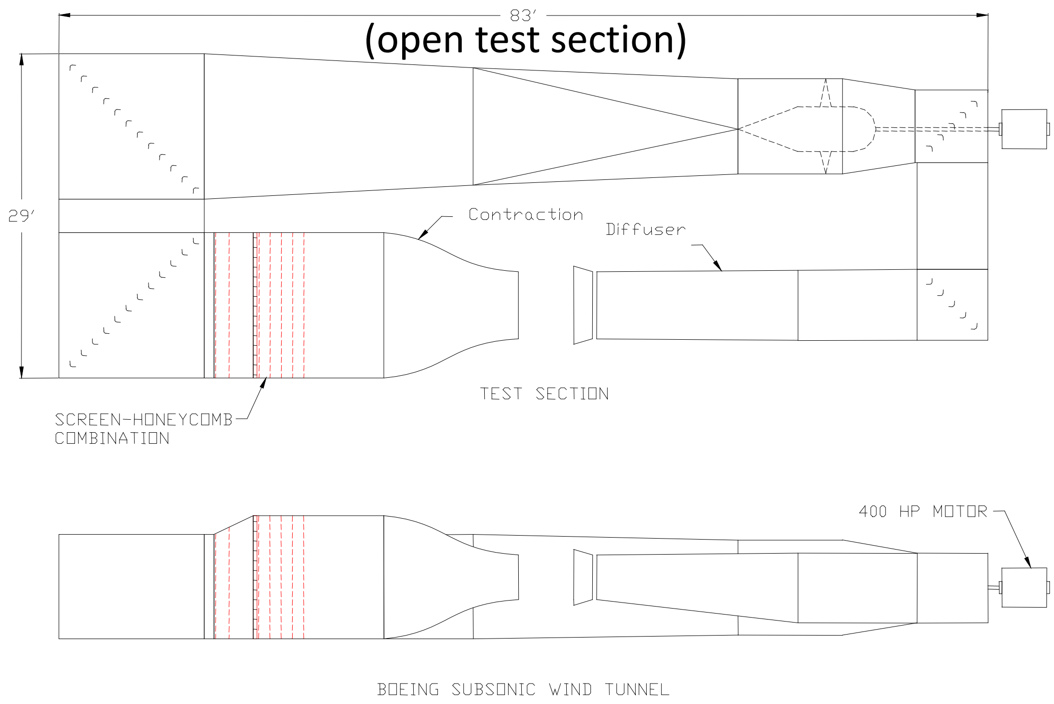


Wind Tunnel Layout

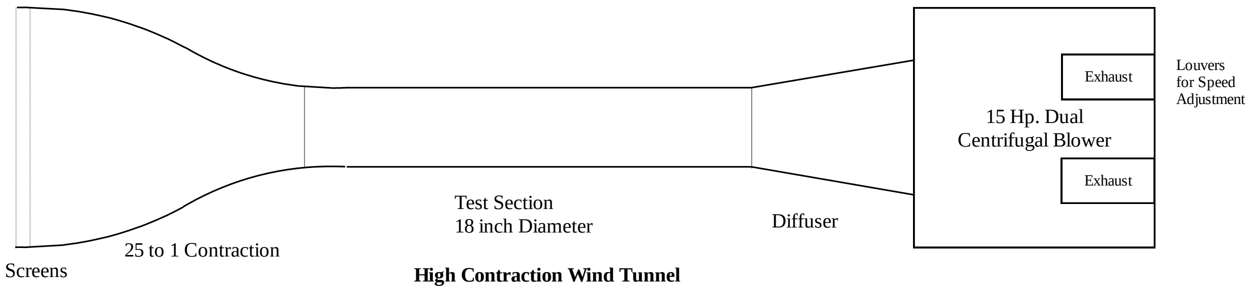
- Closed Return
- Open Return
- Double Return
- Annular Return

Closed Return

(open test section)

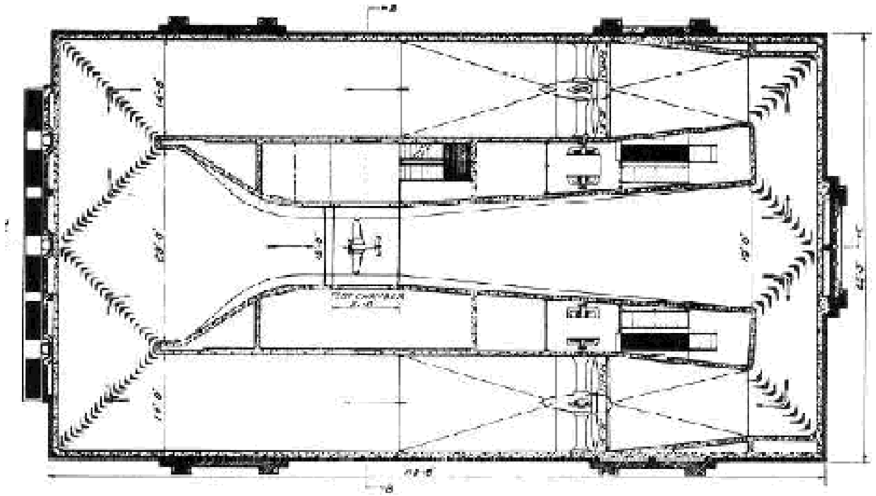


Open Return
Closed Test Section



High Contraction Wind Tunnel
Top View

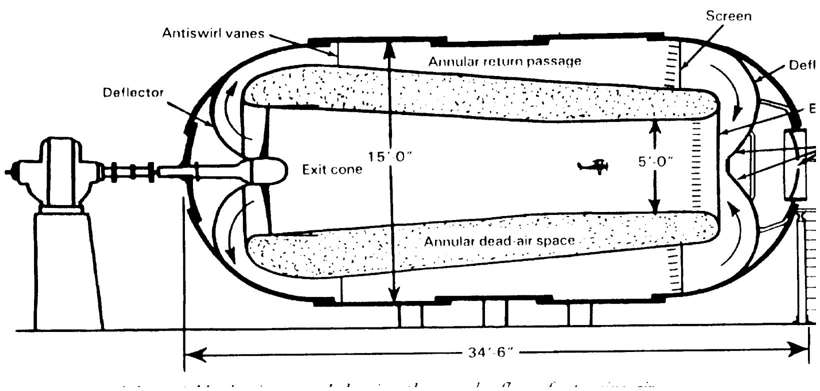
Double Return

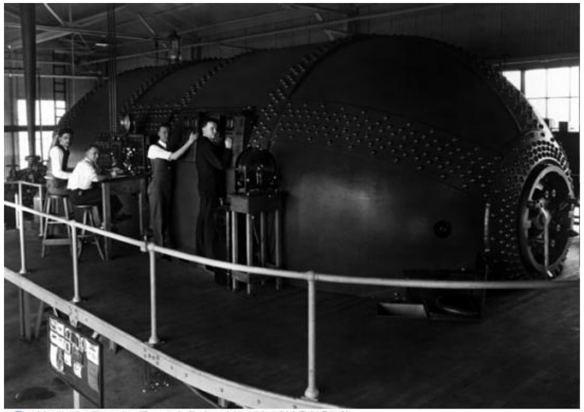


UNIVERSITY OF WASHINGTON
AERONAUTICAL LABORATORY
Kirsten Wind Tunnel

Annular Wind Tunnel

WIND TUNNELS OF NASA





Variable Density Tunnel, Being Used by NASA Staff
NASA Langley Research Center 3/15/1929 Image # EL–1996–00143

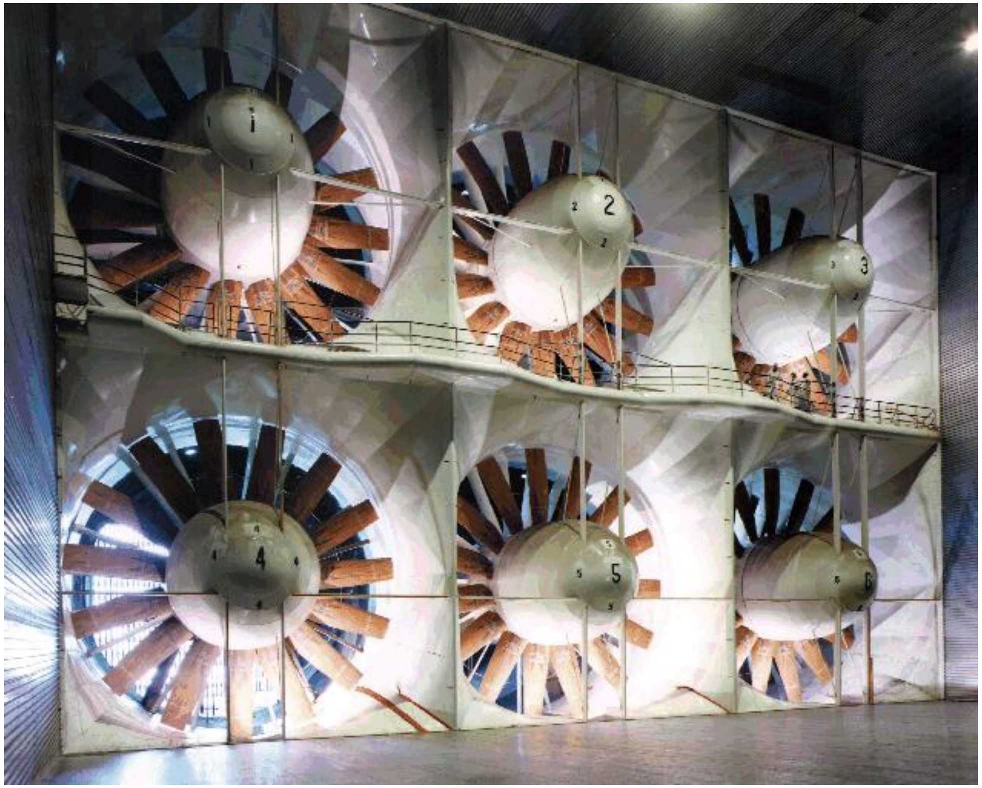
Types of Wind Tunnels

- Subsonic
- Transonic
- Supersonic
- Hypersonic
- Cryogenic
- Specialty
  - Automobiles
  - Environmental- Icing, Buildings,

Subsonic
Wind Tunnels

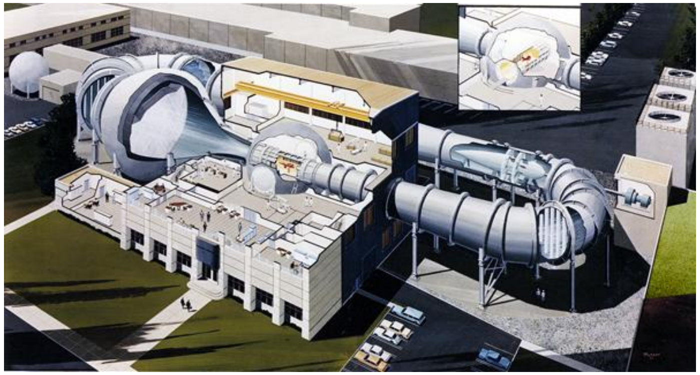
40’ x 80’ and 80’ x 120’
NASA Ames

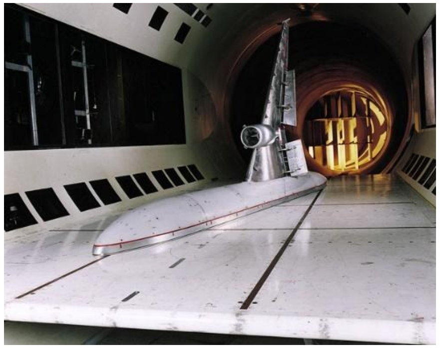
Fans for 40x80 and 80x120



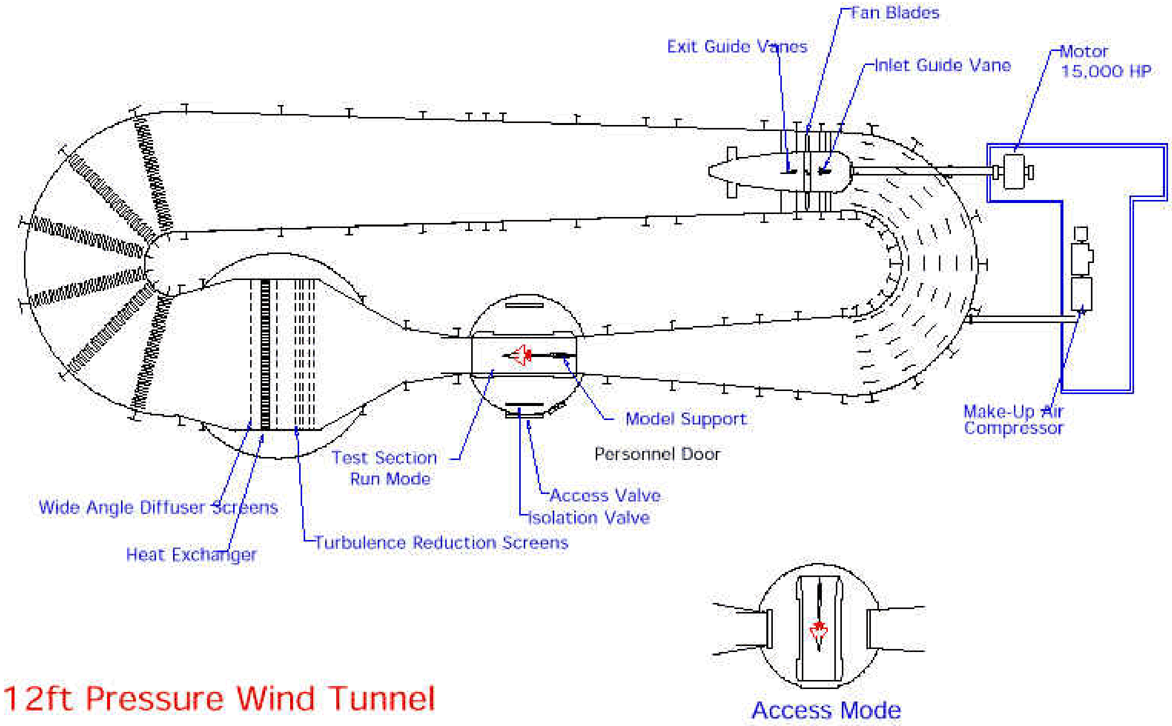
40’x80’

80’x120





12 foot Pressure Tunnel



12ft Pressure Wind Tunnel
NASA Ames Research Center

12-Foot Pressure Wind Tunnel: Specifications

Primary Use:

The facility is used primarily for high Reynolds number testing, including the development of high-lift systems for commercial transports and military aircraft, high angle-of-attack testing of maneuvering aircraft, and high Reynolds number research.

Capability:

- Mach Number: 0-0.52
- Reynolds Number per foot: 0.1 - 12X106
- Stagnation Pressure, PSIA: 2.0 - 90
- Temperature Range: 540 ° - 610 ° R
- Closed circuit, single return, variable density, closed throat, wind tunnel with exceptionally low turbulence
- Model-support systems available:
  - Strut with variable pitch and roll capability
  - High angle-of-attack turntable system
  - Dual-strut turntable mechanism for high-lift testing
  - Semispan mounting system
- Internal strain-gage balances used for force and moment testing
- Capability for measuring multiple fluctuating pressures
- Temperature-controlled auxiliary high-pressure (3000 psi)

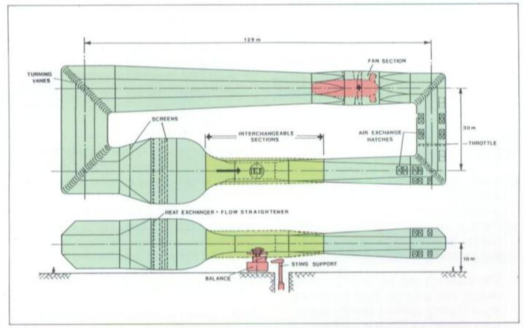


Fig. 3

DNW's lay-out and main dimensions

Fig. 4

The two test sections used for road vehicle aerodynamics: 8 m x 6 m for cars, 9.5 m x 9.5 m for trucks.

Transonic
Wind Tunnels

Transonic Wind Tunnels

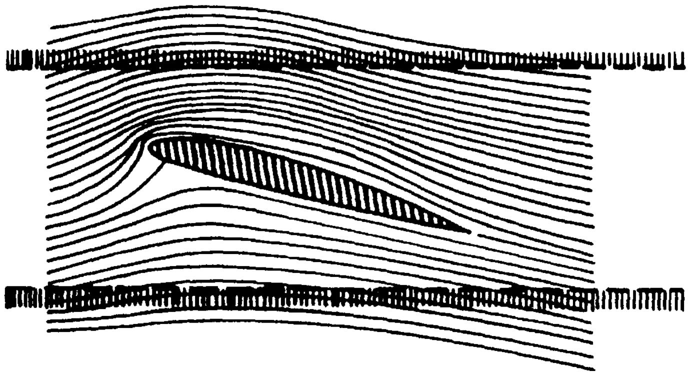
Wall interference is a severe problem for transonic wind tunnels.

Flow can "choke"

- Shock wave across the tunnel test section

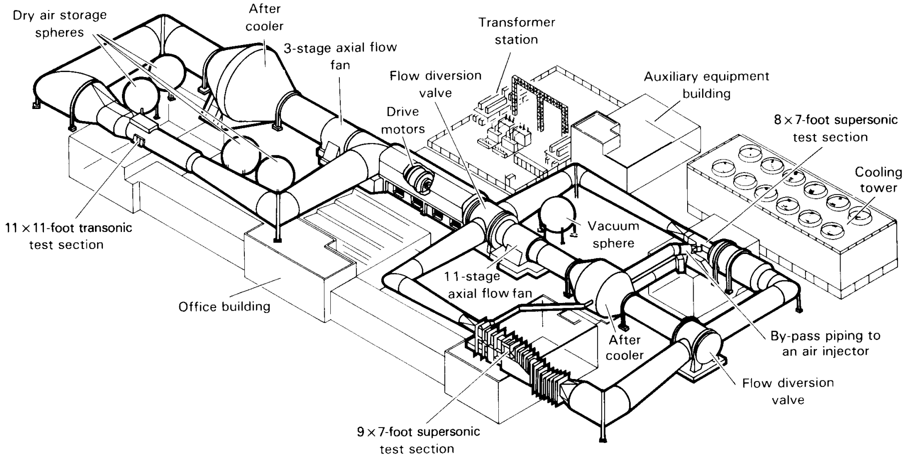
Two Solutions

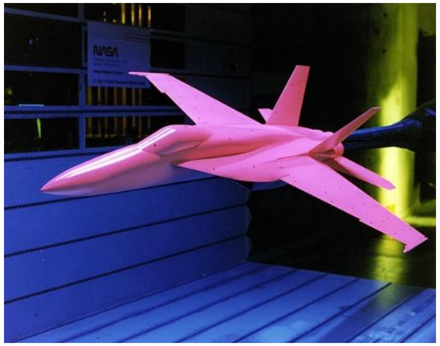
- Porous Walls
- Movable Adaptive Walls



The Unitary Plan wind tunnels are a set of three interconnected tunnels that share a central main drive system that can be used to drive either a transonic leg or a supersonic leg. The Unitary Plan wind tunnels are as follows.

- 11ft Transonic Wind Tunnel
- 9x7ft Supersonic Wind Tunnel
- 8x7ft Supersonic Wind Tunnel





The 8x6/9x15 Complex at the NASA Lewis Research Center in Cleveland, Ohio is, is unique in its dual capacity role as both a high-speed and low speed test facility.

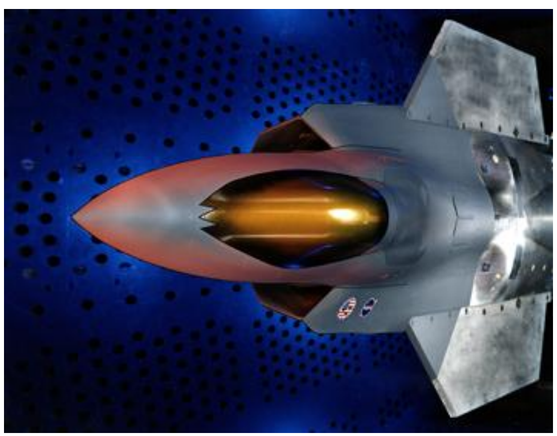
8x6 Functions & Capabilities

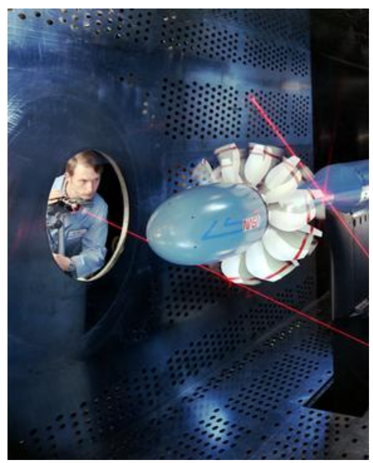
The 8x6 Foot Supersonic Wind Tunnel provides customers with a Facility ca
scale aeropropulsion hardware:

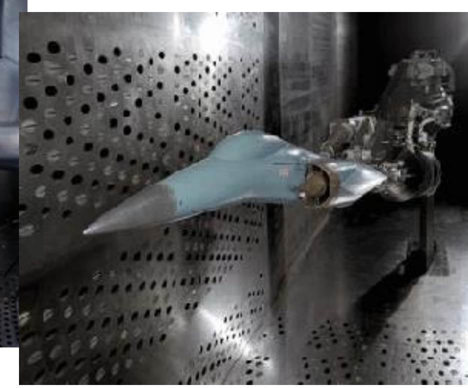
8x6 Characteristics & Performance

Test section size	8ft H, 6ft W, 23.5ft L
Mach number range	0 - 2.0
Relative altitude	1000 - 35000 ft
Dynamic Pressure	3.6 - 4.8 x 106/ft
Stagnation Pressure	15.3 - 25 psia
Temperature	60 - 250oF

8x6 at
NASA Lewis





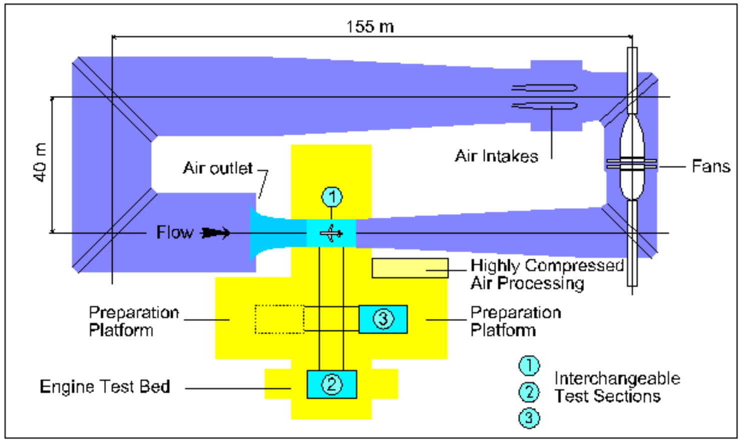


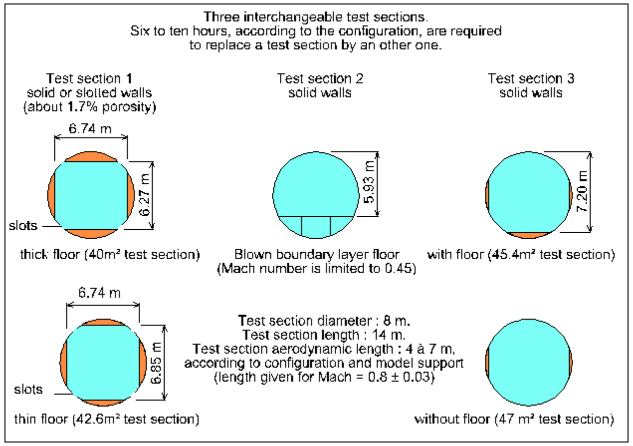
9x15 at NASA Lewis
Back Leg of the 8x6

Modane-Avrieux

S1MA Wind Tunnel Atmospheric, closed-circuit, continuous flow wind tunnel, from Mach 0.05 to Mach 1

S1MA wind tunnel is equipped with two counterrotating fans, driven by Pelton turbines, the power of which is 88 MW;
Mach number is continuously adjustable from 0.05 to 1 by varying the fan speed from 25 to 212 rpm.

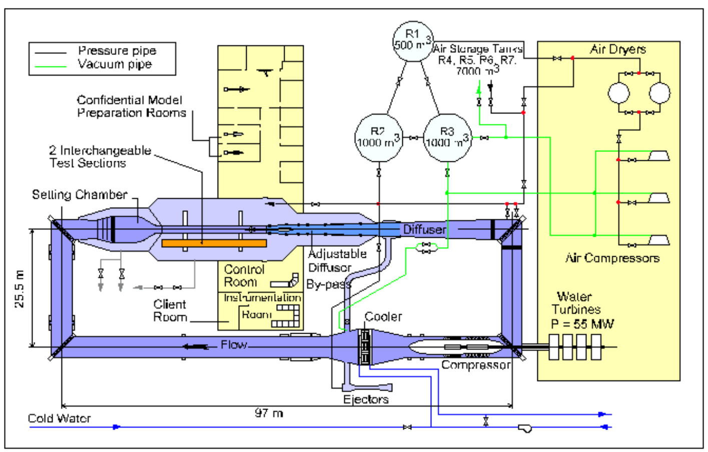


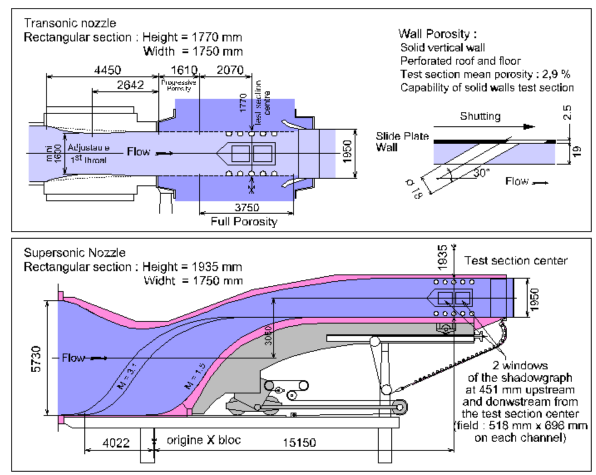


16T at AEDC

A model of Boeing's 767 commercial jet undergoes testing in one of AEDC's large wind tunnels. The 767 tests were the first in a series of tests of Boeing's large commercial jets at the center. AEDC signed a twenty year alliance with Boeing to test commercial aircraft.

S2Ma Wind Tunnel





Supersonic
Wind Tunnels

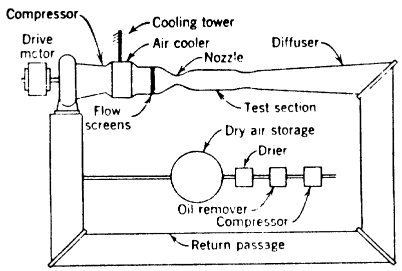


Fig. 1:1 Diagrammatic layout of closed-circuit, continuous flow, supersonic wind tunnel.

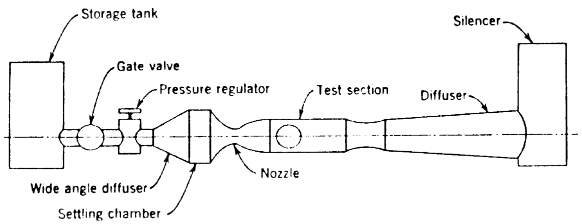


Fig. 1:2 Diagrammatic layout of intermittent blowdown tunnel.

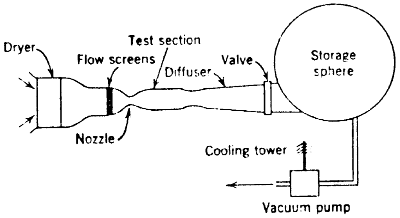
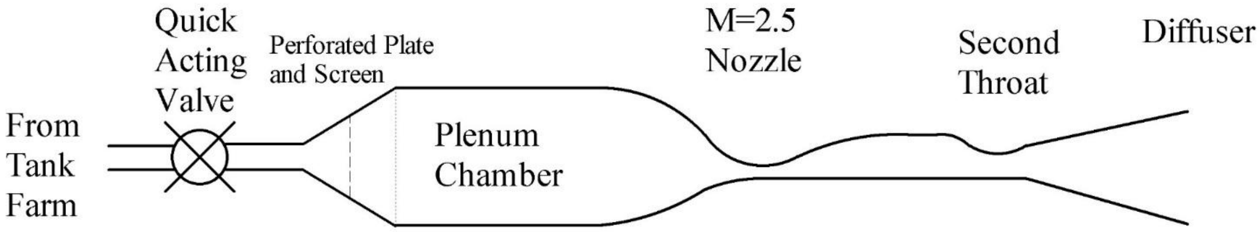
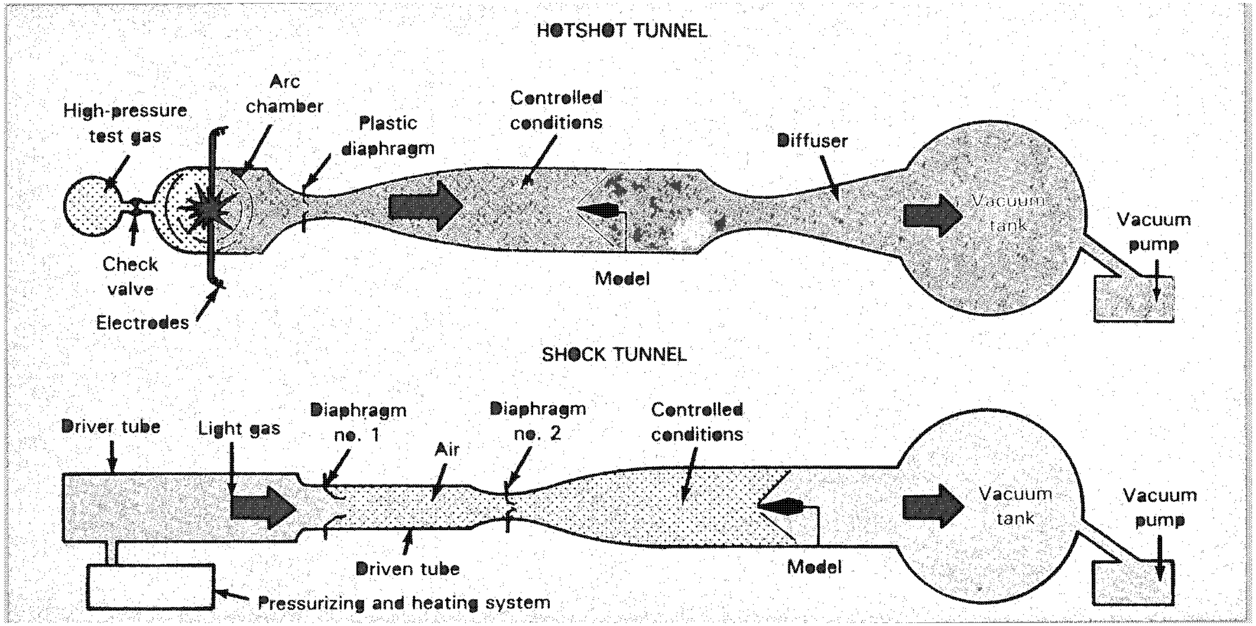


Fig. 1:3 Diagrammatic layout of intermittent indraft wind tunnel.

Purdue University
Aerospace Sciences Laboratory
M=2.5 Supersonic
Blowdown Wind Tunnel

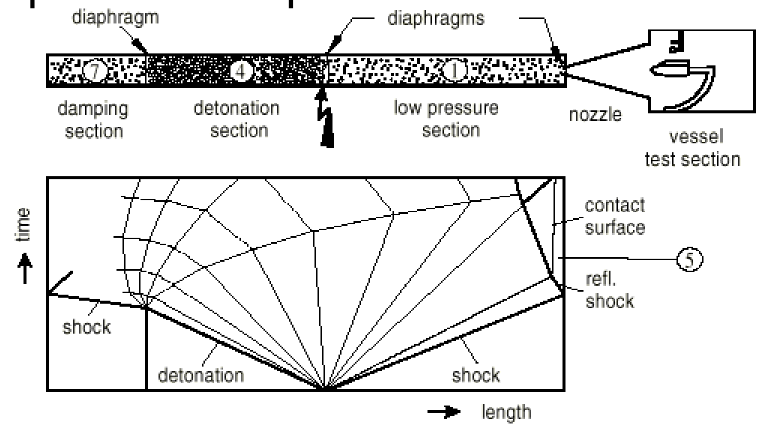


Hypersonic
Wind Tunnels


HOTSHOT TUNNEL
High-pressure test gas
Arc chamber
Plastic diaphragm
Controlled conditions
Diffuser
Vacuum tank
Vacuum pump
Check valve
Electrodes
Model
SHOCK TUNNEL
Driver tube
Light gas
Diaphragm no. 1
Air
Diaphragm no. 2
Controlled conditions
Vacuum tank
Vacuum pump
Driven tube
Model
Pressurizing and heating system

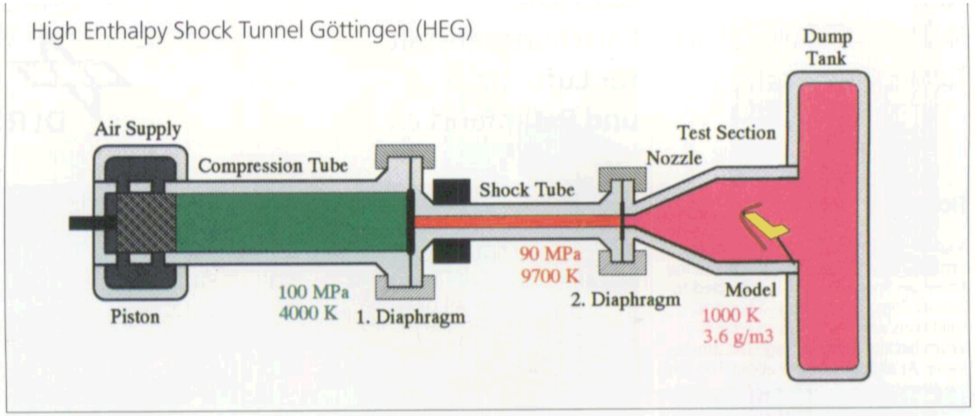
Principle Operation Detonation Driven Shock Tunnel

Set- up and wave plan:



Initial conditions:

- low pressure section: test gas air, about 25 kPa for tailored cond.
- deton. section: oxyhydrogen- helium/ argon mixtures, max. 7 MPa
- damping section: expansion volume; low initial pressures

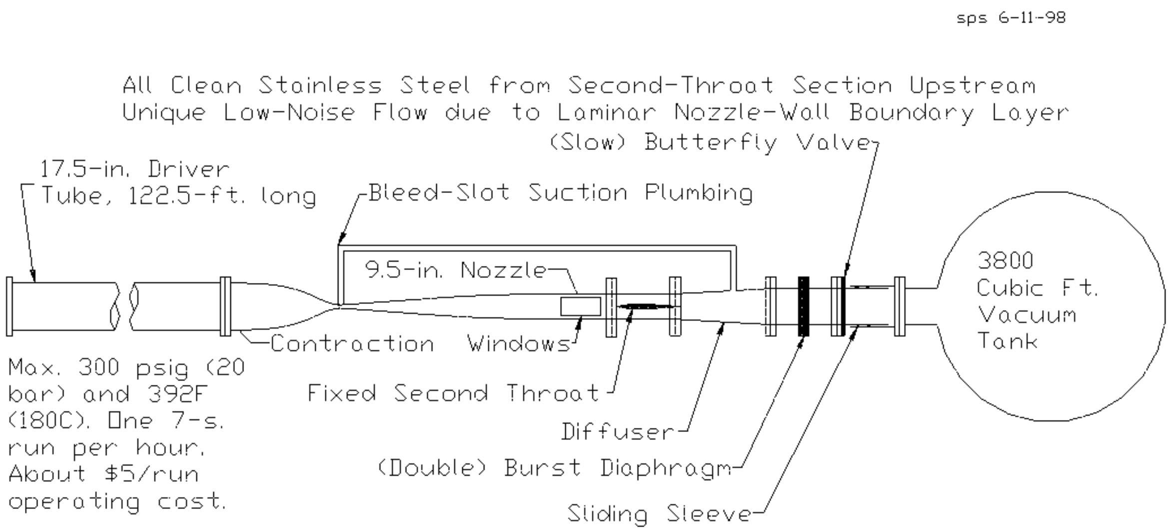


The Facility

The free piston-driven shock tunnel HEG consists of an air buffer, a compression (driver) tube, separated from an adjoining shock tube via a metal diaphragm, and a subsequent nozzle and test section. A piston is accelerated through the compression tube by the air in the buffer, compressing the driver gas helium to high temperatures and pressures, whereby the diaphragm ruptures, leading to propagation of a strong shock through the shock tube. This shock reflects from the end wall, heating up the test gas (nitrogen, air, carbon dioxide, etc.) to high pressures and temperatures – this gas reservoir expands through the nozzle and provides the free stream conditions in the test section. Total available test time is about 1 millisecond.

Condition	I	II	III	IV	V	VI
P_0 (MPa)	40	90	45	110	50	95
T_0 (K)	9100	9700	7300	8100	6400	6500
h_0 (MJ/kg)	21	22	13	15	11	11
p_∞ (Pa)	430	1200	470	1300	520	980
T_∞ (K)	790	1040	550	720	470	480
ρ_∞ (g/m^3)	1.6	3.6	2.8	6.2	3.8	6.9
M_∞	9.7	9.0	10.0	9.5	10.0	10.0
u_∞ (m/s)	5900	6200	4800	5100	4400	4400

HEG standard operating conditions



Schematic of Boeing Mach-6 Quiet-Flow Ludwieg Tube

The NASA Langley 8-Foot High Temperature Tunnel (8' HTT) enables the testing of large hypersonic airbreathing propulsion systems at flight enthalpies from Mach 4 to Mach 7.

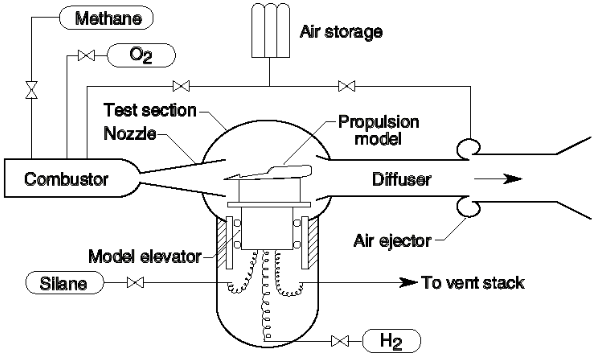


Figure 2. Schematic drawing of the 8' HTT for airbreathing propulsion testing.

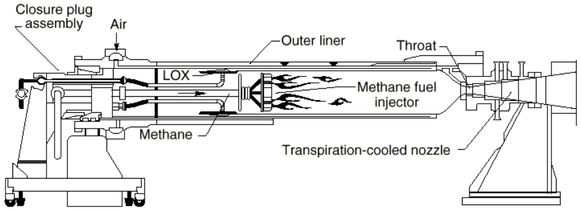
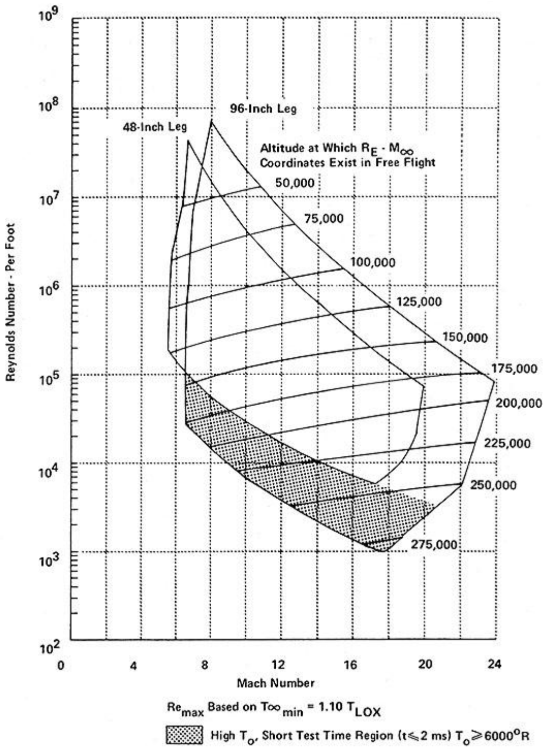


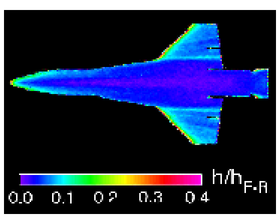
Figure 3. Schematic drawing of the 8' HTT combustor.

Hypersonic Shock Tunnels at Calspan

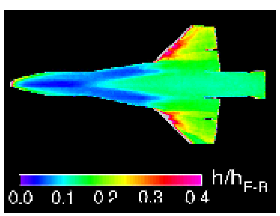
The performance chart shows that the high enthalpy 96-inch tunnel is capable of simultaneously duplicating velocity (total enthalpy) and density altitude over a wide range of hypersonic flight conditions. These test conditions cover the widest range of any in the country.



Calspan Hypersonic Shock Tunnel Performance



Re_∞ = 1.1 x 10^6/ft



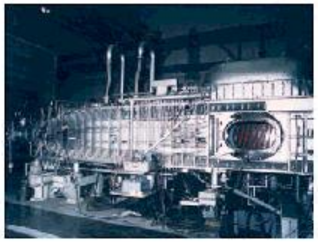
Re_∞ = 7.9 x 10^6/ft

Fig. 6 Effect of Reynolds number on windward heating rates for X-34 at M_∞ = 6, $\alpha = 0°$, and $\delta_{CS} = 0°$.

15-Inch Mach 6 Hi Temp. Air

20-Inch Mach 6 Air

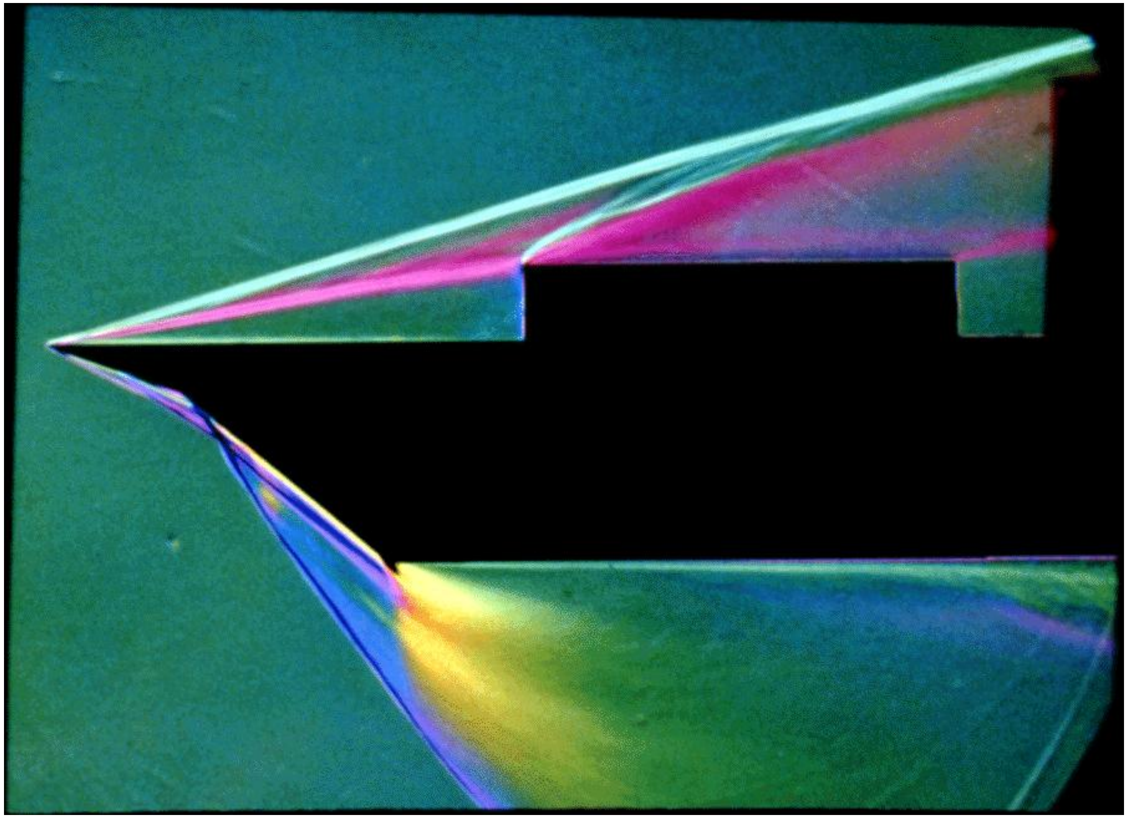
20-Inch Mach 6 CF_4

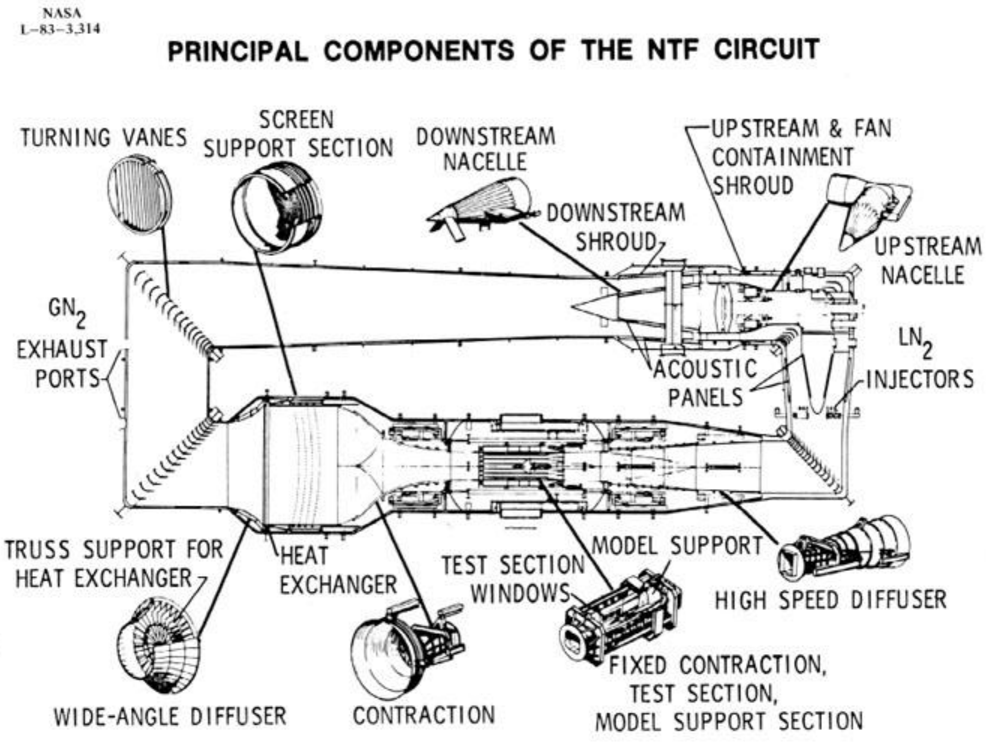


31-Inch Mach 10 Air

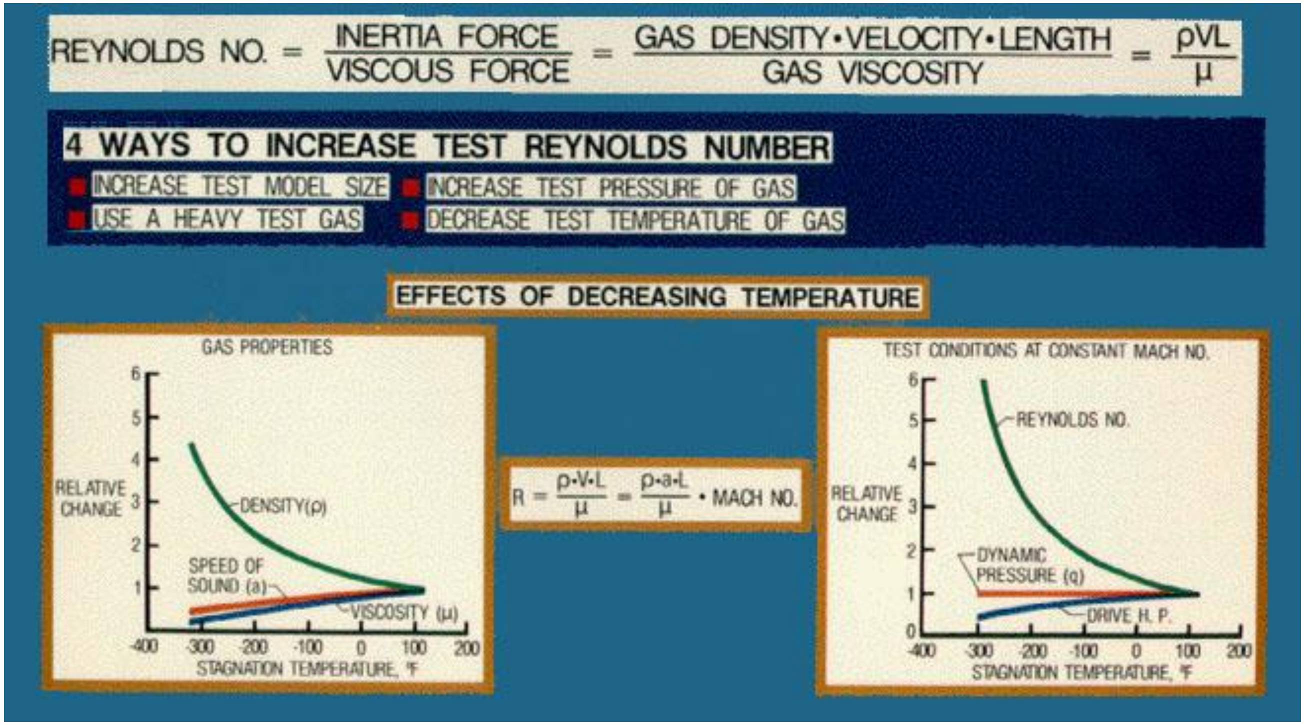
22-Inch Mach 15/20 He

Fig. 1 Facilities of the Aerothermodynamic Facilities Complex.




NASA
L-83-3,314
PRINCIPAL COMPONENTS OF THE NTF CIRCUIT
TURNING VANES
SCREEN
SUPPORT SECTION
DOWNSTREAM
NACELLE
UPSTREAM & FAN
CONTAINMENT
SHROUD
DOWNSTREAM
SHROUD
UPSTREAM
NACELLE
GN2
EXHAUST
PORTS
LN2
INJECTORS
ACOUSTIC
PANELS
TRUSS SUPPORT FOR
HEAT EXCHANGER
HEAT
EXCHANGER
TEST SECTION
WINDOWS
MODEL SUPPORT
HIGH SPEED DIFFUSER
FIXED CONTRACTION,
TEST SECTION,
MODEL SUPPORT SECTION
WIDE-ANGLE DIFFUSER
CONTRACTION

Cryogenic
Wind Tunnels


REYNOLDS NO. = INERTIA FORCE / VISCOUS FORCE = GAS DENSITY•VELOCITY•LENGTH / GAS VISCOSITY = ρVL / μ
4 WAYS TO INCREASE TEST REYNOLDS NUMBER
INCREASE TEST MODEL SIZE
INCREASE TEST PRESSURE OF GAS
USE A HEAVY TEST GAS
DECREASE TEST TEMPERATURE OF GAS
EFFECTS OF DECREASING TEMPERATURE
GAS PROPERTIES
RELATIVE CHANGE
DENSITY(ρ)
SPEED OF SOUND (a)
VISCOSITY (μ)
STAGNATION TEMPERATURE, °F
R = ρ•V•L / μ = ρ•a•L / μ • MACH NO.
TEST CONDITIONS AT CONSTANT MACH NO.
REYNOLDS NO.
DYNAMIC PRESSURE (q)
DRIVE H. P.
STAGNATION TEMPERATURE, °F

NATIONAL TRANSONIC FACILITY

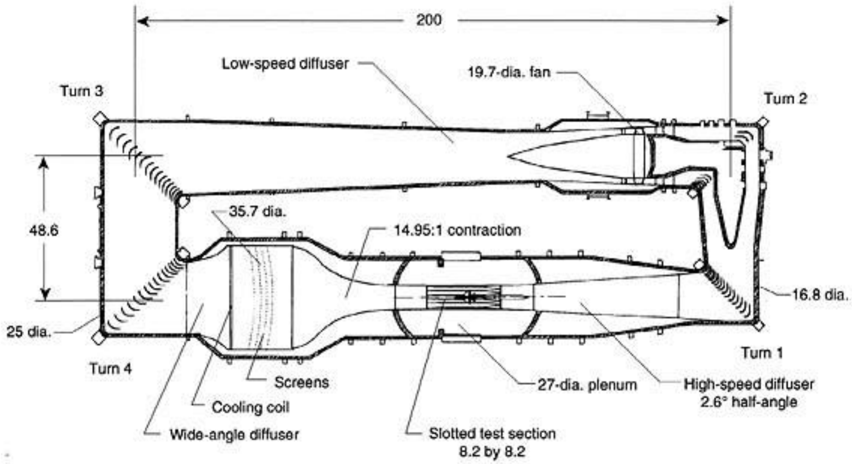
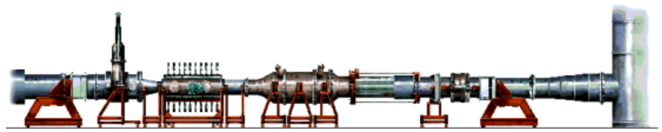
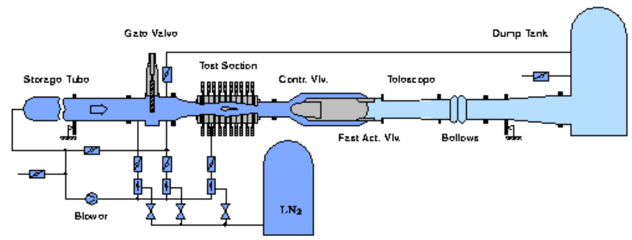


Figure 6. Plan view of NTF tunnel circuit. All linear dimensions are in feet.

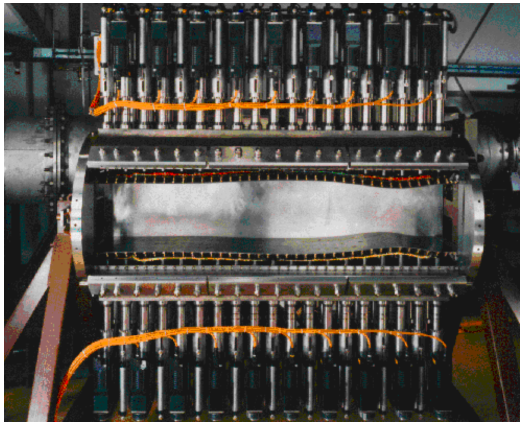
The Cryogenic Ludwieg-Tube at Göttingen (KRG)



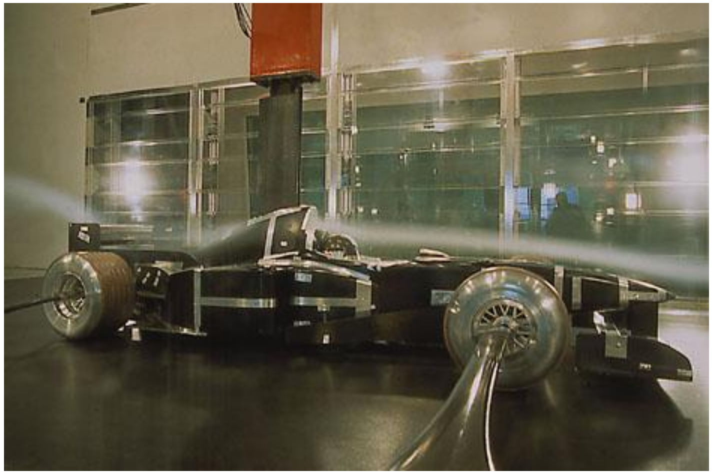


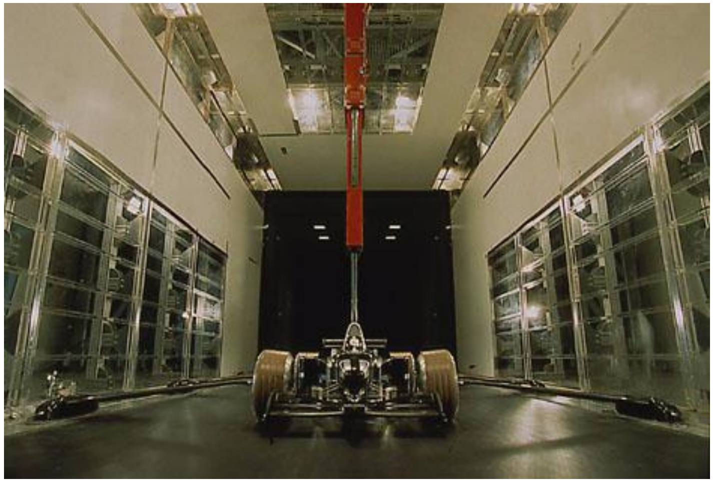
Tube	Diameter	0.8 m	Stag. press. (max.)	10 bar
	Length	130 m	Temperature range	100 to 300 K
	Load. press. (max.)	12.5 bar	Mach number range	0.25 to 0.95
Test Section	Cross section	$0.4 \times 0.35\ m^2$	Reynolds no. (max.)	$60 \cdot 10^6$
	Length	2.0 m	Run time	0.6 to 1.0 s
	Model chord (typ.)	0.15 m		

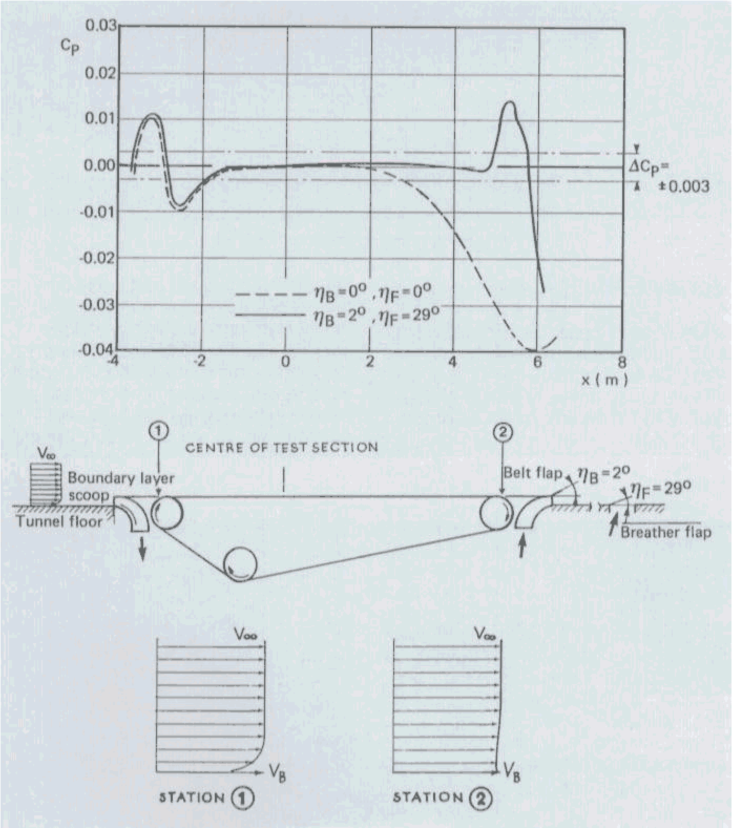
Adaptive wall test section



Automobile
Wind Tunnels



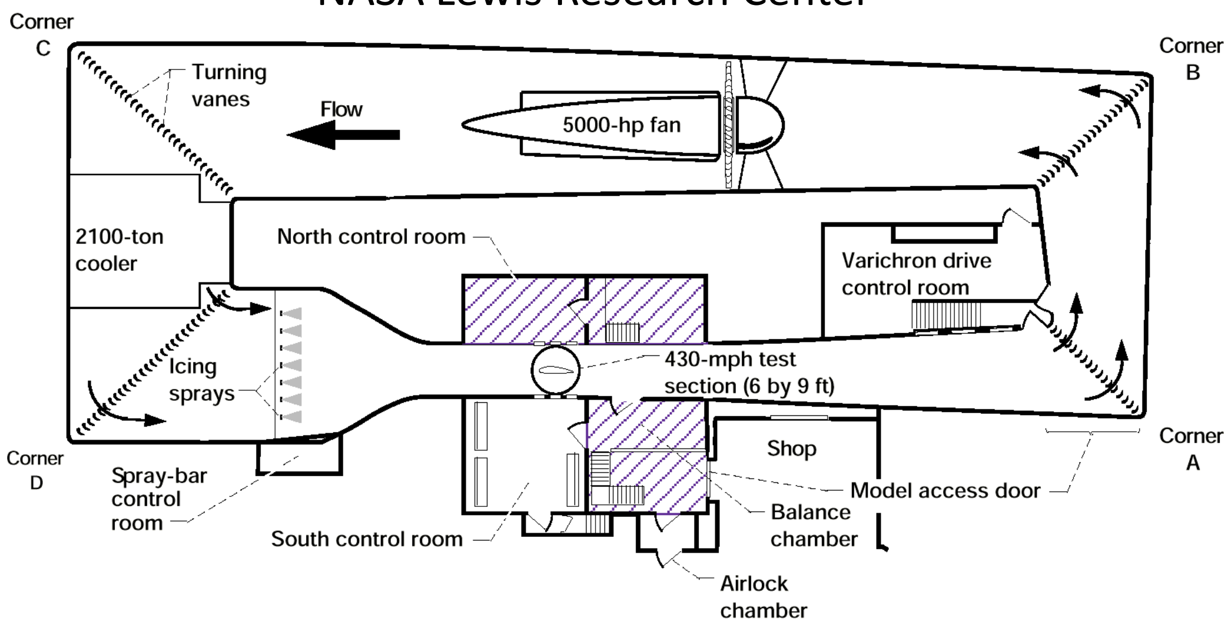


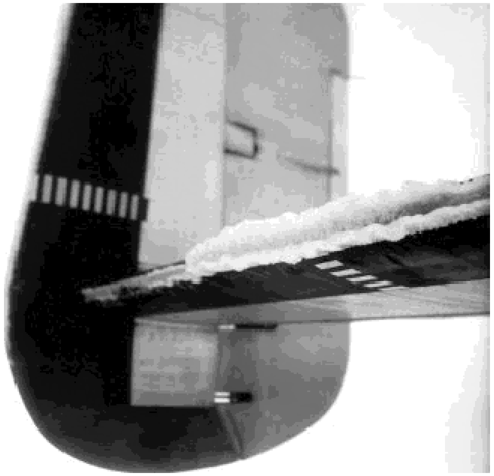


C_p
0.03
0.02
0.01
0.00
-0.01
-0.02
-0.03
-0.04
-4
-2
0
2
4
6
8
x (m)
ΔC_p=
±0.003
$\eta_B = 0^o$, $\eta_F = 0^o$
$\eta_B = 2^o$, $\eta_F = 29^o$
CENTRE OF TEST SECTION
V_∞
Boundary layer scoop
Tunnel floor
Belt flap
$\eta_B = 2^o$
$\eta_F = 29^o$
Breather flap
V_B
STATION 1
STATION 2

Icing
Wind Tunnels

Icing Tunnel
NASA Lewis Research Center





Wings

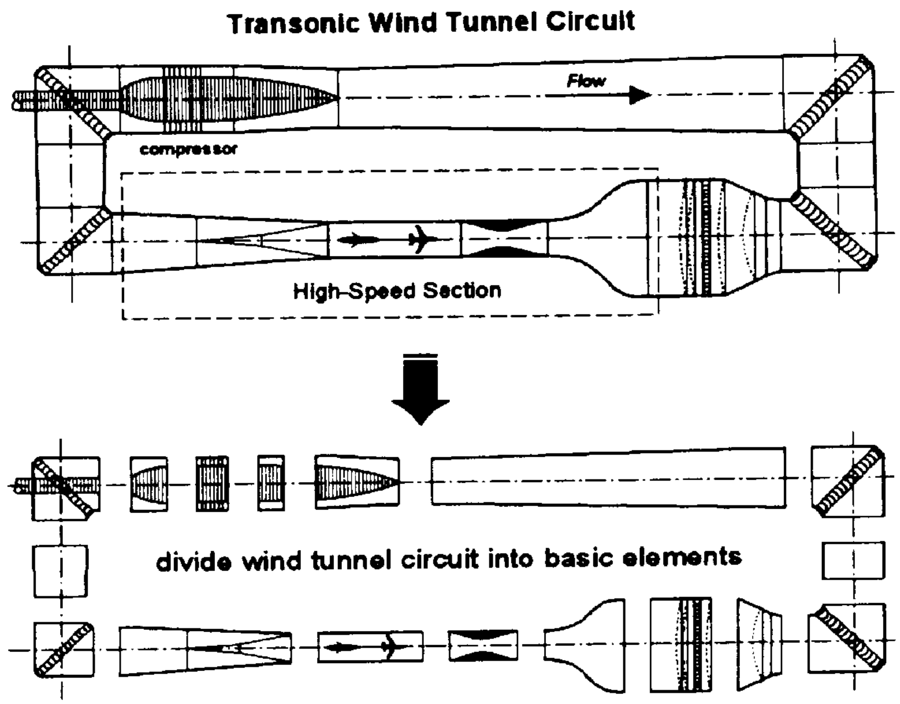
Wind Tunnel Power Requirements

Energy Ratio

$$(E.R.)_t = \frac{\text{Jet Energy}}{\sum \text{Circuit Losses}} = \frac{1/2\rho_0 U_0^3 A_0}{\sum \text{Losses}} = \frac{q_0 U_0 A_0}{\eta P}$$

Subscript 0 refers to the test section
P is the motor power
η is the fan efficiency

Wind Tunnel Circuit Elements



Losses

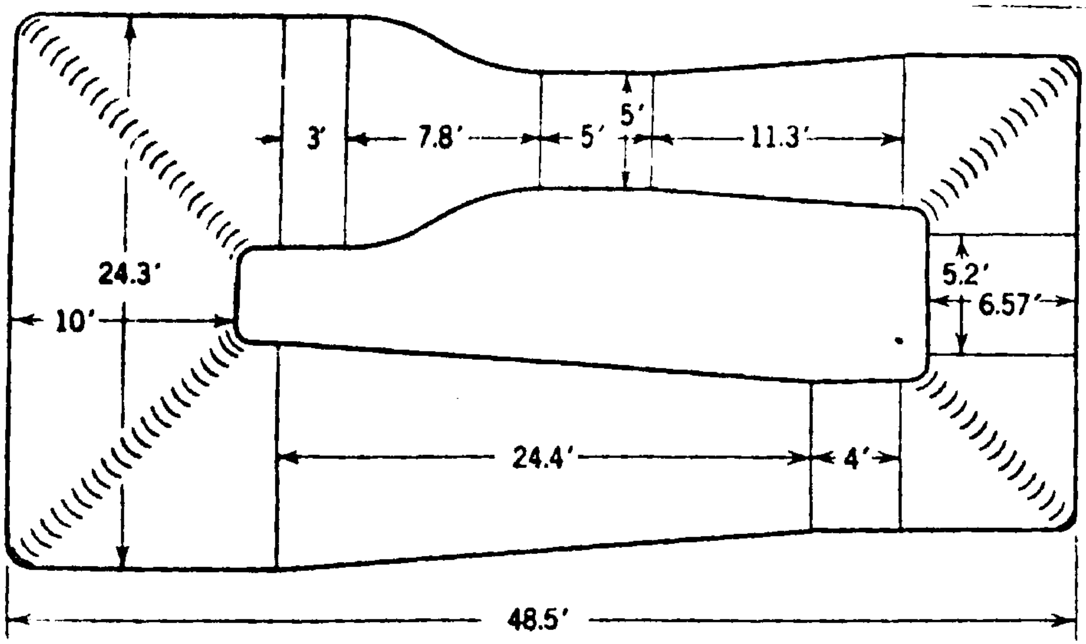
$$K = \frac{p_{t1} - p_{t2}}{q}$$ Local Pressure Loss Coefficient

$$K_0 = \frac{p_{t1} - p_{t2}}{q_0} = K\frac{q}{q_0}$$ Pressure Loss Referred to Test Section

$$\Delta E = K_0 1/2 \rho_0 U_0^3 A_0$$ Section Energy Loss

$$(E.R.)_t = \frac{\text{Jet Energy}}{\sum \text{Circuit Losses}} = \frac{1/2\rho_0 U_0^3 A_0}{\sum \mathrm{K}_0 1/2 \rho_0 U_0^3 A_0} = \frac{1}{\sum K_0}$$

Closed Re



Example - Closed Return Tunnel

	Section	K_0	% Total Loss
1	Test Section	.0093	5.1
2	Diffuser	.0391	21.3
3	Corner #1	.0460	25.0
4	Straight Section	.0026	1.4
5	Corner #2	.0460	25.0
6	Straight Section	.0020	1.1
7	Diffuser	.0160	8.9
8	Corner #3	.0087	4.7
9	Corner #4	.0087	4.7
10	Straight Section	.0002	.1
11	Contraction	.0048	2.7
	Total	**.1834**	**100.0**

$$(E.R.)_t = \frac{1}{\sum K_0} = \frac{1}{.1834} = 5.45$$

Example - Open Return Tunnel

	Section	Ko	% Total Loss
1	Inlet Including Screens	.021	14.0
2	Contraction and Test Section	.013	8.6
3	Diffuser	.080	53.4
4	Discharge at Outlet	.036	24.0
	Total	**.150**	**100.0**

$$(E.R.)_t = \frac{1}{\sum K_0} = \frac{1}{.150} = 6.67$$

Turbulence Management System

Stilling Section - Low speed and uniform flow

Honeycomb - Reduces Large Swirl Component of Incoming Flow

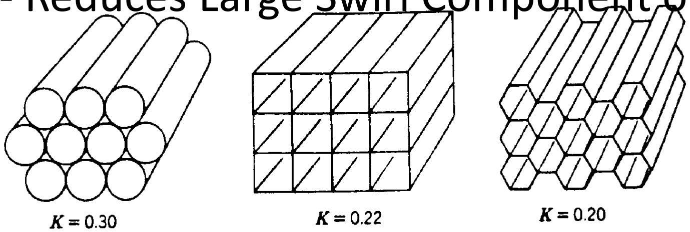


Fig. 2:16 Some honeycombs and their losses.

Screens - Reduce Turbulence [Reduces Eddy size for Faster Decay]
- Used to obtain a uniform test section profile
- Provide a flow resistance for more stable fan operation

Contraction

Establish Uniform Profile at Test Section
Reduce Turbulence

Test Section

Test Section - Design criteria of Test Section Size and Speed
Determine Rest of Tunnel Design

Test Section Reynolds Number
Larger JET - Lower Speed - Less Power - More Expensive

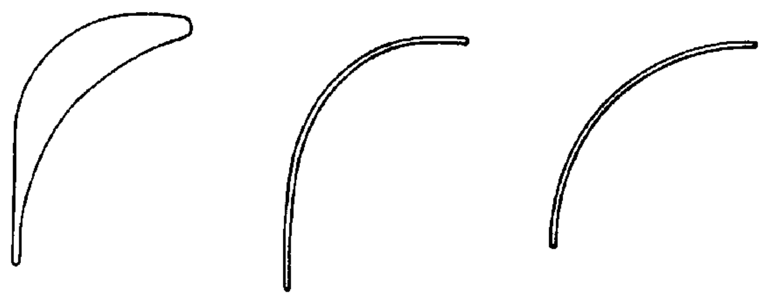
Section Shape - Round-Elliptical, Square, Rectangular-Octagonal with flats for windows-mounting platforms
Rectangular with filled corners
Not usable but requies power

For Aerodynamics Testing 7x10 Height/Width Ratio

Test Section Length - L = (1 to 2)w

Diffuser

Abrupt Corner without Vanes $\eta = 1.0$



a
$\eta = 0.11$

b
$\eta = 0.138$

c
$\eta = 0.20$

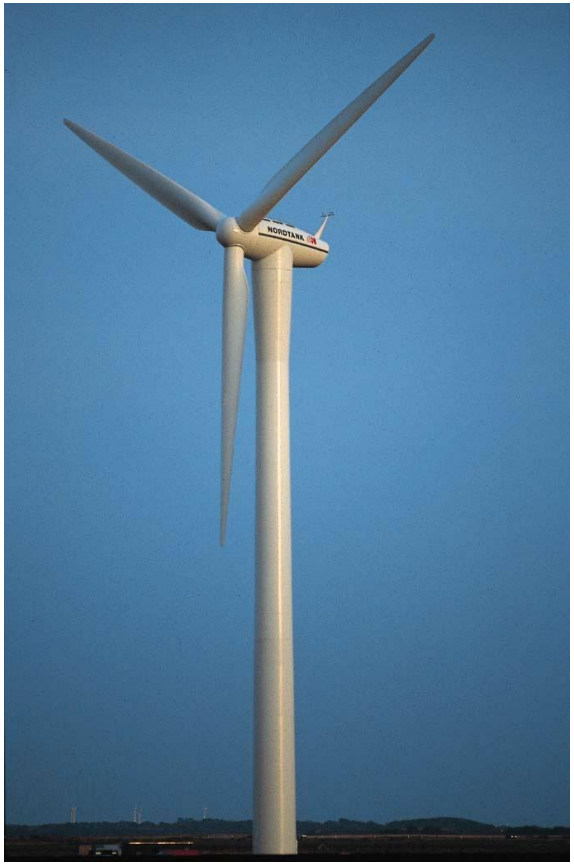

U.S. AIR FORCE
0601


NASA
614
N614NA

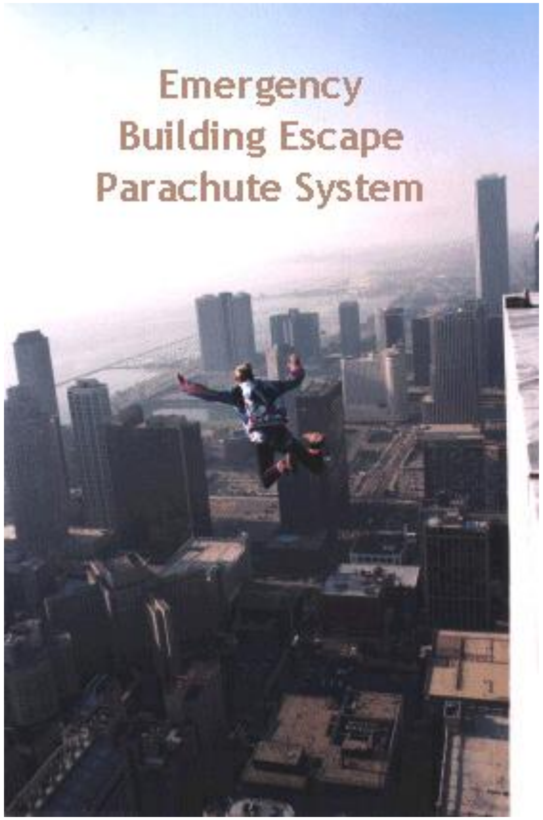

Shell

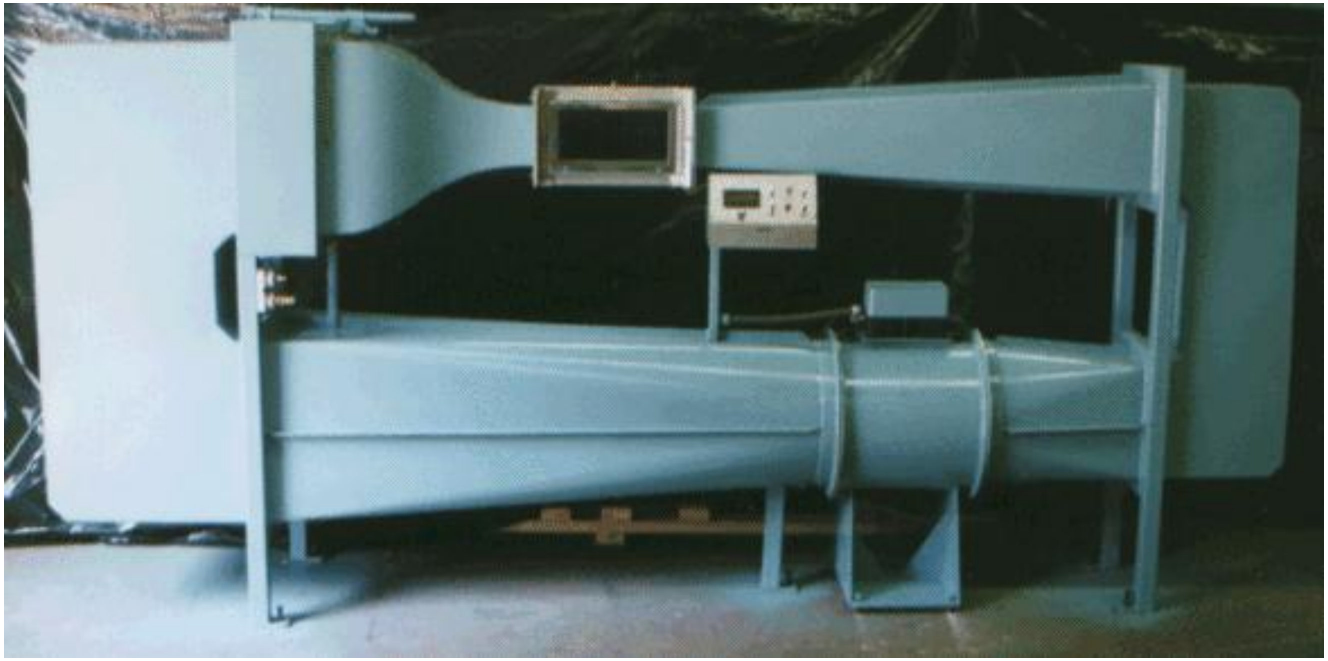

USA

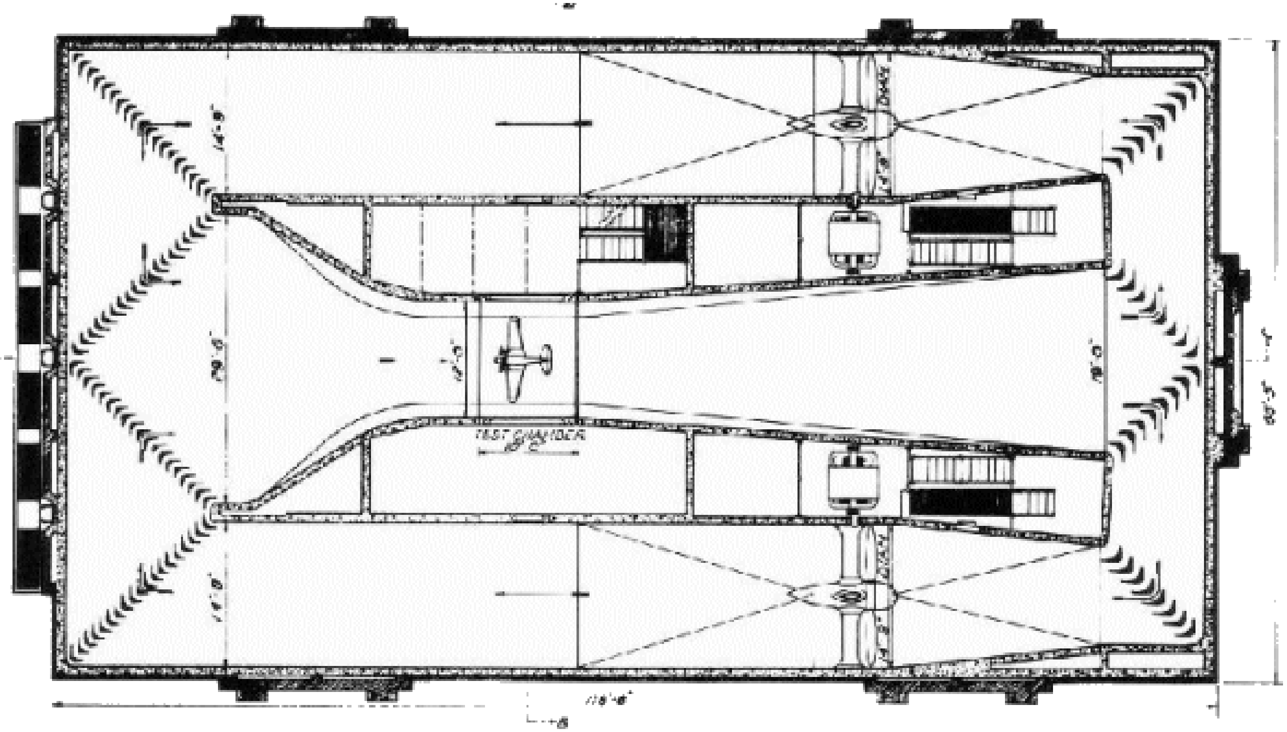

US
366
5-0

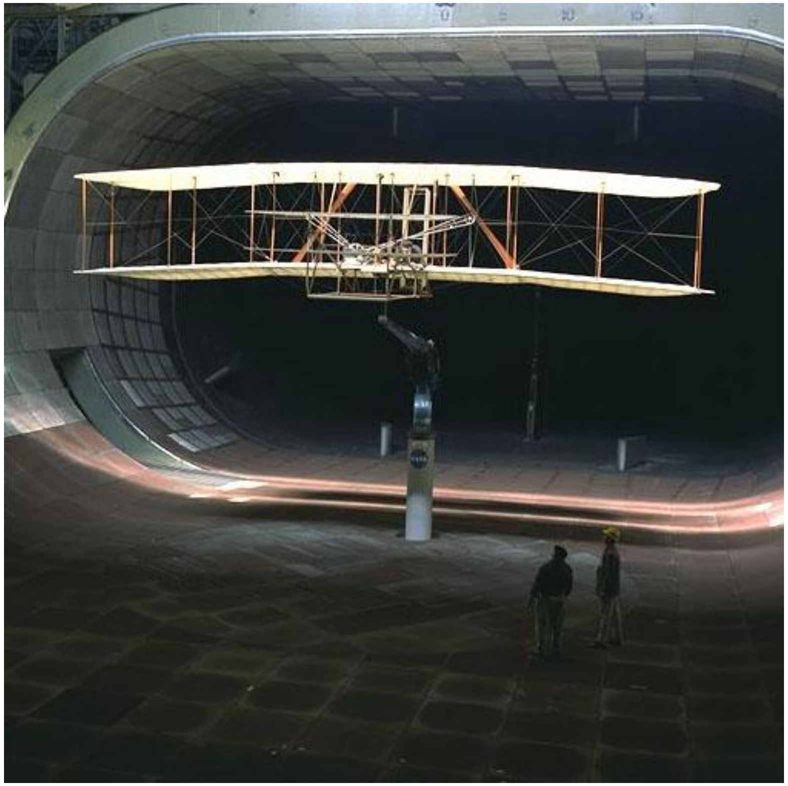

NORDTANK

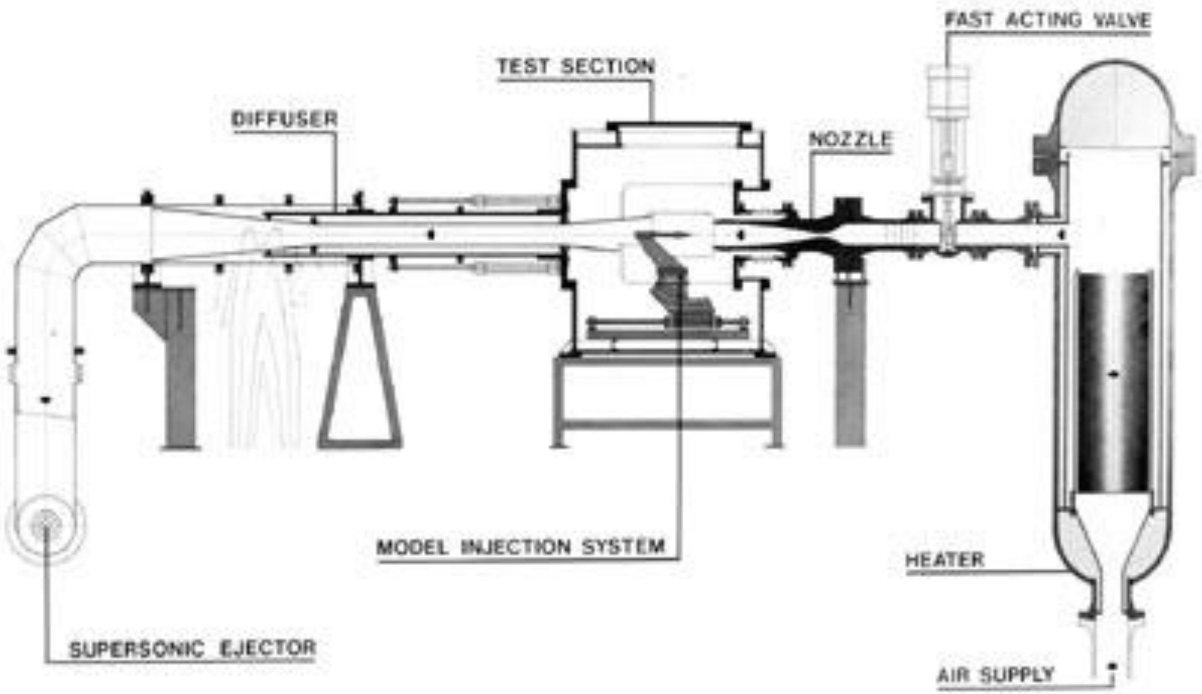

FISCHER
62


Emergency
Building Escape
Parachute System



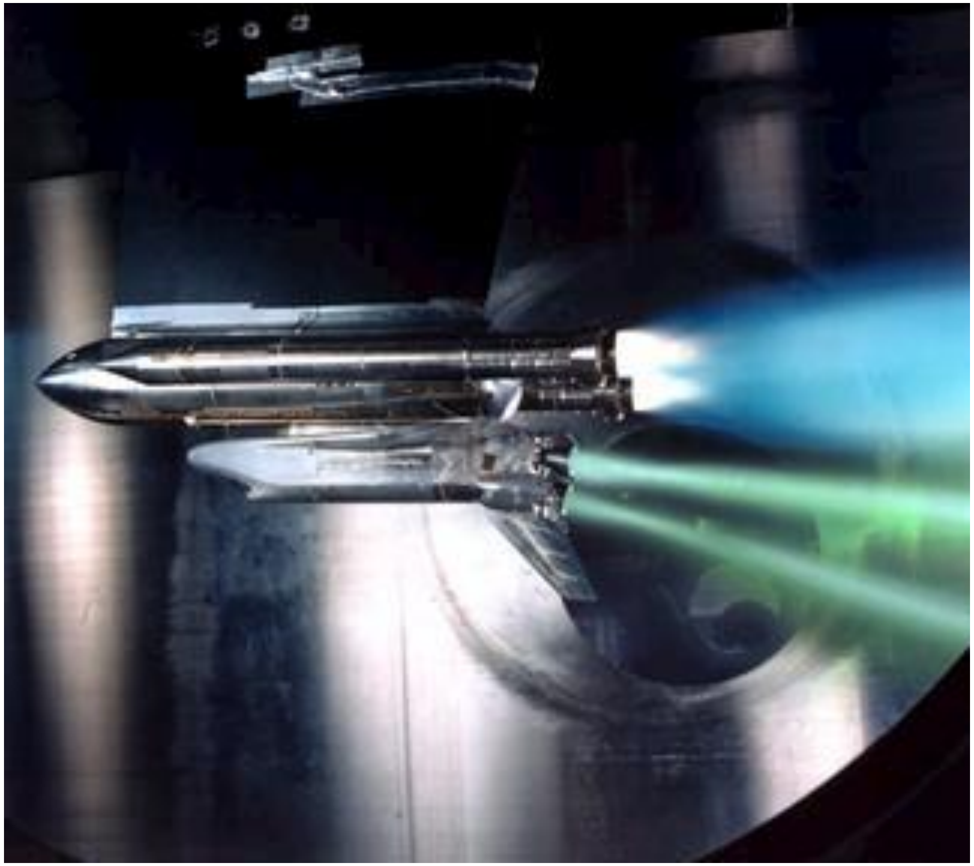

SECTIONAL PLAN THRU TUNNEL AXIS

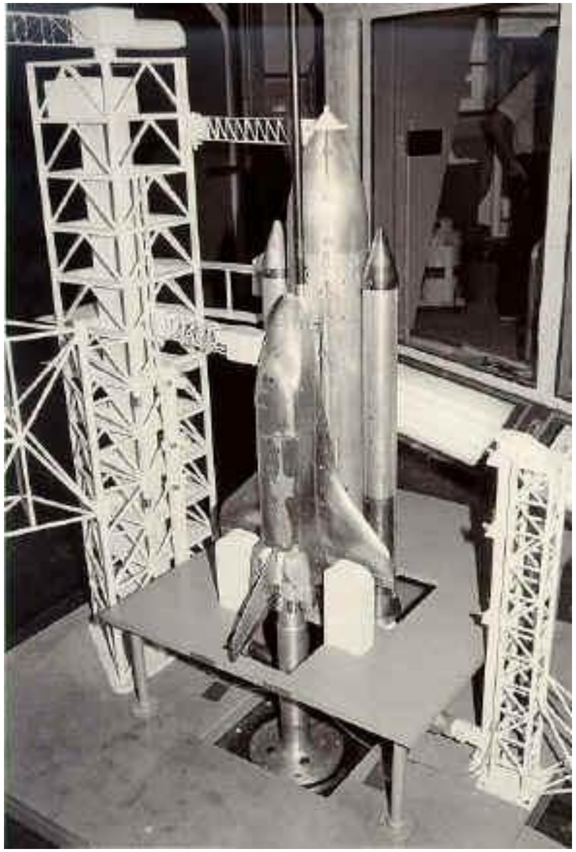



FAST ACTING VALVE
TEST SECTION
DIFFUSER
NOZZLE
MODEL INJECTION SYSTEM
HEATER
SUPERSONIC EJECTOR
AIR SUPPLY

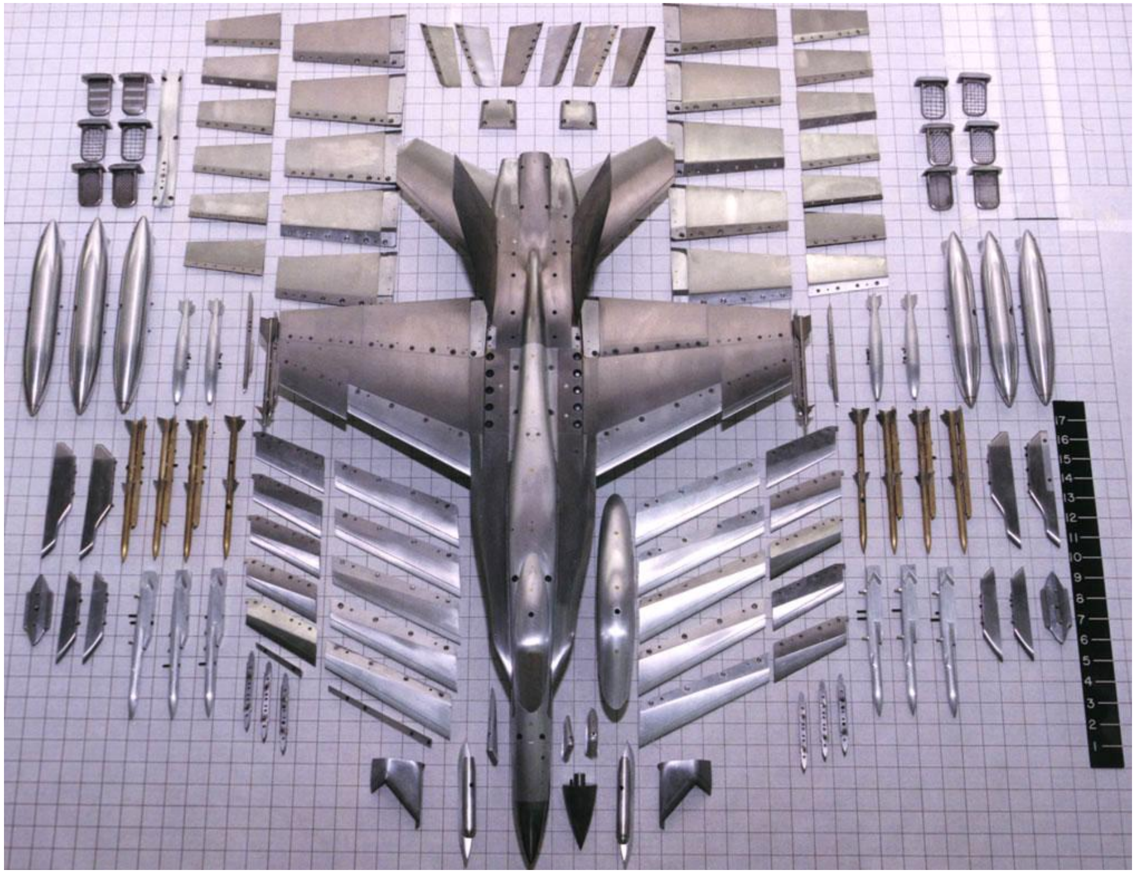
Unit-III

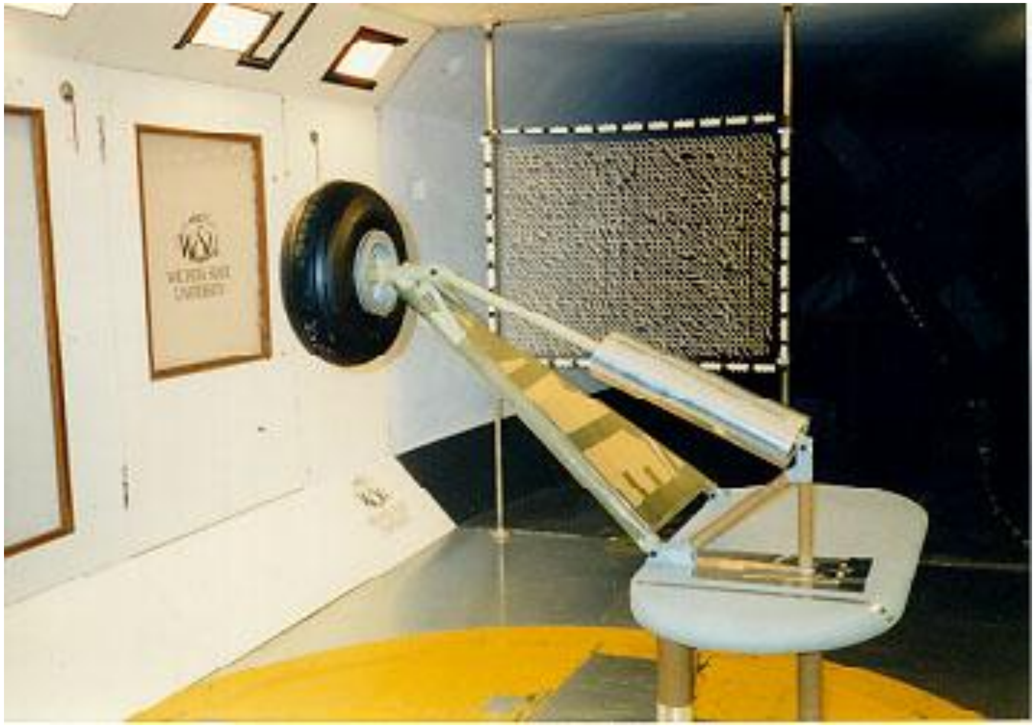
WIND TUNNEL BALANCE

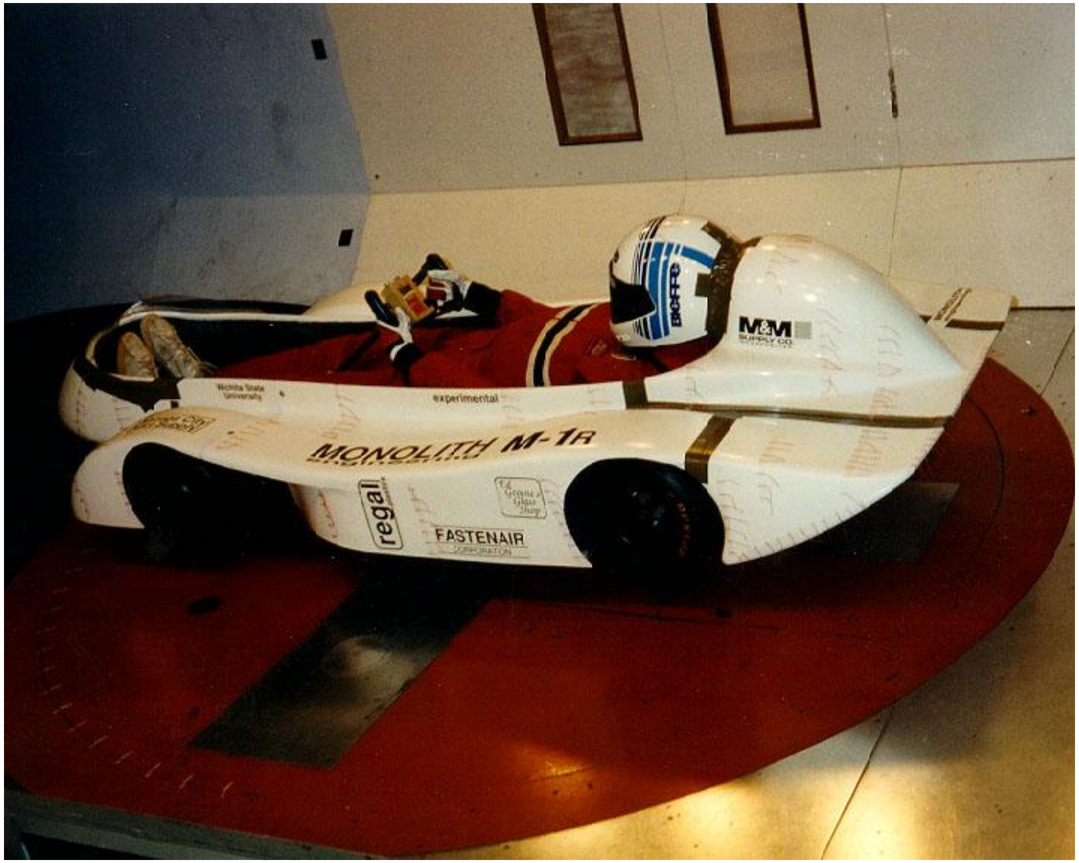


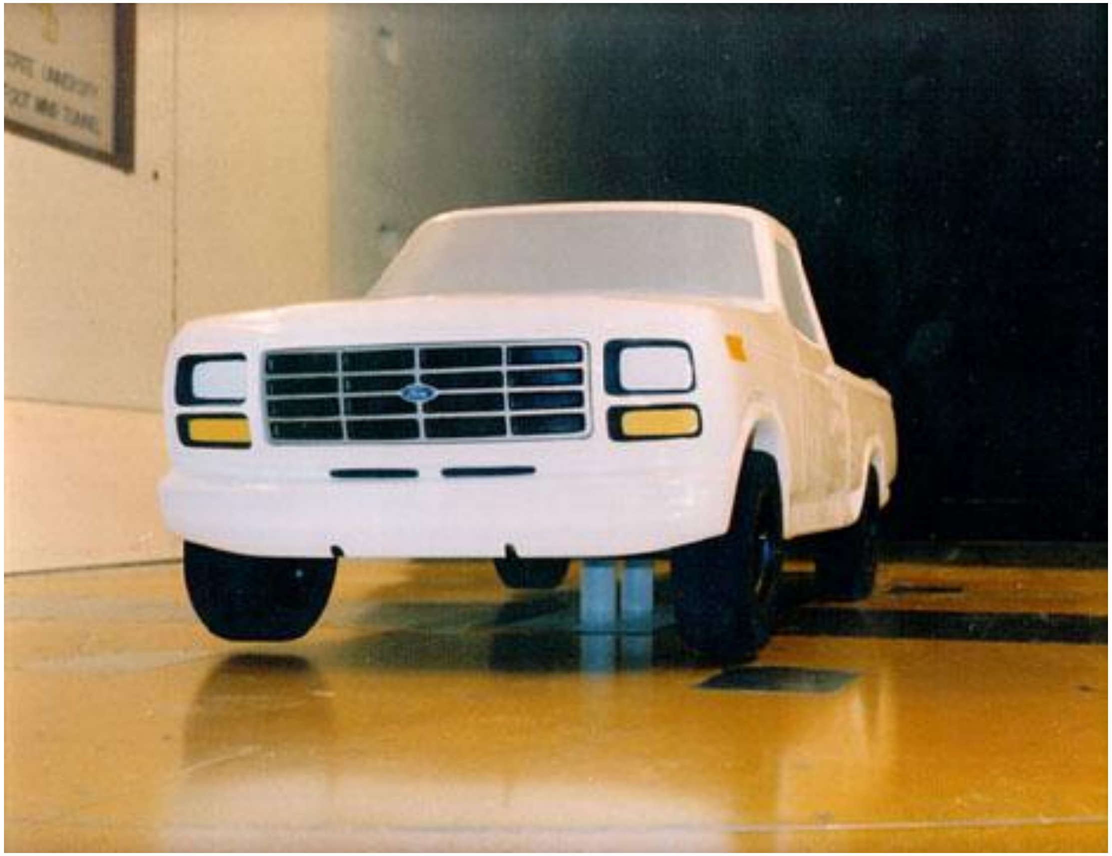



NASA






MONOLITH M-1R
experimental
regal
FASTENAIR
M&M

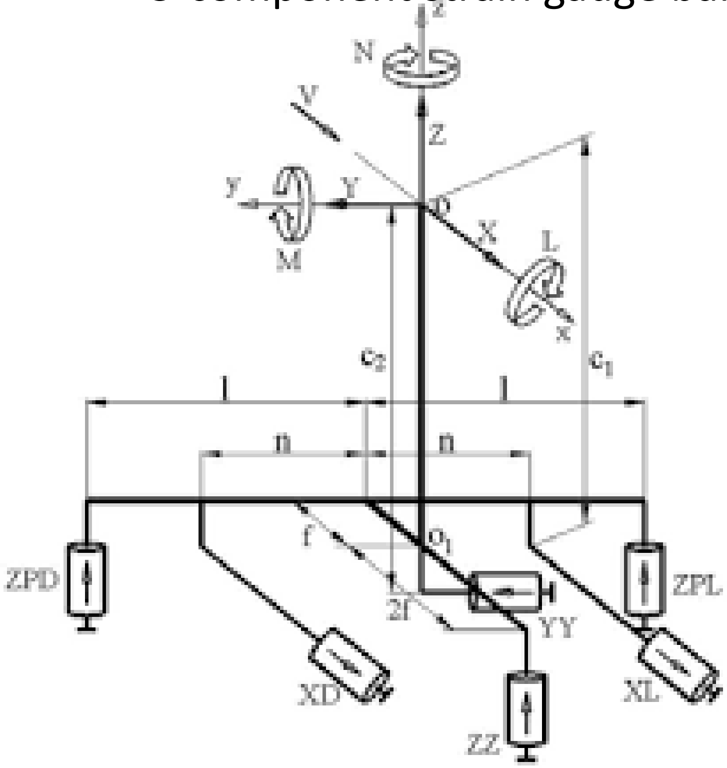


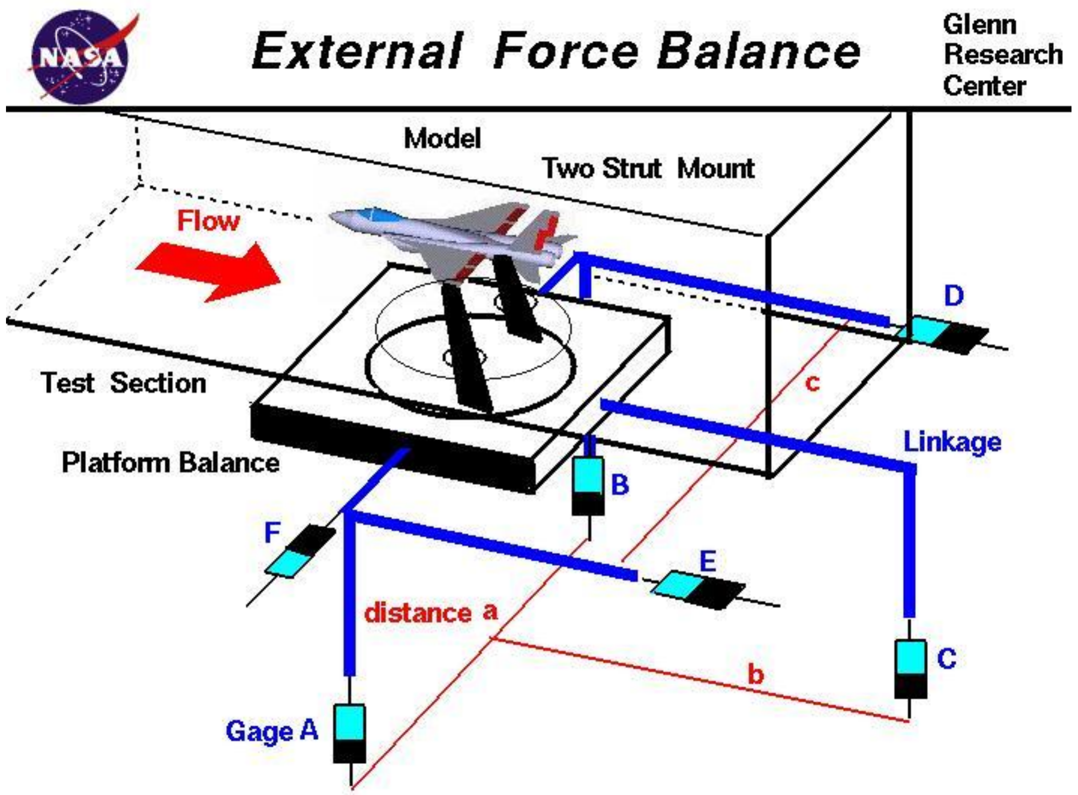

WICHITA STATE
UNIVERSITY

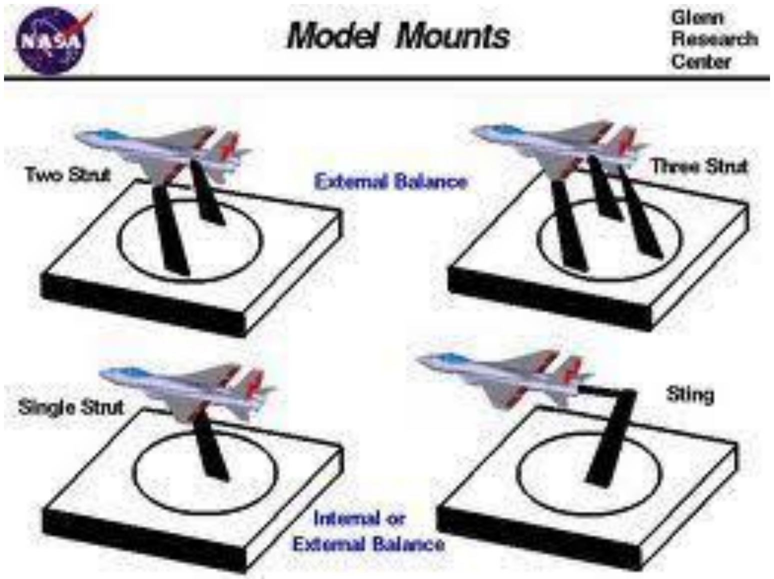

TAYLOR
USA


SIERRA
DESIGNS

3-component strain gauge balance

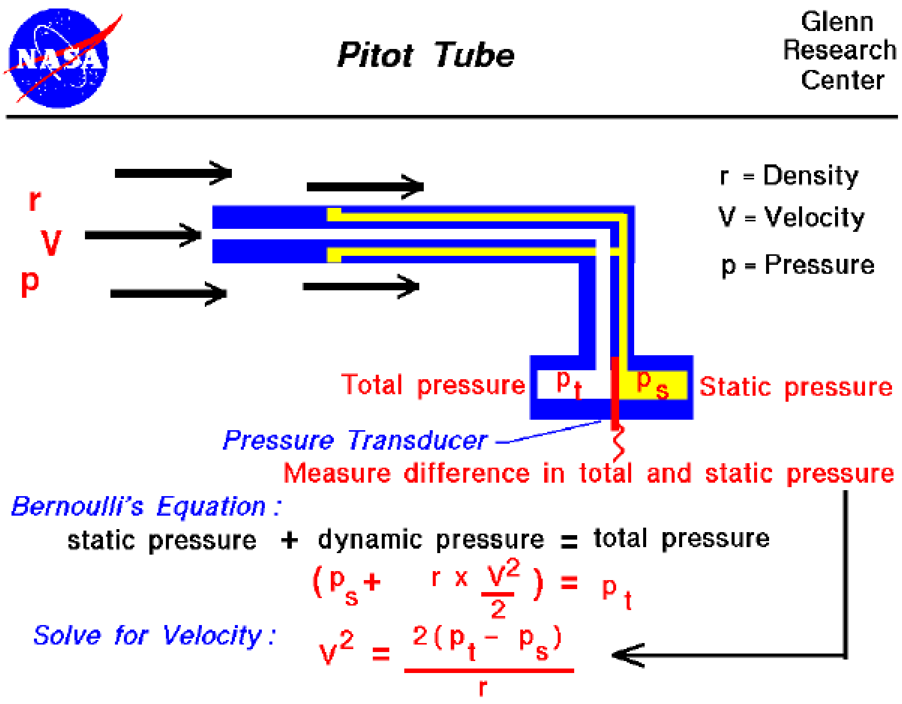


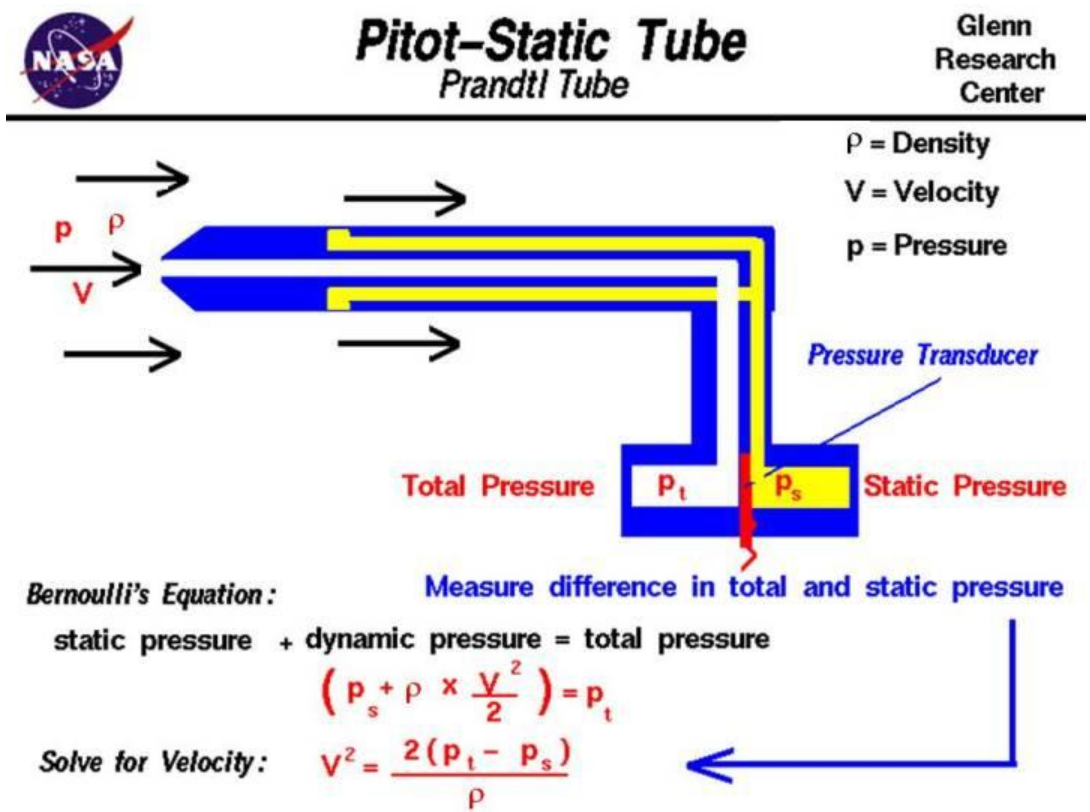

External Force Balance
Glenn Research Center
Model
Two Strut Mount
Flow
D
Test Section
c
Linkage
Platform Balance
B
F
E
distance a
C
b
Gage A


Model Mounts
Glenn
Research
Center
Two Strut
External Balance
Three Strut
Single Strut
Sting
Internal or
External Balance

Unit-IV

PRESSURE, VELOCITY & TEMPERATURE MEASUREMETNS

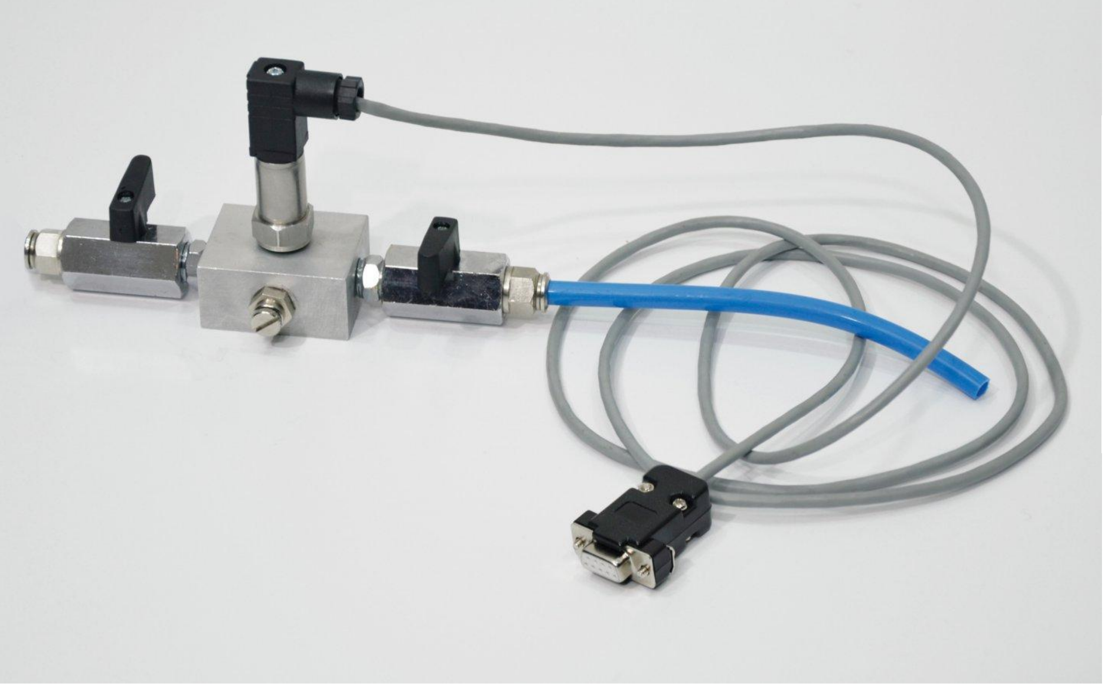

Pitot Tube
Glenn Research Center
r = Density
V = Velocity
p = Pressure
r
V
p
Total pressure
Static pressure
Pressure Transducer
Measure difference in total and static pressure
Bernoulli's Equation :
static pressure + dynamic pressure = total pressure
$(p_s + r \times \frac{V^2}{2}) = p_t$
Solve for Velocity :
$V^2 = \frac{2(p_t - p_s)}{r}$


NASA
Pitot–Static Tube
Prandtl Tube
Glenn Research Center
ρ = Density
V = Velocity
p = Pressure
p ρ
V
Pressure Transducer
Total Pressure
p_t
p_s
Static Pressure
Measure difference in total and static pressure
Bernoulli's Equation:
static pressure + dynamic pressure = total pressure
$\left(p_s + \rho \times \frac{V^2}{2} \right) = p_t$
Solve for Velocity:
$V^2 = \frac{2(p_t - p_s)}{\rho}$

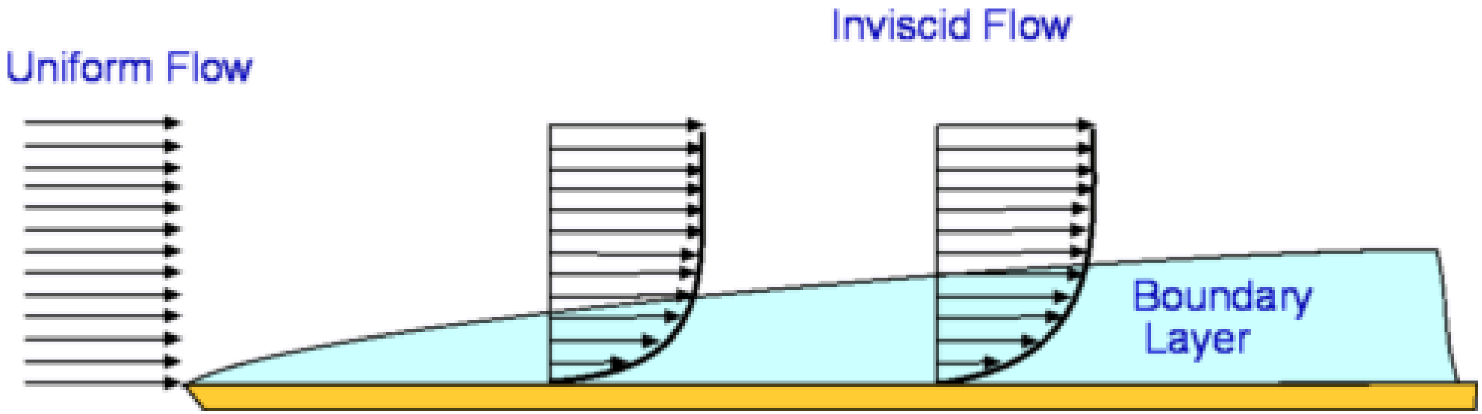
Unit-V

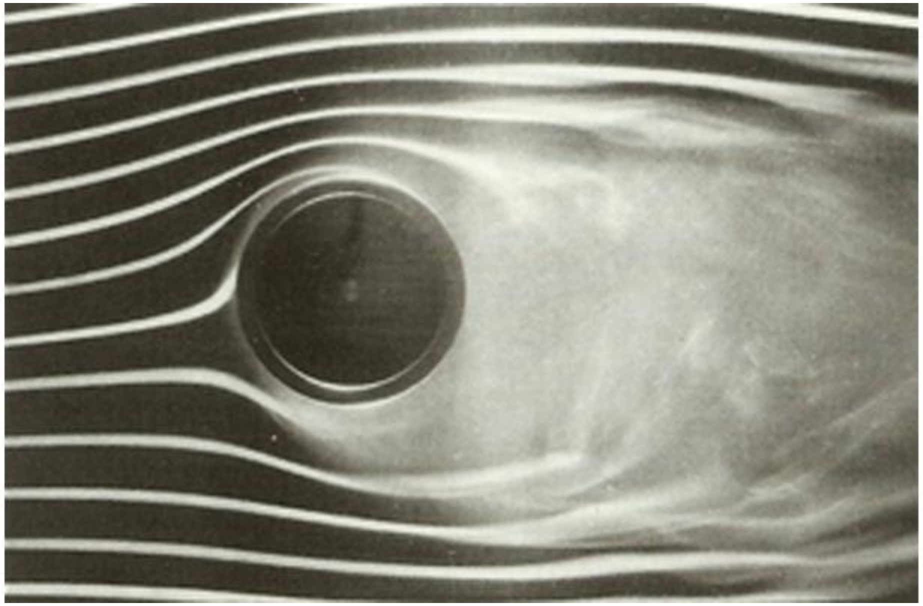
FLOW VISUALIZATION TECHNIQUES

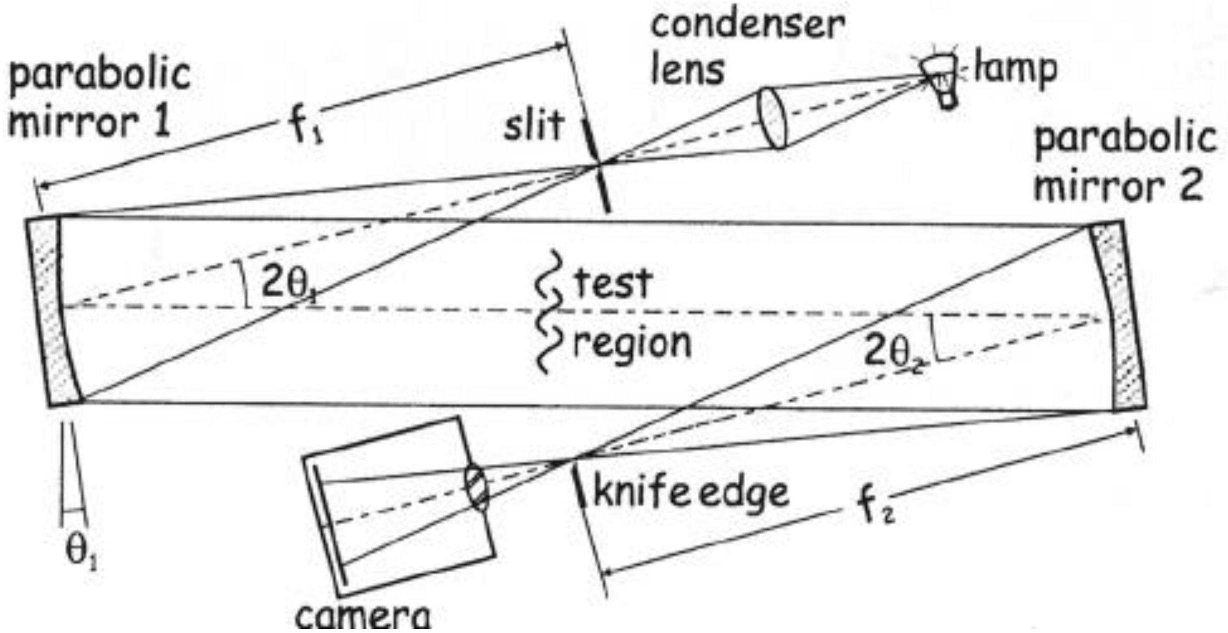
PRESSURE PROBES AND TRANSDUCERS



BOUNDARY LAYER PROFILE USING PITOT STATIC PROBE







parabolic
mirror 1
f1
slit
condenser
lens
lamp
parabolic
mirror 2
2θ1
test
region
2θ2
θ1
camera
knife edge
f2