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AGARD FLIGHT TEST INSTRUMENTATION SERIES VOLUME 2. IN-FLIGHT TEMPERATURE MEASUREMENTS

F. Trenkle, et al

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AGARD Flight Test Instrumentation Series
Volume 2
on
In-Flight Temperature Measurements

by

F.Trenkle and M.Reinhardt

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AGARDograph No.160 Vol.2

IN-FLIGHT TEMPERATURE MEASUREMENTS

by

F.Trenkle and M.Reinhardt

Volume 2

of the

AGARD FLIGHT TEST INSTRUMENTATION SERIES

Edited by

W.D.Mace and A.Pool

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PREFACE

Soon after its foundation in 1952, the Advisory Group for Aeronautical Research and Development recognized the need for a comprehensive publication on flight test techniques and the associated instrumentation. Under the direction of the AGARD Flight Test Panel (now the Flight Mechanics Panel), a Flight Test Manual was published in the years 1954 to 1956. The Manual was divided into four volumes: I. Performance, II. Stability and Control, III. Instrumentation Catalog, and IV. Instrumentation Systems.

Since then flight test instrumentation has developed rapidly in a broad field of sophisticated techniques. In view of this development the Flight Test Instrumentation Committee of the Flight Mechanics Panel was asked in 1968 to update Volumes III and IV of the Flight Test Manual. Upon the advice of the Committee, the Panel decided that Volume III would not be continued and that Volume IV would be replaced by a series of separately published monographs on selected subjects of flight test instrumentation: the AGARD Flight Test Instrumentation Series. The first volume of this Series gives a general introduction to the basic principles of flight test instrumentation engineering and is composed from contributions by several specialized authors. Each of the other volumes provides a more detailed treatise by a specialist on a selected instrumentation subject. Mr W.D.Mace and Mr A.Pool were willing to accept the responsibility of editing the Series, and Prof. D.Bosman assisted them in editing the introductory volume. AGARD was fortunate in finding competent editors and authors willing to contribute their knowledge and to spend considerable time in the preparation of this Series.

It is hoped that this Series will satisfy the existing need for specialized documentation in the field of flight test instrumentation and as such may promote a better understanding between the flight test engineer and the instrumentation and data processing specialists. Such understanding is essential for the efficient design and execution of flight test programs.

The efforts of the Flight Test Instrumentation Committee members and the assistance of the Flight Mechanics Panel in the preparation of this Series are greatly appreciated.

T.VAN OOSTEROM
Member of the Flight Mechanics Panel
Chairman of the Flight Test
Instrumentation Committee

CONTENTS

LIST OF SYMBOLS

Symbol	Abbreviation	Meaning
a		constant or factor
A		area
b		constant
c_p		specific heat at constant pressure
c_v		specific heat at constant volume
C		heat capacity
d		diameter
d_b		contacting diameter
d_s		diameter of solid body
D_s		thickness of boundary layer of flow
D_T		thickness of boundary layer of temperature
E	e.m.f.	electrical voltage
E		error
E_α		attitude error
E_C		conduction error
E_{CAL}		calibration error
E_{DH}		deicing heat error
E_I		instrument error
E_L		lag error
E_M		meteorological error
E_P		position error
E_{PL}		probe location error
E_R		radiation error
E_{RO}		read-out error
E_{SC}		scale error
E_{SH}		self-heating error
E_V		velocity error
E_W		electric lead error
$f(\ldots)$		function of
F		damping ratio
g		acceleration due to gravity
G		thermal conductance
G		dissipation constant
h		heat transfer coefficient
h_c		heat transfer coefficient, forced convection
h_D		heat transfer coefficient, conduction
h_n		heat transfer coefficient, natural convection
h_R		heat transfer coefficient, radiation

I	electrical current
I_d	current in the diagonal branch of a bridge
J	mechanical equivalent of heat
k	thermal conductivity
k_f	thermal conductivity of a fluid
k_b	thermal conductivity of contacting area
K	geometric influence factor
K_a	geometric influence factor for attitude error
K_C	geometric influence factor for conduction error
K_{DH}	geometric influence factor for deicing heat error
K_K	geometric influence factor for construction
K_L	geometric influence factor for lag error
K_n	geometric influence factor for natural convection
K_{PL}	geometric influence factor for probe location error
K_R	geometric influence factor for radiation error
K_{SH}	geometric influence factor for self-heating error
L	length
m	mass
$(\ )^m$	exponent of Reynolds number
M	Mach number
$(\ )^n$	exponent, variable
Ni	nickel
Nu	Nusselt number
p	static pressure
p_T	total pressure
P	power
P_{DH}	power of deicing heater
Pr	Prandtl number
Pt	platinum
Q	heat (quantity)
r	recovery factor
r_1	instrument (probe) recovery factor, subsonic
r'	overall recovery factor, supersonic
R	gas constant
R	resistance (electrical or thermal)
R1, R2 . . .	bridge resistances
R_{25} (R_{100})	probe resistance at 25°C (100°C)
R_{bl}	resistance of boundary layer
R_C	calibration resistance
R_d	load resistance of the bridge output (diagonal branch)
Re	Reynolds number

R_p		resistance in parallel connection
R_s		resistance in series connection
RL		lead resistance
R_x		variable probe resistance
s		sensitivity
s		surface
s_b		contacting surface
SH		self-heating
t		time
T	SAT	static temperature
T_c		calibrated temperature
T_i		indicated temperature
T_{ic}		basic temperature
T_{id}		corrected temperature
$T_{[illegible]}$		local temperature
$T_{[illegible]}$		measured temperature
T_{Mean}		mean temperature
T_A		temperature before a change
T_B		temperature after a change
$T_{[illegible]}$		probe temperature
T_r		recovery temperature
T_T	TAT	total air temperature
$T_{1[illegible]}$		calibrated total air temperature ($\equiv T_T$)
T_{Ti}	IAT	indicated total (air) temperature
T_{Tic}	BAT	basic total (air) temperature
T_{Ticl}		corrected total (air) temperature
T_w		wall temperature
ΔT_k		kinetic (= full adiabatic)temperature rise
ΔT_{kd}		measured temperature rise in dry air
ΔT_{kr}		recovered temperature rise
ΔT_{kf}		temperature rise by friction
ΔT_{kfl}		temperature rise by friction, laminar flow
ΔT_{kft}		temperature rise by friction, turbulent flow
ΔT_{kw}		measured temperature rise in air with liquid water content
ΔT_{ic}		instrument error correction
ΔT_{icl}		lag error correction
ΔT_{pc}		position error correction
ΔT_{kc}		compression correction
U		voltage
U_B		excitation voltage of a bridge

U_A		output voltage of a bridge
v		volume
V		velocity
V_t	TAS	true air speed
w_n		natural frequency
W		liquid water influence factor
x		distance or length
α		temperature coefficient
α	AoA	angle of attack
β		angle of side slip
γ		ratio of specific heats c_p/c_v
ε		emissivity
η		recovery error
μ		dynamic viscosity coefficient
ρ		density
σ		radiation constant
τ		time constant
ϕ		mass velocity or mass flow rate

International system of units (SI) is normally used, where not otherwise indicated. For measurement units see Table 5, page 29.

IN-FLIGHT TEMPERATURE MEASUREMENTS

by

Fritz Trenkle and Manfred Reinhardt
Deutsche Forschungs - und Versuchsanstalt für Luft - und Raumfahrt E.V.
Institut für Physik der Atmosphäre
Oberpfaffenhofen
Germany

SUMMARY

This publication is intended to give practical assistance to all instrumentation and test engineers working in the field of temperature measurements in aircraft at Mach numbers up to 2.3 and altitudes up to 80,000 feet.

After a general discussion of the requirements of aircraft temperature measurements, and the available temperature sensing technology, the detailed techniques of using resistance probes and thermocouples, as well as the associated electrical leads, circuits, and indicators, are explained. A discussion of heat transfer processes, primarily between moving fluids and solids, includes terminology, the systematics of temperature measurements, and the concept of total temperature as the main operational parameter. An extensive section deals with errors in temperature measurements, as functions of various parameters, in gases, liquids and solids. Typical laboratory and in-flight calibration techniques for thermometers are described, followed by discussions of data handling, error analysis, and the limits of present methods.

1. INTRODUCTION

1.1 Aim of this Volume

Temperatures to be measured in and on aircraft presently range between approximately 200 and 1,500°K, and in extreme cases, the limits extend to 20°K and 2,000°K. Extremely different requirements are made on the measurement devices depending on the type of measurement (outside air, exhaust, fuel, etc.). The large number of temperature measurements normally made in a modern production aircraft is further increased for flight test programs, and there are increased requirements for measurement accuracy, as well as additional demands, e.g. for matching with recorders, indicators, and signal conditioners. The information required to make more precise measurements in aircraft is usually not found in the regular textbooks on temperature measurement, but have to be gleaned from different individual publications, and are, therefore, not readily at the disposal of the engineer who suddenly finds himself faced with a new measurement problem. In the more popular handbooks for flight testing, the measurement of the temperature of the outside air is briefly discussed. This discussion is, however, often based on the level of technology in 1945 and the flight regime of the typical piston engine aircraft of the day. The possibilities opened up by the development of modern equipment are not often explored. Special problems are usually ignored, for example, the case of extremely slow-flying aircraft (VTOL) or very fast, high-altitude aircraft, and the measurement of the temperature of the air as it enters a jet engine.

The following survey should serve the test engineer as well as the instrumentation engineer as a practical handbook which can be used to choose the correct probes, indicators and signal conditioners, ascertain their optimum installation, determine their accuracy and correctly interpret the measurements obtained. Special importance is attached to directly applicable graphic representations in order to reduce computational work. The examples also show the extent to which data, that are considered generally valid in normal measurement techniques, can change when measurements are made in air or in other gases. In the discussion of the processes of heat transfer and the effect of the individual parameters, the number of problems involved often necessitated an extremely simplified representation, appropriate for the given measurement purposes, to maintain clarity and because of the limited scope of this paper. Thus, this paper is intended to be a supplement to the references and manuals which are now available to the technician. Additional information can be found in the bibliographic list of primary sources.

For measuring the outside air temperature, the authors have limited themselves to the range of altitudes and speeds of modern production aircraft. These limits were set at an altitude of 80,000 feet and a speed of approximately Mach 2.3. The results of measurements made on experimental and test aircraft which exceeded these limits, insofar as the results have been published, do not enable any generally valid conclusions to be made. But even in the flight regime selected, there are still zones for which there are no valid measurement results. This is especially true at relatively low speeds at high altitudes, in the near-sonic airspeed range at low altitudes with high atmospheric temperatures combined with high humidity, and for flights in clouds and heavy rain.

1.2 Temperature Measurements in Aircraft

Figure 1 gives a survey of the different types of temperature measurements in aircraft as well as the corresponding measurement range.

One of the most important values is the undisturbed outside or static air temperature (SAT). For reasons of measurement accuracy, the increased total air temperature (TAT), which depends on the corresponding Mach number, is almost always determined instead of the static air temperature except on very slow-flying types of aircraft. If Mach number is determined, TAT is readily converted to static air temperature. In many aircraft, the conversion is done automatically in a central air data computer, which uses the temperature value and the Mach number to yield the true airspeed (TAS) and other additional values, e.g. air density, which are necessary for the automatic pilot and the weapons system.

Other important temperature measurements are made in engines to insure that limiting temperatures are not exceeded. For piston engines it is necessary to measure and observe the oil temperature, the cylinder head temperature, the coolant temperature, and if needed, the carburetor temperature or the air inlet temperature for injection engines. For jet engines the compressor inlet temperature determines important operating parameters* so that it must be indicated directly as well as fed to the automatic fuel control system. Accurate total temperature measurements of the exhaust gas temperature are also required. There is a logarithmic relation between this temperature and the life of the engine. The optimum engine output can, however, only be reached at or near the prescribed temperature, so that the exhaust temperature must also be measured and indicated continuously. On supersonic aircraft it is also fed into an automatic nozzle area control system.

Additionally, the measurement and indication of the fuel temperature is often necessary if, for example, aerodynamic heating of the aircraft skin influences the fuel temperature.

For flight test purposes, most of the previously mentioned measurements must be made with increased accuracy and often at more measuring points than normally required in order to make unconditional statements on behavior over the entire range of flight. In addition, there are at times a series of other measurements, such as measurement of the liquid oxygen temperature (LO_2Temp.) in the oxygen apparatus or the liquid nitrogen temperature (LN_2Temp.) in infrared equipment, in order to make sure that the systems function as planned under all possible flight conditions.

In the same way, several surface temperatures must be measured on the brakes, and on various structural components that are exposed to the thermal radiation of the engine or the exhaust stream, as well as aerodynamic heating, at different points on the aircraft skin and interior structure. The fuselage of the aircraft runs through a temperature cycle in every flight, where certain components heat up and then cool faster than adjacent components, so that there are differences in expansion and corresponding material stresses.

Temperature measurements in aircraft are made at widely different locations (outside air, fuel, etc.); they encompass very different ranges and are subject to very different accuracy requirements. Figures 2 and 3 show examples of typical temperature curves for a supersonic aircraft as a function of altitude and flight time.

1.3 Thermometers Used in Aircraft

The temperatures mentioned in the previous section are measured using, to a certain extent, various measuring devices, which for the sake of simplicity will be called thermometers in the rest of this volume. Since the points at which the measurements and readings are taken are widely separated in aircraft, a typical thermometer usually consists of a temperature probe attached at a measuring point,

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* E.g. for supersonic aircraft, the value of the maximum permissible speed.

and a measurement indicator in the cockpit. In addition, there may also be, if necessary, a computer, data converter, or signal conditioner to feed signals to recording or telemetry systems.

The thermometers used in aircraft can be roughly divided according to the way they are used:

(a) Bimetallic thermometer, usually used in light aircraft to measure the outside and cabin air temperature.

(b) Liquid-in-glass thermometer, e.g. for meteorological purposes: dry-bulb and wet-bulb temperature.

(c) Liquid-in-metal thermometer, as an integrated component of older fuel control units for jet engines and simple true airspeed transmitters.

(d) Resistance thermometer, the most accurate and widely used thermometer in aircraft for all types of measurements in the range of temperatures between -220°C and approximately +300°C, although some resistance thermometers are accurate up to +800°C.

(e) Thermoelectric thermometer (thermo-couple), generally used for aircraft measurements in the range of temperatures between +200°C and approximately +1,500°C, although some thermoelectric thermometers are accurate as low as -175°C.

(f) Thermometers for special purposes (based on acoustic velocity measurements, infrared radiometry, color change, or change in hardness of a material), almost exclusively used for measurements on the ground, e.g. bench tests for jet engines.

The thermometers most generally used in aircraft are listed in Table 1. The magnitude of the errors to be expected for a specific thermometer depends on factors such as temperature range and choice of thermometer materials.

TABLE 1. TYPES OF THERMOMETERS USED IN AIRCRAFT

Thermometer Type	Temperature Range °C			Limit of Error
Bimetallic	-60	to	+400	1 to 3% of the range
Liquid-in-glass (Special applications)	-58	to >	+60	0.2 to 1°C
Liquid-in-metal	-35	to	+500	1 to 2% of the range
Resistance	-220	to	+300	0.2 to 5.0°C
Thermoelectric	+200	to >	+1500	1 to 10°C

In bimetallic thermometers, the differential thermal expansion of two metals is usually made to move a pointer positioned over a circular scale. For flight testing, the advantage of being relatively inexpensive is outweighed by serious disadvantages such as: its use is restricted to measuring the air temperature at low speeds, its response to temperature changes is slow, and it is impractical to connect a satisfactory recording device for use in aircraft. This is also true for the most part for liquid-in-glass thermometers where a thin column of liquid is positioned in front of a straight line scale. In addition, they must be protected against shock and vibration.

Liquid-in-metal thermometers have not been used in aircraft as isolated devices but only as components of automatic fuel controls for jet engines and of simple TAS transmitters. In this case, the temperature measurements are obtained by virtue of the expansion of a liquid enclosed in a metal tube used as the temperature probe. The resulting increase in pressure is transmitted by a thermally insulated extension of the tube to a pressure sensor. The type of pressure sensor used is selected to give the kind of output desired (voltage, indicator deflection, etc.).

Electric thermometers are the ones most generally used. In the case of a resistance thermometer, a temperature sensitive resistor (called a "resistance probe" for short) is mounted at the measuring point, and usually forms one leg of a bridge circuit. Bridge unbalance, which is a function of temperature, is sensed electrically. The thermoelectric thermometer, commonly called a thermo-couple, is simply a junction of two wires of different metals or metallic alloys. A voltage is produced that depends on the temperature difference between the junction at the measuring point and the junction of the other ends of the wires which is maintained at a reference temperature. This electrical signal is then converted, using a sensitive ammeter, or by using a reference voltage and a compensator, into the corresponding temperature value. Since both types of electric thermometers can be used for practically all possible problems of temperature measurement in aircraft with a great degree of accuracy, flexibility, and reliability, and since the output can be easily prepared for recording and/or telemetry, they are used almost exclusively in flight tests. For these reasons, only these two types will be discussed in greater detail in the following sections.

The special-purpose thermometers include the temperature sensitive materials whose color depends upon the highest temperature to which it has been exposed. There are even certain types which go through a multiple color change and can indicate up to four different temperature levels (cf. Section 6.1.4). Mechanical temperature indicators are also used on occasion in engine tests. In this case, the change in hardness of a material is a measure of the maximum temperature reached. Optical temperature measurements are often made, in research and development tests, of the hot gases in combustion chambers and exhaust streams. They have the advantages that no probes are mounted in the gas stream and hence gas flow and temperature distributions are not disturbed, the possibility of thermometer damage and parts entering an engine is minimized, and almost inertialess recordings are feasible. Some of the instruments can be used to obtain a measurement of the temperature distribution in the gas stream. However, this technique has not yet been extended to in-flight applications.

There is a close relationship between thermometers and hot-wire anemometers which, in addition to measuring the speed of the gas stream, can also be used at times to measure the temperature. This measurement principle is based on the convective heat loss of an electrically heated wire mounted in a gas stream. There are, however, many problems when using them as thermometers. Acoustic thermometers must also be mentioned. These thermometers measure sound velocity as a function of the temperature. There is little information on their use in aircraft.

Novel temperature probes such as specially-cut quartz crystals, temperature sensitive condensors, etc. are often used in temperature measurements. Until now, however, no models of these devices have been produced that are suitable for the measurement conditions in aircraft. The successful trials of a Fluidic-TAT-probe up to Mach 6.7 in an X-15 aircraft are reported. It can be assumed that in the next few years, "digital probes" of this sort, which do not produce a genuine digital signal, but rather a variable frequency which can easily be converted to digital form, will come into use.

Almost all temperature measurements in aircraft are relative measurements, i.e. the indicators give the temperature of an object relative to the melting point of ice measured in ^{o}C, and seldom the absolute temperature in ^{o}K. In special flight tests two temperature probes of the same type are mounted at different measuring points and connected differentially to the same measuring instrument, thereby giving the difference between two temperatures ΔT in °C. The rate of change of temperature may also be measured, in which case, two temperature probes with different time constants are connected differentially at the same measuring point and to the same measuring instrument to indicate the rate that the temperature is increasing or decreasing ($\Delta T/\Delta t$ in $\pm$ ^{o}C/min).

2. RESISTANCE TEMPERATURE MEASUREMENTS

2.1 Resistance Probes

Metals, i.e. electrical conductors, are crystal structures that are composed of the corresponding metal ions, i.e. of atoms whose valence electrons* have been freed by Brownian movement. When a voltage is applied, a directional flow of free electrons is superimposed on this random motion. The electrons collide with each other and with ions of the crystal lattice (which oscillate about an equilibrium position) and thereby impede the flow, i.e. generate electrical resistance. As the conductor heats up, the oscillations of the electrons and ions increase, i.e. resistance increases with temperature. The increase in resistance in ferroelectric ceramics and the so-called "cold conductors" is much greater than in metals.

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* Valence electron = the electron in the outermost electron shell, which is bound only very loosely to the atom.

In some types of the so-called "semiconductors", the valence electrons are bound more strongly to the atoms and are freed only at higher temperatures so that the flow increases with the temperature (when voltage is applied), since more electrons can join the flow of the current. In other words, the resistance decreases as the temperature increases. These semiconductors which are used as temperature sensitive resistors are, therefore, also called "hot conductors".

In general, both phenomena mentioned above (increase or decrease in resistance with increase in temperature) take place simultaneously, but generally one of the processes is predominant. However, certain metal alloys, the so-called "constant resistance alloys", can be produced which exhibit no appreciable change of resistance with temperature over a rather wide temperature range.

The useful temperature range of a resistance probe is determined by the magnitude of the temperature sensitivity, chemical reactions (heavy oxidation or reduction), permanent changes in the crystal structure, etc. At very high temperatures, it is important to note that all insulators (e.g. glass above 550°C) become electric conductors (due to the release of all the valence electrons), and all materials lose their impermeability to gas (important for shield tubes).

The slope of the R/T curve (resistance over temperature) can be expressed as a change in resistance ΔR per change in temperature ΔT of 1°C, or for purposes of comparison as the temperature coefficient α, i.e. as the relative change in resistance $\Delta R/R_x$ per °C, according to the equations:

$$\Delta R = R_x \cdot \alpha \cdot \Delta T \quad \text{and} \tag{1}$$

$$\alpha = \Delta R/(R_x \cdot \Delta T) \tag{2}$$

In practice, α (multiplied by 100) is usually expressed in %/°C, and α is positive if resistance increases with temperature, but negative if resistance decreases with temperature. Although α is usually treated as a constant, in reality it bears a non-linear relation to temperature, and a given value of α is exact at only one value of R_x.

It is characteristic of temperature sensitive resistors that they are passive elements, and that a current must be sent through them (e.g. in a bridge circuit) which can perceptibly heat up the resistor through the "Joule effect", causing a self-heating error. The definition of the "sensitivity" of a temperature probe as the relationship between the variation in the output voltage ΔE for a variation in temperature ΔT of 1°C for purposes of comparison with active temperature probes (thermo-couples, which generate voltage of themselves) is very misleading. This applies especially when measuring in media with a poor thermal conductivity (air or other gases, etc.), and when not all the conditions are named, such as permissible self-heating error, magnitude of the probe resistance, circuit design, type and velocity of the medium, etc.

In the following, R_x generally expresses the resistance value of an element at any temperature T_x, and R_{x1} is the value at a temperature T_1, R_{x2} at T_2, but R_o is the value at 0°C, R_{25} the value at +25°C, etc.

2.1.1 Elements with Positive Temperature Coefficients

Temperature probes of pure platinum wire were selected as the international standard for temperature measurements between the boiling point of liquid oxygen (-182.97°C) and the melting point of antimony (+630.5°C), since the resistance-to-temperature ratio of pure platinum wire in an annealed and strain-free state is especially stable and reproducible. The temperature coefficient increases rapidly from zero at approximately 10°K (-263°C), reaches a maximum (0.42%/°C) at approximately 30°K (-243°C) and then gradually decreases as the temperature is increased (Figures 4 and 5). The upper temperature limit for continuous operation is approximately 800°C (for short-term applications, 1,100°C). Tabular data for the resistance values of coiled platinum wire elements as a function of temperature are found, e.g. for elements where R_o = 100 ohms, in DIN 43760 (Reference 70). For elements where R_o = 500 ohms, the values for R_o = 100 ohms should be multiplied by 5; where R_o = 50 ohm, the values should be halved, etc. Platinum with only a relatively high degree of purity is often used in various aeronautical devices, and especially in total temperature probes, in order to obtain minor variations in R/T values; these are set forth in MIL-P-25726 B ASG and MIL-P-27723A. Platinum probes for aircraft are usually manufactured with a resistance R_o of 50 or 500 ohms in the U.S.A. However, other values are also used, e.g. 100 ohms in Germany and France, and 200 or 700 ohms in Sweden. The deviations from the corresponding standard R/T curves are generally less than $\pm 1^{\circ}C$ (cf. also Section 5.2.4).

In special cases, e.g. when measuring surface temperatures, elements are also used in the form of coated platinum foil. The R/T curve is less predictable for platinum foil, but is dependent on the thickness of the foil and must be found by individual calibrations (Reference 71).

Temperature probes made of nickel can be used in the temperature range from -190 to +180°C. It is less expensive than platinum and has a somewhat higher temperature coefficient which increases as the temperature is raised, but is also less stable. Above +180°C it changes its internal structure as it approaches the Curie point, so that its R/T curve can no longer be reproduced. Tables giving the data on resistance values R_x as a function of temperature can be found in DIN 43760 (Reference 70) where R_o = 100 ohms or in MIL-B-7258 where R_o = 1,200 ohms. Probes for which R_o is listed as 90.38 ohms are actually composed of a temperature sensitive element for which R_o is approximately 60 ohms and a series,30.38 ohm resistor housed in the base of the probe. The resulting R/T curves differ from those of the other probes for this reason and is given in MIL-B-7990A (Figures 4 and 6). For the same reason, approximately 30 ohms must be subtracted when determining the Joule heating in this type of probe. Other resistance values, e.g. 90 ohms, can be found in use in the UK. Since the temperature coefficient of nickel increases with temperature, the R/T curve can be made more linear by connecting a resistor in parallel with R_x (cf. examples in Figure 6). A shunt resistor approximately 3.6 times greater than the average value of R_x for the range provides reasonable linearity but appreciably decreases the slope of the characteristic curve. Another resistor in series with the parallel combination is often necessary for bridge design purposes, and further reduces the slope. The reduced slope of the corrected curve naturally must be taken into consideration when measurements are made.

Copper of all the known metals has the temperature coefficient with the best linearity over an extremely wide temperature range, however, it is not used for temperature probes in aircraft for various reasons. Its temperature characteristics can, unfortunately, result in variable line resistance in the probe conductors,this is especially troublesome in probes with low resistance.

Silicon semiconductor elements are available that have a higher positive temperature coefficient than metals and are suited for a temperature range from approximately -70 to +176°C. The variation in α for the individual types is rather broad, however. Their R/T curve is practically linear over the given range (Figure 7). There is as yet no data on the long-term stability of the material.

Occasionally, elements of a ferroelectric ceramic, the so-called cold conductors, also called PTC resistors*, are used. They consist primarily of sintered barium titanate with metallic oxides and have a very high temperature coefficient, depending on their composition, over narrow temperature ranges between approximately +20 and +160°C (Figure 7). At low temperatures, the resistance value is low,e.g. 40 ohms, and varies only slightly with changes in temperature. When the Curie temperature is reached, barrier layers are formed in the ceramic and the resistance increases exponentially with temperature by more than three orders of magnitude. Above this usable range with large temperature coefficients, α again decreases and sometimes even becomes negative. The initial resistance R_{25}, at 25°C, lies between 3 and 1,000 ohms depending on the model. The variation between probes of the same type necessitates individual calibrations. For rough measurements within small temperature ranges, the high temperature coefficient permits the fabrication of relatively inexpensive, but quite sensitive, resistance thermometers. There has not been enough experience in using them in aircraft to make statements on their stability, aging, etc. (Reference 73).

2.1.2 Resistance Materials with Negligible Temperature Coefficients

Resistors of manganin (a Cu-Mn-Ni alloy) wire exhibit a nearly constant resistance value over a wide range of temperatures. For this reason, this material is very suitable for manufacturing precision resistors, which are necessary in the bridge circuits for temperature probes. The temperature coefficient of constantan (a Cu-Ni alloy) wire is also close to zero, but exhibits a large thermoelectric effect when used with copper, so that it is not recommended for use in resistance-temperature measurements.

*PTC = positive temperature coefficient.

2.1.3 Elements with Negative Temperature Coefficients

Semiconductors with negative temperature coefficients, the so-called thermistors (also called Newi, Urdox, Thernewid or NTC resistors) are made of sintered ceramics, usually of mixtures of metal oxides, e.g. iron, manganese, nickel, cobalt, copper, titanium or uranium oxide, preferably with values for R_{25} between 100 ohms and 100 kilohms. As is the case with most semiconductor probes, the same types do not have identical temperature coefficients, so that a recalibration of the thermometer arrangement is usually necessary when the probe is changed. However, there are selected models with very close tolerances and probe assemblies that are directly interchangeable. In the latter case, a resistance (e.g. 3 kΩ) is applied in series with the probe and a resistance (e.g. 30 kΩ) in parallel with this series arrangement so that the resultant R/T curve exhibits a somewhat lower temperature coefficient than the element itself. The range of applicability of thermistors usually lies between the limits of approximately -100°C and +300°C, but an individual type rarely has a temperature range greater than 100 to 250°C. Larger measurement ranges often require a change of the bridge or different types of probes. Special types of thermistors are available where the lower limit at the present time is approximately 10°K (approximately -263°C), and the upper limit is approximately +400 to +450°C.

Because of their small size, thermistors are especially suitable for point measurements or for determining the temperature gradients between closely spaced points.

The resistance curve of thermistors with respect to temperature is approximately exponential (Figure 7) and can be expressed by the equation

$$R_x = a \cdot e^{b/T} \tag{3}$$

where "a" (in ohms) is a constant determined by the form and material and "b" (in °K) is a constant determined primarily by the material. The temperature coefficient can be described by the equation

$$\alpha = \Delta R/(R_{25} \cdot \Delta T) = -b/T^2 \tag{3a}$$

and is typically -3 to -5%/°C (depending on temperature T). In specifications, the resistance R_{25} is usually given (i.e. R at 25°C or 298°K), as well as the b-value or the resistance ratio R_{x1}/R_{x2} at given temperature values T_1 and T_2 (in °K). The connection between both definitions is expressed by the equation

$$R_{x2} = R_{x1} \cdot e^{b(1/T_1 - 1/T_2)} \tag{3b}$$

This equation enables us to calculate the resistance value R_2 for any value of T_2 (Reference 78).

The advantage of the large temperature coefficient, which is approximately 10 times greater than those of Pt and Ni resistances, can only be completely exploited when measurements are made in good thermal conductors. The self-heating in air is very great so that very small measuring currents must be used in the probe. As a result of the differences in construction of beads or rods in contrast to wire-wound resistors, the fact that the current flow in the material is not uniform when there is a change in temperature or when there are large currents must be considered for all types of semiconductors, since heat can only be absorbed or released at the surfaces*. For voltages of less than 1 volt at the probe, the contact resistance at the pigtails must also be considered, since there is a relationship between the voltage and the contact resistance (Reference 78). Probes supplied with inexpensive connectors often cannot be used to measure air temperatures for exactly this reason.

In order to measure extremely low temperatures (Figure 8), elements of doped germanium can be used (in addition to the special low-temperature thermistors mentioned previously) in the range from approximately 1 to 40°K (approximately -272 to -233°C). Also carbon resistors can be used in the range from approximately 0.1 to 15 or 20°K. Above approximately 23°K (-250°C) these materials are not as good as platinum, both with respect to the value of their temperature coefficients and their stability, so

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*It could be that "recovery factors > 1.0" found in air with thermistors are based on this effect (cf. Section 4.3.3.2).

they will not be discussed further in detail.

2.1.4 Construction of Resistance Elements

Because of the different requirements (time constants, ability to withstand vibration or chemical corrosion, etc.), or their relative importance in any individual application, construction techniques vary considerably. Figure 9a shows a so-called open-wire element in which a platinum wire is wound on a perforated coil form. Therefore, the wire is in direct contact with the gas or liquid whose temperature is to be measured. This element generally has an excellent response time, a small conduction error, and a small self-heating error. The gases or liquids must not be corrosive or conductive, nor leave sediments behind. In the case of tubular elements (Figures 9b and 9c) the resistance wire is protected and withstands high pressures, fluid speeds and vibrations, and is less easily damaged by small foreign bodies. In the more expensive model, Figure 9b, the wire is hermetically sealed between two platinum tubes, whereas in the model shown in Figure 9c, the wire has an organic coating and is embedded in ceramic. Since both the inside and outside of the tube is exposed to the medium, both a good time constant and good protection of the resistance wire is obtained. Figure 9d shows a ceramic coated miniature element which must be protected against mechanical damage by a wire net or cylinder, but has a relatively small time constant as a result of its small size. Too high bridge currents can overheat this type of element. The well-type elements, Figures 9e, f and g, protect best against fast flowing, electrically conductive or corrosive liquids, even when there are heavy oscillations, by enclosing the resistance wire inside of a stainless steel or platinum tube which is sealed at both ends. These types are primarily used as top-sensitive elements for measurements in solid bodies and stem-sensitive elements for measurements in liquids and gases. The well-type elements that require a solid core inherently have very high time constants. In stem-sensitive models without a core (Figure 9g), the time constants are sharply reduced. In the case of flat elements (Figures 9h and i, showing elements of quadrilateral form*) the resistance wire is embedded in insulating material and housed in a thin metal chamber (stainless steel, aluminum, platinum etc.). In some models, a metallic layer is vaporized on the insulation material, which, however, is quite subject to mechanical damage. Flat elements are mainly used to measure surface temperatures. For this purpose, there are special models in which the resistance wire is sandwiched between two flexible pieces of resin-impregnated glass paper so that it can be wound around small tubes.

The widely different types of element construction result, among other things, from the fact that the thermal expansion of the resistance wire, insulation, coil forms and chambers must be as equal as possible in order to prevent changes in the resistance by elongation (mechanical stress) of the resistance wire (similar to the effect of a wire strain gauge). Otherwise, the calibration of a probe element would be almost or completely non-reproducible. The required vibration resistance and the resistance value, i.e. the required wire length, also are important considerations. Semi-conductor elements are manufactured in very different forms, e.g. as spheres (Figure 9k) or disks, spheres embedded in a glass bead, layered glass laminates, or even in special mountings (Figure 10), and are characterized by their small size. Special models for use in aircraft have not yet been developed.

2.1.5 Construction of Resistance Probes

When a resistance element, which is usually manufactured without a mounting and with loose lead wires, is provided with a socket, a protective shield or a housing, and a radiation shield, it is known as a temperature probe.

The well-type probe is one of the most common types of construction. The types that are intended for temperature measurements in liquids and gases are made up of a long tube, a threaded base and a socket for the electric connection. The resistance element is housed in the rod-shaped protective tube and is so constructed that the temperature changes on the surface of the cylinder have more effect than those at the upper end (stem-sensitive type). Examples are shown in Figures 12a and b. There are models in which a second concentric protective tube can be attached (as can be seen on the bimetallic thermometer in Figure 11). This serves to reduce the radiation error, to channel the flow between the protective tube and the element so that the recovery factor becomes larger, and to decrease the time constant with

*Cf. also Figure 94.

respect to the temperature changes of the liquid or gas (see Section 4.3). For the same reason, certain types are given a right-angle bend (Figures 12c and d) so that the air current flows parallel to the temperature sensitive part of the surface. They are also manufactured with an additional radiation shield. Another type of construction, which is only used when measuring air temperatures, is the knife-edge-probe that is mounted perpendicular to the surface of the aircraft, but parallel to the air flow (Figure 13b). Figure 13a shows a type, which was earlier quite common, that is equipped with a protective housing that also serves to a certain extent as a radiation shield. A similar structure is the flush-bulb (Figure 14). It is mounted flush with the airplane skin, but thermally insulated from it. The air flows parallel to its surface. This probe provides a relatively inexpensive method of measuring the air temperature, but the measurement accuracy decreases quickly as the speed is increased, even when the mounting location is carefully chosen, since the temperature is measured in the boundary layer of the aircraft (cf. Sections 4.3 and 6.4.2).

All modern probes for measuring the outside air temperature on aircraft are total temperature probes (TAT) with a tube shaped housing, which is mounted parallel to the free-flowing air outside the boundary layer of the aircraft, and in which the resistance element is arranged concentrically (Figures 15, 16 and 17). In order to minimize dissipation of heat, the element is mounted on supports which have a low thermal conductivity and it is surrounded by one or more concentric heat shields (cylinders) to prevent heat loss by radiation. The lead wires to the element are also constructed in such a way that they conduct and dissipate as little heat as possible. The air which enters through the open front of the chamber is almost completely decelerated and the air is compressed nearly adiabatically. The resistance element measures the sum of the static air temperature and the temperature rise by compression, i.e. the total air temperature, with only a small error. In order to minimize the time lag with respect to the rate of temperature change, and the losses resulting from dissipation and radiation, openings near the rear or downstream end of the chamber provide a controlled air flow. This flow rate reaches it's maximum value (typically 0.3 Mach) at an aircraft speed of 0.7 to 0.9 Mach and remains constant at higher speeds.

In addition to the earlier type of TAT probe described above (Figure 17 left side), models with internal air flow deflection ahead of the element (Figure 18) are used. In this case, the element is arranged with its radiation shield perpendicular to the direction of flow, and the air is deflected 90° before it passes through the measurement element. Particles of water and dust can leave the probe housing directly without coming in contact with the measurement element. This provides greater stability in relation to external influences and also yields a significantly lower error of measurement when flying in a rain storm. Most models can be manufactured with a heater winding near the air intake to prevent icing. Laterally arranged small openings allow the heated internal boundary layer in the probe housing to be removed (boundary layer control) , so that errors in the measured temperature caused by the deicing heat are kept within tolerable limits as long as the Mach number does not go below 0.3 Mach (for housings with one element).

Two other types of probes for measuring temperatures outside the boundary layer of aircraft are of historical interest. It seemed logical to counteract the temperature rise of the air by appropriately lowering the temperature at the temperature probe element. Using an effect discovered by Ranque as early as 1933, which showed that there is a decrease in temperature at the center of a rotating stream of air (vortex), probes were built with tangential and axial flows (Figures 19a, b and c), i.e. the so-called vortex probes. These probes were used to directly measure the static air temperature. However, the increase in temperature caused by a varying mixture of friction and compression was compensated by means of a temperature reduction caused by adiabatic expansion, and hence the desired effect could only be realized exactly for one flight altitude and up to a certain speed. This was determined by the aircraft speed at which the accelerated internal flow reached the speed of sound. This critical velocity was between 0.45 Mach or 300 knots at sea level and approximately 0.8 Mach or 350 knots indicated airspeed at 30,000 feet. Even at low speeds there were positive or negative errors of up to 2°C depending on altitude and on the probe design, with distributions of up to double that error. Therefore, this type of temperature sensor is seldom used because, among other things, there is no possibility of deicing the probe (References 27 through 30).

The other probe of historic interest, the so-called reverse flow probe shown schematically in Fig. 19d, could only be used for temperature measurements at speeds up to approximately 100 or 150 knots, and consisted of a cylindrical housing installed parallel to the air flow with a coaxially mounted resistance element. The air stream flowing over the housing sucks air from the housing through a circular slot near the nose, the principle of the jet pump. The air is replaced through the rear end of the cylinder which is open. An almost constant internal rate of flow can be achieved within a certain range of aircraft speeds. This type of probe was also calibrated, so the instrument directly indicated the static air temperature. Since it is almost insensitive to rain, snow, etc., it is still occasionally used in flight meteorology and for aircraft icing tests. A new design (cf. Figure 19c) for use on helicopters (with speeds up to 175 knots) has the housing axis perpendicular to the air flow with two lateral slots and one slot in the rear of the housing. This design is said to be very insensitive to large angles of attack and also to be very insensitive to icing. It will still function even if one of the lateral slots should be totally iced. As indicated above, deicing of probes by heaters introduces excessive measurement errors at very low speeds.

TAT probes to measure engine compressor inlet temperature (Figures 20 to 22) must be especially vibration resistant to minimize the possibility of parts breaking off and damaging the engine. For this reason, and the need for low overall height, they may have relatively large errors (cf. Section 5.2). The simplest construction is similar to that of the well-type probe with a protective shield. However, a large aperture facing the flow and only small exhaust openings insure that there is adiabatic compression of the air at the element. In contrast to the hermetically sealed element perpendicular to the air flow, shown in Figure 20b, the flow is parallel to the horizontal wires of the open-wire element shown in Figure 20a. Figure 21 shows two probes mounted on a horizontal support for measurements in lift engines. The probes shown in Figure 22 have much more favorable characteristics. These are intended for mounting in the wall of the air inlet (cf. Section 6.1.2).

Temperature sensors for measurements in solid bodies are shown in Figures 23a, b and c. They are also well-type probes but have their greatest sensitivity at the end or head of the cylinder (tip sensitive types), which touches the bottom of a cylindrically shaped bore in the body (cylinder head) whose temperature is to be measured. The model in Figure 23c has a movable tip below a spring which equalizes the elongation at high temperatures.

For temperature measurements on surfaces, complete probes are not generally used, only the appropriate sensing elements (see above and Section 6.1.4).

2.2 Circuits and Indicators for Resistance Thermometers

2.2.1 Bridge Circuits

The resistance, and hence, the temperature of a resistance element can be precisely measured in bridge circuits. The simplest form is represented by the half bridge shown in Figure 24a. It consists of a precision resistance R_1 connected in series with the temperature sensitive resistance R_x. The circuit must be supplied with a highly stable energy source, since changes in voltage enter directly into the measurement. The load can be applied either in parallel with R_x or in parallel with R_1 (preferable when the temperature coefficient of the probe is negative). An arrangement of several half bridges connected in parallel to the same power supply does not permit exact measurements to be made because of mutual coupling, especially when common returns are used.

Full bridge circuits, which are variations of the Wheatstone bridge, are used much more often. The most common type (Figure 24b) consists of 4 bridge resistances R_1, R_2, R_x and R_4, where R_x is the resistance which varies with the temperature. The load resistance R_d is applied along the diagonal and, depending on the application, may either be a galvanometer (used either as a temperature indicator or a null indicator) or an amplifier input. The diagonal current I_d has a specific magnitude and direction depending upon the amount and direction of the deviation of the bridge from a state of equilibrium, and the direction of the current is reversed when it passes through the point of balance.

In the case of the so-called unbalanced bridge circuit, only the probe resistance varies during the measurements, and the diagonal current is proportional to the change of the probe resistance from the selected point of balance. The diagonal current can be calibrated directly in units of resistance or temperature, but it must be remembered that the bridge arrangement is generally nonlinear, i.e. an absolutely linear calibration can only be obtained under certain conditions.

In the case of the balanced bridge circuit, one of the two resistances R_1 or R_4 that are adjacent to R_x, is either manually or automatically adjusted during each measurement so that the diagonal current becomes zero. The resistance of R_x, or temperature, is read on a scale for the variable resistance R_1 or R_4. In the case of manually balanced bridge circuits, the load along the diagonal R_d is usually a null indicating galvanometer. In the case of self-balancing bridge circuits (servo indicators) on the other hand, R_d is the input impedance of an amplifier. The amplifier supplies power, proportional to the unbalance voltage, to a motor that adjusts the potentiometer, R_1 or R_4, in the correct direction, so that the current or voltage along the diagonal becomes zero.

The step balanced bridge circuit has characteristics between unbalanced and balanced bridge circuits. A temperature indicator having a relatively narrow range is used, and the variable resistance R_1 or R_4, is switched as needed to keep the indicator on scale. Thus the accuracy of the reading is increased. However, an increase in the bridge supply voltage is generally needed, resulting in a considerable increase in self-heating of the temperature probe.

The resistance bridge can be supplied with either DC or AC power (usually 400 Hz). The latter type of supply necessitates certain safety procedures to guard against pick-up from external sources.

A totally different type of arrangement is constant current excitation shown in Figure 24c. This type will be discussed in greater detail below.

2.2.2 Compensation of Lead Resistances

The probe resistance R_x is usually mounted some distance from the bridge so that the lead resistances must be taken into account if the minimum value of the probe resistance is not at least 100 times larger than the sum of the lead resistances. In practice, this usually means that the lead resistances must be considered whenever the minimum resistance value of the probe is less than 300 ohms or whenever especially long probe leads are unavoidable.

The lead resistances can be largely compensated by the use of a three-wire connection as shown in Figure 25a, whenever the lead resistances RL_1 and RL_3 have the same magnitude (and always vary in the same direction by the same amount). This technique is used in full and half bridges in which the resistance R_1 is approximately the same as the average value of R_x. The lead resistance RL_1 increases resistance R_1, and the lead resistance RL_3 increases resistance R_x by the same amount. The lead resistance RL_2 is in series with the load R_d and has practically no effect on the current flowing through the diagonal if the resistance of R_d is relatively high.

The circuit in Figure 25b is used whenever R_4 is the same as the average value of R_x or R_4 is adjusted to the appropriate value of R_x by a manual or servo arrangement. In this case RL_1 increases the value of R_4 and RL_3 increases the value of R_x by the same amount; RL_2 is in series with the bridge supply voltage. If there are very good grounding possibilities at point M directly at the probe and at point E, then the third lead (RL_2) can be eliminated; point N, however, must be insulated from ground. In this way, a compensating effect can be obtained with a two-wire connection. In the circuit shown in Figure 25a, this technique (grounding point M) is not possible.

The compensated Wheatstone bridge shown in Figure 25c is seldom used. It requires a four-wire connection between the bridge and the probe location, but only two of the leads are actually connected to the probe. A four-wire connection is used much more often with a Kelvin double bridge, shown in Figure 25d. This yields the best compensation of all the circuits shown, but its sensitivity is less than that of a simple bridge. For this reason it is most often used with an amplifier circuit. Its method of operation is based on the fact that the lead resistances RL_1, RL_2 and RL_3 are in series with relatively large resistances so that small fluctuations in the lead resistances are of little importance. The bridge is also laid out in such a way that in its normal state, there is no current flowing through RL_4. However, if there are appreciable changes in the lead resistances, then a current flowing through RL_4 adjusts the potential conditions of the lower half of the bridge so that there is compensation of the effect of the changes in the leads.

In using the three-wire compensation techniques (Figures 25a and b) there is true compensation only when the lead resistances are absolutely equal (and change by the same amount in the same direction), and when the bridge is balanced. In the case of unbalanced bridge circuits, the lead resistances must be taken into account when calibrating the thermometer. Thus for example, an increase in the resistance of one lead by 4% (in this case = 0.02 ohms) in a bridge where the probe resistance is 50 ohms and the lead resistances RL_1 and RL_3 are each 0.5 ohms, results in a measurement error of approximately 0.1°C.

For special cases where high lead resistances (in comparison with R_x) are unavoidable, special bridge circuits, (e.g. the triple bridge, Reference 17) may be used to eliminate the effect of greatly varying lead resistances. It must be noted that in all complex bridges the sensitivity always drops to less than half that of a simple bridge. This means that amplifiers are usually necessary for sensitive measurements, especially when the self-heating of the probe must be kept low. These amplifiers must be carefully compensated with respect to temperature in the case of unbalanced bridge circuits.

Figure 24c shows a special half-bridge circuit which uses constant current excitation instead of constant voltage excitation. A self-regulating power unit supplies a constant current in the series circuit that consists of a temperature probe R_x, the lead resistances RL_1 and RL_2 and the calibration resistance R_c. Since the current is always kept constant, even when the resistances change, the voltage drop across the probe resistance R_x is a direct measure of its resistance value, or its temperature. Changes in the lead resistances have no effect, and the linearity of this circuit is better than that of the others if the voltage at R_x is measured with separate leads (four-wire connection of the temperature probe) and with a high-resistance indicator (Reference 37). The calibration resistance serves to adjust and monitor the desired current value. The stability of the current source must, however, be better than 0.01% and good hum filtering should be provided, since the noise across the output in this case equals the hum voltage. The circuit requires a power supply unit for each half bridge. A variation with a suppressed zero is possible (Reference 84) but requires a second power supply unit with an extremely high stability, better than 0.001%.

2.2.3 Design of Bridge Circuits

When designing bridge circuits, there are always a series of spec. ic conditions to be met, which lead to very different methods of solution depending upon the given conditions and the purpose for which the circuit is to be used. This fact explains the extremely large number of circuits in use, and also explains the apparent contradiction of design rules found in the pertinent literature. For measurements in aircraft, there is an additional series of conditions, which are at most merely hinted at in the general literature and which may lead to incorrect measurements if they are ignored. Besides the rougher environmental conditions such as temperature range, vibration, long leads, variations in the aircraft voltage supply, etc., there is also the problem of the maximum permissible self-heating of the probe by the measuring current, which must be taken into account.

In addition to the indicator, a recording or telemetry output is also necessary for the thermometer.

When designing a bridge circuit, the following factors must be taken into consideration:

(a) The power dissipated in the probe resistance R_x, and therefore the self-heating error, is always greatest for a given bridge supply voltage when the instantaneous value of R_x is the same as the value of R_1 (at a given temperature).

(b) The self-heating error and the lead resistance errors are inversely proportional to the resistance value of R_x for a given supply voltage and a given ratio R_1/R_x. Probes with higher resistance values for a given type yield smaller self-heating and lead errors.

(c) For a given type, wire wound probes with high resistance values (>500 ohms at 0°C) usually exhibit a greater response time with respect to temperature changes than those with low resistance values. Probes with extremely high resistance values, e some types of thermistors, increase the susceptibility of the circuit to insulatio. failures (e.g. from corrosion) and pick-up from external sources (hum voltage). Since the insulation resistance of all insulation materials is reduced at very high temperatures, only probes with relatively small resistances may be used for measurements at high temperatures. If low resistance probes are used, together with relatively high resistance leads, they should be used with three- or four-wire connections.

(d) Since all bridge circuits are nonlinear, the bridge output voltage is always a nonlinear function of the probe resistance. The latter, however, is also a nonlinear function of the temperature. Nevertheless, when the bridge is designed appropriately, a nearly linear indicator scale can be obtained, as long as the indicator range is not extremely large.

(e) Bridges with recording or telemetry outputs should be designed so that the connected equipment has as little effect as possible on the calibration of the circuit.

(f) Changes in the environmental temperature of the bridge should not have any effect. This can be achieved by using bridge resistors of manganin wire and all the connecting leads of copper wire. In certain cases, the bridge resistances can be made using a copper wire section and a manganin wire section. The use of constantan and other metals, which in combination with copper generate an appreciable thermoelectric voltage, must be avoided in all cases. Even the temperature sensitivity of the leads connected to the bridge, including that of the instruments, amplifiers, etc., must be studied and if necessary compensated for.

(g) The bridge supply voltage should be regulated to better than 0.1%, since voltage changes yield corresponding read-out errors. Exceptions are bridges with ratio meters, and servo bridges.

(h) In addition to the requirements for high insulation resistances, extremely small contact resistances in plugs, switches, potentiometer slip rings, etc., the usual environmental requirements for aircraft equipment, such as vibration resistance, must also be met.

When designing a circuit, it is therefore always necessary to compromise between requirements that are to a certain extent contradictory.

2.2.3.1 Half Bridge Circuits

This circuit, which is usually used only for recording or telemetry purposes, consists primarily of the probe resistance R_x and a series resistance R_1 (Figure 24a). The use of a three-wire connection is possible, if R_1 is approximately equal to the mean value of R_x. For reasons mentioned below, relatively high resistance probes are selected so that the effect of the leads is very small; in this case, the magnitude of R_1 can be any selected value. If the load R_d, which should be as high in resistance as possible, is connected in parallel with R_1, the relationships are more easily seen as the magnitude of R_x varies. It must be taken into account that the polarity of the output voltage depends on whether R_d is applied in parallel with R_1 or R_x, and also depends on whether the probe temperature coefficient is positive or negative.

In Figure 26a the ratio of the output voltage U_A (across R_1) to the supply voltage U_B is given as a function of the temperature, when nickel probes are used (the curves would be similar for platinum probes). The useful change in the output voltage is relatively small for Case I but is increased by selecting higher probe resistances and lower values for R_1/R_o (Case II). In Case II, a relatively high bridge supply voltage U_B was also selected. In order to prevent excessive self-heating of the probe by the measuring current, the maximum bridge voltage is determined by the maximum permissible power generated in the probe.

Example (refer to Figure 26a):

Range of measurement	= 0 to 100°C
Probe resistance at 0°C R_o^* (nickel element)	= 100 ohms
Dissipation constant in flowing water	= 112 mW/°C
Permissible self-heating error	= 0.2°C
Permissible power at the probe $P = (112 \cdot 0.2) = 22.4$ mW	= 0.0224 W

*When R_o = 100 ohms, the lead resistances will have a significant effect.

Probe resistance at 100°C R_{100} $= 161.7$ ohms

Maximum probe voltage at 100°C U_{100} $= \sqrt{P \cdot R_{100}} = 1.9$ V

Resistance R_1 $= 1{,}100$ ohms

Sum of R_1 and R_{100} $= 1{,}261.7$ ohms

Bridge voltage U_B $= \frac{1.9 \cdot 1261.7}{161.7} = 14.85$ V

Since in this example, R_1 is greater than R_x over the entire range, the maximum probe resistance value (at 100°C) must be selected for the computation of the self-heating of R_x. The voltage across R_1 is approximately 13.61 V at 0°C and approximately 12.97 V at 100°C so that there is a useful change in the output voltage ΔU_A of approximately 0.64 V. Using a probe having a resistance of 1,200 ohms at 0°C in series with a resistance R_1 of 4,000 ohms, and a bridge voltage supply of 18.7 V, ΔU_A increases to 2.75 V. As previously stated, the given values are only valid when the input resistance R_d of the load is significantly greater than R_1.

When thermistors are used, relatively high changes in the output voltage can be obtained even when the bridge voltages are very small, in contrast with the above example. Figure 26b gives the corresponding curves for a fixed value of R_1 (10 kΩ) in series with different thermistor resistance values at 25°C (R_{25}); Figure 26c, however, shows the curves for a fixed value of R_{25} (4,000 ohms) in series with different values of R_1. It can be seen that in spite of the nonlinearity of the R/T curves of the thermistors, the output voltage curves can be made somewhat linear within a limited range. By appropriately selecting the probe resistance R_{25} and the series resistance R_1, the very nonlinear portions of the curves (corresponding to large scale deviations) can be shifted beyond the desired measurement range.

Table 2 shows several design possibilities for R_x and R_1 in a thermometer circuit with a measuring range of 0 to +100°C, and assuming 0.2°C is the maximum permissible self-heating error E_{SH}. The self-heating (in °C/mW) or its reciprocal value, the dissipation constant G(in mW/°C), and the time constant τ must be determined using the same measurement method (e.g. cooling water flowing at a rate of 5 m/sec). Maximum power P_{max} is generated in the probe at the temperature where R_x and R_1 are equal. Thus, the maximum permissible bridge supply voltage U_B is double the output voltage (across R_1) at which P_{max} occurs ($U_A/U_B = 0.5$). Since the output voltage U_A is measured across R_1, the minimum voltage value, when thermistors are used, occurs at the lower limit of the range, and the maximum value at the upper end of the range. This pattern is reversed when nickel or platinum probes are used.

TABLE 2. LAYOUT OF BRIDGE RESISTANCES IN HALF-BRIDGE CIRCUITS

R_{25} kOhm	R_1 kOhm	G mW/°C	τ sec	$E_{SH\ max}$ °C	P_{max} mW	$U_{B\ max}$ V	U_A V	ΔU_A V
4	3	10	10	0.2	2	4.8	0.48-3.76	3.28
4	3	1	1	0.2	0.2	1.6	0.16-1.25	1.09
15	11	1	1	0.2	0.2	3.0	0.30-2.40	2.10
100	75	1	1	0.2	0.2	7.6	0.76-6.08	5.32
(a) 0.1	1.1	112	6	0.2	22.4	14.85	13.61-12.97	0.64
(a) 1.2	4	112	6	0.2	22.4	18.7	15.3 -12.55	2.75

(a) Probe with nickel element; resistance at 0°C.

A comparison of the values shown in Table 2 for thermistors shows that larger useful changes in the output voltage ΔU_A can be obtained, either by using probes with higher* designed resistances (R_{25}) or types with higher dissipation constants and thus, larger time constants. In Table 2 the output voltages of nickel probes are not directly comparable with those of the thermistors as the dimensions of the nickel probes are larger than those of the thermistors, as appears from the large dissipation-constant. Same dimensions would have resulted in a much larger U_A for thermistors due to the greater slope of the R/T curve. The use of thermistors instead of platinum or nickel elements is often recommended wher the measurement ranges are not too great and relatively high useful voltages are required for small bridge supply voltages (e.g. 1.5 or 3.0 V), as in the case with portable measurement devices. However, thermistors are not generally recommended for aircraft measurements.

2.2.3.2 Full Bridge Circuits

For full bridge circuits with an indicator (Figure 24b), and for a given bridge supply voltage, the output power available in the diagonal branch of the bridge, and, therefore, the indication of a connected instrument, is greatest when the four bridge resistances R_x, R_1, R_2 and R_4 and the instrument resistance R_d in the diagonal branch are of approximately the same magnitude. However, when it is necessary to avoid excessive self-heating errors, the maximum output power occurs when there is a certain maximum allowable power dissipation in the probe resistance R_x, and very different design rules must be used. When there is constant power dissipated in R_x, the output power increases with the ratio R_1/R_x, when there is a corresponding increase in the bridge supply voltage. Thus, if the ratio R_1/R_x is increased from 1/1 to 10/1, the output power in the diagonal branch is approximately doubled. When the ratio is further increased, the output power increases only slightly.

In the case of simple indicators such as galvanometers, the instrument resistance R_d is approximately equal to or less than R_x. In this case, an additional power increase of 50% can be obtained in the diagonal branch if the resistances R_4 and R_2 in the right half of the bridge are reduced by a factor of one-half to one-fifth. Further decreases yield no substantial power increase, and require uneconomically high supply power from the source. By combining both techniques, a power increase of approximately 200% can be obtained in contrast to a design with equal resistances, for constant power dissipation in R_x. The following are the approximate values for the optimum values of R_d/R_x:

when $R_1/R_x = 1/1$ and $R_4/R_x = 1/1$, then R_d/R_x should be approximately 1/1.
when $R_1/R_x = 5/1$ and $R_4/R_x = 1/1$, then R_d/R_x should be approximately 1.8/1.
when $R_1/R_x = 5/1$ and $R_4/R_x = 0.2/1$ then R_d/R_x should be approximately 1/1.

Even larger values of R_1/R_x or even smaller values of R_4/R_x have no significant effect on R_d/R_x. Deviations of R_d/R_x by a factor of 2 from the given ratios are not critical and have more effect on the output linearity than on the sensitivity of the indicator.

In the case of balanced bridge circuits where negligible power is required in the diagonal branch, the ratios R_1/R_x (and R_2/R_4) are often 150/1 or more, permitting a small amount of self-heating and a simple circuit for the power supply unit, since in this case the current requirement of the bridge circuit remains nearly constant. A high input resistance for the amplifier in the diagonal branch also minimizes the nonlinearity of the bridge circuit and thereby makes linearization of the indicator easier. (cf. also Section 2.2.5).

When a thermistor is used, the bridge must be balanced at the lower end of the temperature range. According to Figure 26c, the best linearity is obtained for the measurement range of 0 to +50°C when a probe with a nominal resistance R_{25} of 4,000 ohms is selected and used with a resistance R_1 which also equals 4,000 ohms. The resistance R_2 is assumed to be 4,000 ohms so that R_4 must be the same as the value of R_x at 0°C, or 11,400 ohms. With a bridge supply voltage of 1.5 V, the maximum probe power is approximately 0.2 mW, i.e. the amount of self-heating remains small. When a 50 μA indicator is used, it must be connected in series with a resistance of 10,000 ohms. The exact process of calculation for this type of instrument, which exhibits excellent linearity, is described in Reference

*When using probes where $R_{25} = 100\ k\Omega$, difficulties can be expected with the insulation resistance.

80. Larger measurement ranges are hard to obtain with thermistors, since the improvement in linearity by applying a resistance in parallel with the thermistor is very small. In this case, it is preferable to use a silicon semiconductor with a positive temperature coefficient.

Linearization of the indicator is often unnecessary in bridge circuits when nickel probes are used, since the nonlinearities of the nickel probes and those of the bridge circuits tend to counteract each other. Increased linearization can be achieved by applying a resistance R_p in parallel with R_x (Figure 27a), the value of which depends on the design of the bridge branches and the diagonal branch. Since this changes the balance of the bridge, either a relatively small resistance R_s must be applied in series with the parallel combination of R_x and R_p, or one of the two bridge resistances R_1 or R_4 must be appropriately reduced. Since the resistances necessary for linearization tend to counteract the lead resistance compensation effect of a three-wire connection, it is advisable to use high resistance probes, e.g. 1,200 ohms.

When platinum probes are used, the nonlinearity of the characteristic probe curve and that of the bridge circuit are added. For this reason, one can only try to keep the non-linearity of the bridge circuit, and by the same token, the amount of self-heating, as small as possible, by using a probe in which the resistance is as high as practical. If a probe with a designed resistance of 500 ohms at 0^oC is selected, R_4 should be approximately equal to the value of R_x at the geometric center* of the measurement range. The value of R_1 must be at least 2,000 ohms, and the instrument resistance R_d should lie between approximately 1,500 and 2,000 ohms. Since in the case of unbalanced bridge circuits with platinum and nickel probes (or probes with similar characteristic curves), the point of balance** must lie at the geometric center of the measurement range if good linearity of the output voltage is to be obtained, galvanometers with null at the center of the scale must be selected.

In order to measure temperature difference (Figure 27b), it must be noted that calibration of the instrument scale independent of the given temperature level is only possible when the characteristic curves of temperature probes that are connected differentially are exactly linearized. Since the use of a three-wire connection would be of little value in reducing the effect of lead resistances that vary in magnitude, and the complete circuit would only be made unnecessarily complex, it is advisable to use high resistance probes in this case also. Resistances of approximately 4,100 ohms applied in parallel with 1200-ohm nickel probes ensure the applicability of the bridge to the temperature range between -70 and $+50^oC$. When the mean temperature is at a higher level, the parallel resistances must also have higher values corresponding to the higher probe resistances. For differential measurements, both probes must have the same characteristics and especially the same time constants.

For temperature-rate measurements, i.e. measurements of the direction and rate of the temperature changes, the circuit in Figure 27b is also used. In this case, the measurement is taken with the probes mounted at the same point. The probes should also be as similar as possible, except they must have time constants that are as different as possible. Since very small time constants can only be obtained with open-wire elements, which are only manufactured in low resistance models (nominal resistance of 100 ohms is maximum), this is a case where a low resistance probe must be used in a two-wire connection. This is not as critical in rate measurements if both probes have approximately the same lead resistances. In certain cases a lead-balancing resistance must be applied.

Excitation of bridge circuits in aircraft instruments directly from the aircraft power supply (28V DC) through simple regulation circuits (e.g. a series resistance and a zener diode) is most convenient as a source of constant voltage. In the case of constant current excitation (e.g. through a variable resistance whose value depends on the current) there is a greater influence of the ambient temperature, since more circuit elements are usually required. When connecting bridges with telemetry circuits, the stabilized voltage supply for the amplifier should be used as the bridge source in order to avoid

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*E.g., for a measurement range of -70 to $+50^oC$ or 203 to 323^oK, the geometric mean temperature $T_M = \sqrt{203 \cdot 323} = 257^oK = -16^oC$.

**I.e, the point at which the bridge current is zero.

uncontrollable idle currents and pickup. If this supply voltage is too high, it is advisable to reduce it by means of a series resistance and a zener diode to the level best for the bridge, to take advantage of the additional stabilization. In half-bridge circuits, variations of the supply voltage appear directly at the output terminals; in full bridge circuits they are attenuated by one-half; when using ratio meters they are reduced by more than 90%; and in servo bridges, they do not appear at all (as long as the normal operating limits are not exceeded).

In the initial design of bridge circuits, it must be remembered that practical hints can only partly replace experience and that prototype models must be subjected to time-consuming environmental tests before they can be used in aircraft. Despite the higher costs, it is better to use industrial signal conditioners (References 17, 18 and 19) for precision measurements with recording or telemetry devices. The best solution is to use a servo-indicator (Figure 33) with two recording outputs (coarse and fine) and a counter wheel indication (cf. Section 2.2.6).

Whenever several temperature probes are to be switched alternately to a common bridge or indicator circuit, a three-pole switch must be used. Two poles switch the probe leads, and the third pole switches off the bridge supply voltage before the probe leads open and switches it on again after the process is completed. Otherwise, there would be an overload and damage to the instrument, since the voltage through the diagonal branch jumps to many times its normal value when a branch is opened. Figure 27c shows a circuit of this type with three probes.

2.2.4 Direct Indicators

In the simplest form of full bridges, the bridge resistances R1, R2 and R4 remain fixed while the circuit is operating. Figure 28a shows a typical unbalanced bridge circuit of this type connected to a galvanometer. In order to obtain as linear a scale as possible over a large measurement range, a galvanometer with a centered null point is used. The bridge supply voltage is stabilized with a zener diode that has a very small temperature coefficient. Since this voltage is usually high enough, for a bridge with approximately equal bridge resistances, to cause excessive self-heating of the probe, a series resistance R'_q must be used, or the values of R_1 and R_2 must be selected approximately 3 to 8 times greater than those of R_x and R_4. In the latter case, the amount of self-heating varies less than when the series resistance R'_q is used. In designing this type of circuit, the most sensitive indicating instrument available (e.g. ± 5 mV, R_d about 80 ohms), and at the same time a probe with as high a resistance as possible (Pt 500 ohm or Ni 1,200 ohm), must be selected. The bridge output voltage is then adjusted so that the full deflection of the instrument is obtained for the desired measurement range. In the case of high-resistance probes, the amount of self-heating is tolerable, but is very high for low resistance probes.

Figures 28b, c and d show direct indicator circuits utilizing <u>ratiometers</u>. In contrast to galvanometers, ratiometers usually have higher measurement accuracies and greater resistance to mechanical and electrical stresses. Their main advantage, before introduction of the zener diode, was that they were relatively unaffected by variations in the bridge supply voltage. On the other hand, they require a relatively high current through the probes so that when using the circuit in Figure 28b, and even more so the one in Figure 28c, they cause excessively high values for the self-heating error when measurements are made in gases, i.e. air*. In some of the circuit diagrams of Figure 28, trim potentiometers are shown for scale balance and range expansion, but the instrument temperature compensation for fluctuations of the ambient temperature is not shown. Usually compensating resistances of manganese wire, partly bridged by resistances with a negative temperature coefficient, are used.

2.2.5 Servoed Indicators

Self-balancing bridge circuits exhibit up to 20 times greater measurement and reading accuracy than simple instruments utilizing galvanometers or ratiometers, depending on the model and precision of the balancing potentiometer, and whether an indicator with several dials or a four-digit counter is used. This, however, presupposes linearization of the indicator scale (see below). Servocircuits with AC

*See Section 5.2.2.

amplifiers and AC motors are generally used in aircraft. Until recently, DC systems in the same price range had considerably higher drift rates than AC systems. Modern DC systems, however, provide very good results.

2.2.5.1 Servoed AC Bridges

When AC servocircuits are used (typically at 400 Hz) it appears to be simpler to supply the bridge with AC power as well, so that an AC signal can be directly obtained in the diagonal branch, whose amplitude and phase relationship are directly proportional to the amount and direction of the displacement from zero on the scale. Figure 29a shows a circuit of this type, with a linearized nickel probe, in which the bridge resistance R_4 serves as a balancing resistance. It is composed of a fixed resistance and a linear potentiometer that is adjusted in such a way that the sum of R_4 and R_4' is the same as the instantaneous value of R_x (assuming that $R_1 = R_2$). When there is a displacement from the point of balance, the unbalance voltage that occurs across the diagonal branch is amplified by amplifier A and fed to an AC motor, which drives, through speed reduction gears, the potentiometer R_4' and an indicator on a temperature scale and/or a counter. Depending on the amplitude and phase relationship of the error voltage, the motor turns the axis of R_4' in the direction that causes the error voltage to becomes less, and stops when the unbalance voltage reaches zero. The response characteristics of the servocircuit can be improved by a generator that provides a feedback voltage, proportional to motor speed, to the amplifier.

The use of AC power to supply the bridge has certain disadvantages. When the supply to the probe is 400 Hz, inductive reactance of the probe is not noticeable (measurable changes in the calibration are only apparent at frequencies above approximately 2 to 3 kHz), but stray capacitances to the aircraft structure (50 pf is not uncommon) may affect the adjustment of the null point. Also, AC power voltages (and VHF or UHF voltages) induced in the leads can cause false readings and can overdrive the amplifier if their amplitudes are large. When the frequency of the pickup voltage is near the bridge supply frequency, the system can oscillate at the beat (difference) frequency; when the frequencies are equal, there is a constant error which depends on the amplitude and the phase relationship of the pickup voltage. For these reasons, the leads to the probe should be twisted, shielded and separated from AC power leads as much as possible.

2.2.5.2 Servoed DC Bridges

When the bridge is supplied with DC power, the unbalance voltage of the bridge is often chopped for use in AC amplifiers (Figure 29b). Modern instruments use only solid-state choppers for longer reliable service. The other details are generally as described above. Stray voltages induced in the probe leads would have the same effect as above. For this reason filters with high attenuation characteristics (40 db at 400 Hz) are usually used at the bridge and chopper inputs. New types of servoed indicators use DC differential amplifiers and DC motors (not shown in the figures), giving a lower power consumption and higher accuracy. To avoid false readings during the operation of radio transmitters, all chassis plugs are fitted with RF filters. In the circuit shown in Figure 29b, the alternate form of the three-wire connection was selected for the probe resistance, and has the advantage that different probes (e.g. Pt with 50 ohms, 100 ohms or 500 ohms nominal resistance) can be used by simply changing the bridge resistance R_1 (45 ohms, 90 ohms or 450 ohms), without having to change R_2, the potentiometer R_4', or the calibration. The linearization of the indicator is achieved either by using a functional potentiometer or by an additional resistance R_5 connected in parallel with a linear potentiometer (Reference 15).

Figure 29c shows a slightly different bridge circuit that is only recommended for high resistance probes of 500 or 1,200 ohms. Range adjusting resistors are connected in series with R_2 and R_4 and the probe is only connected with two leads. Typically, the bridge is calibrated for a total lead resistance of 5 ohms, and for a specified application, the lead resistances are adjusted to 5 ohms with the"lead adjust" resistance R_{LA}.

The so-called "digital servocircuits", which have been developed during the last few years, also operate with resistance probes, i.e. with an analog voltage at the input. They will not be discussed here since up until now they have only been used as integral components of air data computers. in contrast to the computing bridges which will be discussed next.

All the bridge circuits that have been discussed so far have included no computing functions, i.e. the indicator is exclusively dependent on the resistance value of the temperature probe. When temperature measurements are made in fast-flowing air or other gases, the resistance of a well-designed probe is an accurate function of total air temperature, but not of static air temperature. By changing the bridge resistance R_1 as a function of Mach number (from flight altitude and speed) static temperature can be computed and directly indicated (Figure 30a). Usually the resistance R_1 is connected in series with a function potentiometer P_1, which is located in a servoed Mach transmitter or more frequently in an air data computer, and is adjusted according to the instantaneous Mach value. As a result, the point of balance of the bridge in question shifts from a value corresponding to the total temperature (TAT) to a lower value corresponding to the static air temperature (SAT) according to the following equation

$$SAT = TAT/(1 + 0.2\ M^2). \tag{4}$$

When the outside air temperature is low, the difference between TAT and SAT at a constant Mach number is smaller than at higher outside air temperatures. This means that not all bridge circuits are suited for the installation of a computing or function potentiometer. Figure 30a shows on the left side a typical AC bridge circuit with Mach correction for indicating the static air temperature (SAT). The servomotor that drives the linear follow-up potentiometer P_4 also drives another linear potentiometer P_{12}, which together with a linear balancing potentiometer P_{14}, a second Mach function potentiometer P_{13} and a fixed resistance R_{11}, make up the bridge (on the right side of Figure 30a) of an indicator for true airspeed (TAS).

Figure 30b shows a newer type of DC bridge circuit to indicate SAT in which only a two-wire connection of the probe is possible (i.e. no compensation for the lead resistances and hence, only a probe with a nominal resistance of 500 ohms or higher should be used). The resistances in the left branch of the SAT bridge, R_1, R_3, R_p and R_x, are chosen so that the voltage at point A is not a linear function of temperature T_T, but is proportional to the square root of T. This voltage is applied to two non-linear, voltage-dividing potentiometers(P_1 for SAT and P_{13} for TAS), both of which are displaced in proportion to the Mach number. The following voltage appears at the pick-off of the potentiometer P_1

$$U_1 = U \cdot \sqrt{T_T}\ /\sqrt{1 + 0.2\ M^2} \tag{5a/b}$$

and must be equalled by the voltage of the non-linear follow-up potentiometer P_4

$$U_4 = U \cdot \sqrt{T}$$

so that the unbalance voltage in the diagonal branch of the bridge becomes zero. (For simplicity, the constant factors in the above equations have been omitted). This circuit has the advantage that the TAS indicator is independent of the SAT indicator.

Examples of the types of instruments discussed above will be given at the end of the following section.

2.2.6 Bridges with Special Output for Recording

In this section, the concept of "output for recording" is always understood to mean an output in the form of an analog voltage which is used as an input to a galvanometer in an optical type recorder, a subcarrier oscillator in a telemetry system or an A/D converter in a digital tape recorder. In servoed bridges an electrically isolated output for recorders can readily be obtained by coupling transmitters such as potentiometers, synchros or shaft position encoders to the servomotor. In all nonservoed types of bridge circuits, with few exceptions, the two output leads have a specific potential with respect to the aircraft ground, so that ground conditions must be carefully considered to avoid short circuits or cross coupling, when connecting recording or telemetry devices.

In the case of thermometers with the usual types of simple indicators, it is almost impossible to get an output for recording, if the case of the instrument cannot be opened. Connections to probe leads might be permissible (but not advisable) if extremely high resistance loads are used. When connections can be made in the bridge circuit, an output for recording is readily obtained in the case of bridges for galvanometer type indicators as is shown in Figures 31a and b. In each of these examples, the indicator bridge has a resistance of approximately 100 ohms in each branch. For 500 ohm bridges, all values would have to be multiplied by 5. In the diagrams, R_r includes the input resistance of the recording device (e.g. recording galvanometer, subcarrier oscillator, etc.) since it must be conditioned to the bridge. The simplest circuit is the series arrangement of R_d and R_r shown in Figure 31a; in this case, however, the bridge sensitivity and hence the indicator calibration is changed. In order to obtain the same calibration as before, the bridge supply voltage must be increased approximately 1.5 times, in which case the self-heating error for the probe R_x increases to 220% of its previous value. When the indicator is moved by acceleration forces, its galvanometer acts like a generator, and pulse-like error voltages are generated that can invalidate the recording. A direct parallel arrangement of R_d and R_r instead of a series arrangement is usually worse unless the resistance of R_r is high in comparison with R_d.

Figure 31b shows a circuit in which R_d and R_r are differentially uncoupled by additional bridge elements (R_{52}, R_{53} and R_{55}), i.e. feedback from R_d to R_r and vice versa is impossible when they are properly connected. However, the bridge supply voltage must be increased to 200% of its original value, in which case the self-heating error for the probe increases to 400% of its previous value.

Figure 31c shows the circuit of a commercial <u>signal conditioner</u> with a four-wire connection of the probe, a Kelvin double bridge and a DC amplifier. It provides an output voltage of 0 to 1.4 VDC or 0 to 5.0 VDC that increases linearly with the temperature. This circuit is not affected by changes in the length of the probe leads (up to 30 meters), but should be operated at ambient temperatures between +10 and +40°C if the errors are not to exceed ±0.4° when measuring temperatures between -50°C and +100° C(References 17, 18 and 19). Similar bridge circuits are also available without amplifiers. They deliver a maximum output voltage of only approximately 50 mV and require a highly stabilized supply voltage. In the case of recording or telemetry systems with multiplexing, self-manufactured half bridge circuits or the previously mentioned bridge units without amplifiers can be connected through the multiplexer to an AC amplifier with a chopper and a demodulator. Another possibility is to use the bridge units with built-in amplifiers and connect their outputs to the multiplexer without subsequent amplification. This last technique is quite expensive, but gives the greatest accuracy and is least affected by external disturbances, since the signal level at the multiplexer is usually much higher than that of stray voltages. The most inexpensive, but most unreliable and inaccurate solution is the former, where it is extremely difficult to avoid common mode problems.

In the case of <u>servoed bridges</u> (Figure 31d) one or several output shafts are usually available to which a potentiometer P_2 (or even a shaft position encoder) can be coupled in order to obtain a voltage for recording purposes that is electrically isolated from the actual thermometer circuit. The resolution and accuracy of this type of recording output is not very high when a single-turn potentiometer with approximately 1% linearity is used. Better resolution is obtained with a second (fine) potentiometer P_1 (also single-turn) coupled so that it rotates faster, e.g. 20 times faster than P_2, the one used for coarse measurements. If P_1 rotates through 360° 20 times over the total temperature range of -70°C to +50°C, there is a fine recording with 20 intervals, each being 120°C/ 20 or 6°C. It should be noted that potentiometers generally have a deadband of about 1/100th of a turn or about 3.6°. Since this would amount to only 0.06°C, and the coarse recording is also available, this is considered to be tolerable.

Figure 32 illustrates several common temperature indicators with galvanometers (G) ratiometers (R) and servocircuits (S), of which the latter can be manufactured with an output potentiometer for recording purposes. Figure 33 shows servoed bridges with digital indication for 100 ohm platinum probes. They can be equipped with both coarse and fine recording potentiometers. The measurement range is -100 to +100°C with an accuracy of ±0.2°C. The instrument on the right in Figure 33 can also be switched for SAT indication (when connected to a Mach function potentiometer), operating in this case as a computing bridge. Figure 34 shows other computing servoed indicators for single and combined indication of SAT, TAT and TAS. They can all be manufactured with outputs for recording (potentiometer, synchro, etc.) and be connected to a source of the Mach function, e.g. an air data computer.

3. THERMOELECTRIC TEMPERATURE MEASUREMENTS

3.1 Thermocouple Probes

3.1.1 Thermocouples

A thermocouple is formed by joining (twisting or welding) two electrical leads made of different metals or metallic alloys. In each lead, the valence electrons, which are free to move between the crystal lattices, are under a "pressure" that is proportional to the temperature and the electron concentration of the lead in question. At the point of contact of the leads, the electrons diffuse through the boundary layer between the two leads as a result of the "pressure differential". Therefore, the one lead loses electrons, i.e. becomes positive, and the other gains more electrons, i.e. becomes negative, and an electromotive force ("contact e.m.f.") is produced. If the other ends of the leads are connected in the same way, an e.m.f. is also produced at that end in the opposite direction from the first one. The e.m.f. is highest at the point of contact where the temperature is highest. The difference between the contact e.m.f.'s, is proportional to the temperature difference, and causes a thermocurrent to flow. This phenomenon is called the "Seebeck effect". In addition, a number of other effects occur (Reference 42) but they are usually negligibly small in temperature measurements*.

When a millivoltmeter is inserted at any point in the above-mentioned arrangement of two leads, the Seebeck effect enables measurement of the temperature difference between the two points of contact. If however, one contact is kept at a constant (reference) temperature, the instrument scale can be calibrated to directly indicate the temperature at the other contact (thermocouple). The contact where the temperature remains constant is called the "cold junction", the other the "hot junction".

The Seebeck coefficient or sensitivity S of a thermocouple is $S = \Delta E/\Delta T$ and is measured in microvolts /°C. Table 3 gives nominal sensitivities near 0°C for various materials compared with platinum. The output voltage of two given materials compared with each other is obtained from the difference of the S values, e.g. chromel vs.alumel = (+25) - (-15) = 40 μV/°C.

TABLE 3. SENSITIVITY OF THERMOELECTRIC MATERIALS VERSUS PLATINUM

(a) Bismuth	-72 Microvolt/°C
Constantan	-35
Alumel (and Nickel)	-15
Platinum	0
Aluminum	+3.5
Rhodium	+6
Copper	+6.5
Iron	+18.5
Chromel	+25
(a) Selenium	+900

(a) For purposes of comparison

Table 4 is a list of the most important thermocouples and their characteristics, including information on the limiting temperatures, which when exceeded cause oxidation or reduction. At these limiting temperatures, unprotected thermocouples cannot be used. When the Curie point for nickel is exceeded, there is no effect on the resulting thermoelectric voltage as contrasted to the effect on resistance change with temperature, since the former depends on the electron concentration. Figure 35 gives representative values for the thermoelectric voltage as a function of temperature difference for the most often used thermocouples over the range where they are primarily used. The actual values for thermocouples from different companies vary somewhat, depending on the material used and the way it is treated.

3.1.2 Construction of Thermocouples

Just as in the case of the resistance probes, different requirements (time constant, vibration resistance, corrosion resistance, etc.), or their relative importance in a given application, necessitate different construction techniques for thermocouples.

*Exceptions are discussed in Section 3.2.

TABLE 4. THE MOST IMPORTANT THERMOCOUPLES USED IN AIRCRAFT

Type	Designation	Composition	Range/Accuracy	Characteristics
Type E (ISA Std.)	Chromel (+)/ Constantan (-)	90% Ni 10% Cr/ 57% Cu 43% Ni	-18 to $+315^{\circ}$ C/ $\pm$ 1.8°C +315 to +870° C/$\pm$ 0.5%	High e.m.f. (Parasite e.m.f.)
Type K (ISA Std.)	Chromel (+)/ Alumel (-)	90% Ni 10% Cr/ 94% Ni 3% Mn 2% Al 1% Si	-18 to +276°C/ $\pm$ 2.2°C +276 to +760°C/ $\pm$0.5%	Suitable for oxidizing or reducing gases
(a)Type J (ISA Std.)	Iron (+)/ Constantan (-)	Fe / 57% Cu 43% Ni	-18 to +276°C/ $\pm$2.2°C +276 to +760°C/ $\pm$0.5%	Inexpensive, parasite e.m.f. above 530° oxidation to 760°, reduction above 870°
Type R (ISA Std.)	Platinum (-)/ Platinum-Rhodium (+)	Pt / 87% Pt 13% Rh	0 to +1,000°C/ $\pm$0.25%	Used in older equipment, calibration deviates from newer type S
Type S (ISA Std.)	Platinum (-)/ Platinum-Rhodium (+)	Pt / 90% Pt 10% Rh	+450 to +1,500°C/ $\pm$0.25%	Highest accuracy, stability and freedom from parasite e.m.f., small thermo e.m.f., heavy absorption above 1,000°
Type T (ISA Std.)	Copper (+)/ Constantan (-)	Cu / 57% Cu 43% Ni	-200 to +93°C/ $\pm$0.8°C	Especially suited for negative temperatures, copper oxidizes above 350°C

For further information see: Instrument Society of America ISA Recommended Practice RP 1.1-1.7; VDE/VDI-3511; DIN 43710; etc.

(a)For purposes of comparison; seldom used in aircraft.

Figures 36a through d show "open" elements (exposed hot junctions). The first one (Figure 36a) is made of a single lead thermocoaxial cable; the center conductor is constantan wire and the outer conductor is copper tubing. More typically two insulated wires are used, and they are joined at the hot junction by twisting or better yet, by welding or silver soldering. When measurements must be made in a medium that is moving at high speeds, or if the element must be mechanically protected, an open element with a protective shield (Figure 36g) is used. In all the previously mentioned types, the medium to be measured is in direct contact with the thermocouple so that as a result of the small mass, very small time constants are obtained.

For use in media at high pressures, or if the medium is corrosive or an electrical conductor, the thermocouples must be housed in sealed tubes (Figures 36e and f). In these well-type elements, the wires are often welded to the tube and hence grounded at the contact point (Figure 36e). This normally excludes any arrangement of several elements in parallel (for indication of mean temperature with a single instrument), but yields smaller time constants than the well-type elements with an ungrounded junction shown in Figure 36f. Examples of another type of open element, the so-called foil thermocouples that are used for surface temperature measurements, are shown in Figure 37. They are made of flat ribbons, and are provided with temporary carriers of polyamide film. These elements are directly attached to the surface with a ceramic cement and are sometimes sprayed with a protective ceramic coating.

3.1.3 Construction of Probes with Thermocouples

If a thermoelement is equipped with a threaded or bayonet fitting, some other type of fitting, or even a housing, the arrangement is usually called a thermocouple probe. In contrast to the resistance probes which are usually provided with an electrical connector, the thermocouple probes have lead wires without terminals, eyes or even connectors with pins made of the same material as the element.

Depending on the purpose for which they are used, there are numerous types of probe construction, so that only examples of the most important types can be given here. Probes for measurements in liquids and gases are usually cylindrical in shape (Figure 38a), and the threaded fitting is often manufactured with a clamp so that the immersion depth of the element can be adjusted (Figure 38b). The element itself may be open, equipped with a shield (Figure 38a) or closed (Figure 38c). Figure 38d shows an open model with two separate thermocouples, one of which can be used for individual measurements, and the other for obtaining average temperature values when connected with other elements.

For measurements in a rapidly moving gas stream, the gas temperature probes (TAT or stagnation probes) shown in Figures 39 and 40 are used. They are always open elements and the housing usually has large intake openings, but small exhaust openings, so that the entering gas stream is adiabatically compressed. Figures 39a and 39b show simple models; Figure 39c, however, shows a model that is better suited for measurements in high speed gases, e.g. exhaust gas temperatures. In order to measure the outside air temperature in supersonic aircraft, where stagnation temperatures can be +150°C or higher, special types must be used. Figure 40 shows a model with two concentric radiation shields in the housing which significantly reduces the otherwise relatively large radiation error.

Finally, Figure 41 shows various probes for measuring temperature in solid bodies and on surfaces. The one in Figure 41a is a special spring-loaded model for measuring the temperature of the cylinder head of an aircraft engine. The spring presses the enclosed probe element against the bottom of a bored hole in the cylinder head. Figure 41b shows a semi-open probe with a threaded fitting designed to measure the temperature of brake drums. Figure 41f shows a clamp type probe which can be mounted on a pipeline. The gasket probes shown in Figure 41c and d are mounted like gaskets under spark plugs or screws. Figure 41e gives a front and side view of a probe that is riveted into a hole in the surface whose temperature is to be measured.

At this point, it must be mentioned that there are a number of other special types of probes, e.g. those that are more applicable for directly measuring the heat flux from and to surfaces, rather than for temperature measurements (see References 91 through 94).

3.2 Connecting Wires and Extension Leads

In Figure 42a, two leads x and y made of different materials are shown, which are connected to each other at points P_1 and P_2 by welding, twisting, etc. When P_1 and P_2 are at different temperatures (T_1 and T_2), a current is generated that is proportional to the temperature difference. The direction of the current depends on whether T_1 is greater or less than T_2. A temperature difference, therefore, produces an electromotive force (thermal e.m.f.), if this temperature difference occurs between two points of contact of different metals. Temperature gradients anywhere within lead x or lead y will have no effect if each is made of a homogeneous material*.

In thermocouple circuits, symmetrical connection points (or even lines) of a different material can be arranged differentially without consequence as long as these points (or the connection points of the lines) are at the same temperature (i.e., as long as the voltages that are generated at the points where the two different materials are connected are cancelled). Hence, if one of the conductors shown in Figure 42a is broken and a millivoltmeter inserted (Figure 42b) with leads made of material z (e.g., copper) and both points of contact (P_3 and P_4) of the broken lead are kept at the same temperature, there is no effect on the thermal e.m.f., although two new - but differentially arranged - thermocouples P_3 and P_4 have been formed. Their common temperature T_3 can be different from T_1 and T_2 and even from the temperature at the instrument, since in the latter case the temperature gradients appear in similar leads (both of material z). If however, points P_3 and P_4 were at different temperatures, their thermoelectric voltages would no longer cancel out and the resulting thermoelectric voltage of the circuit would be equal to the algebraic sum of all the individual voltages. This is equally true for a point P_5, which per se must have a single temperature, and it can also be considered a differential connection, y'/P_5 to P_5/y. This circuit (Figure 42b) is useful for measuring temperature differences or rate of temperature changes** (cf. sect. 2.2.3.2).

Usually, it is desirable to directly measure the temperature T_1 at point P_1, in which case the circuit shown in Figure 42c is better for several reasons. Instead of opening one lead and using two thermocouples, point P_2 is opened and the instrument is connected to points P_2 and P_3. Again, two new differentially arranged thermocouples have been formed, and their temperature must be equal. If this temperature T_2 is kept at $0^{\circ}C$ as a reference temperature, using for example a container with melting ice, then the temperature difference ΔT between T_1 and T_2 would equal T_1 in $^{\circ}C$. If on the other hand, a thermostat is used to keep T_2 at $+40^{\circ}C$, then $\Delta T = T_1 - 40^{\circ}$. The indicator scale must then be shifted by $+40^{\circ}C$ so that instead of the temperature difference, the temperature $T_1 = \Delta T + 40^{\circ}C$ is directly indicated. In aircraft, another type of reference point formation is often used (see below).

When the leads of the thermocouples are short and relatively high temperatures are to be measured, e.g. temperatures in an engine, it would be difficult to maintain a constant temperature at the junctions P_2 and P_3 because of the large fluctuating temperature gradients between the engine and the indicator. In this case, points P_2 and P_3 have to be moved, using so-called <u>extension leads</u>, to a more favorable position with points P_{12} and P_{13} as reference points (Figure 43). The extension leads x' and y' must, however, be made of the same material as x and y, otherwise additional thermoelectric voltages are generated. In other words, Chromel-Alumel thermocouples are lengthened with Chromel-Alumel leads. The tie lugs at P_2 and P_3 can be made of any material w and can be at any temperature with respect to T_1 and T_3 (and also with respect to each other). It is essential, however, that the junctions x/w and w/x', as well as y/w and w/y', are at the same temperature, so that there are no parasitic thermoelectric currents. This condition does not always hold true in plug type connectors or in junction boxes, and it is advisable to use special inserts of thermoelectric material, or a parallel arrangement of the input and output leads for approximately 10 cm and then twisting them so that they will be close to the same temperature.

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*Cf. below the effect of heterogeneous leads.

**Non-linear thermocouple output voltages must be taken into account.

Figures 43c and d illustrate rather drastically the effects of temperature gradients in the extension lead connections. The connection points P_2 and P_2', as well as P_3 and P_3', are assumed to be terminals that are connected with lead material w. In the extreme case of Figure 43c, the entire thermocouple x-y is at the same temperature and, hence, there is no thermoelectric voltage between P_2 and P_3. There is also no voltage between P_2 and P_2', or between P_3 and P_3', despite the large temperature difference (because of the two similar leads w), although there are voltages between P_2' and P_{12} as well as between P_3' and P_{13}. Figure 43d gives an extreme example of the reversal of current that can occur when the measuring point is cooler than points closer to the instrument (Reference 43).

When leads must be quite long, Chromel-Alumel thermocouples are often lengthened with copper-constantan leads to reduce circuit resistance, if the temperature of points P_2 and P_3 in Figure 43b is not more than approximately +65°C. This can result in a considerable reduction in weight where a large number of thermocouples are used, but at the expense of measurement accuracy. In the case of platinum and platinum-rhodium thermocouples, extension leads of copper or copper-nickel alloy are usually used for economic reasons. The thermoelectric voltage of these extension leads with respect to this thermocouple remains small within a limited temperature range.

Thermocouple circuits (in contrast to those utilizing resistance elements) are characterized by measurement errors that easily occur as a result of parasitic voltages or currents, which often cannot be detected by the indicator or are difficult to explain in the case of faulty indication values. Special attention must be paid to the correct polarity of the extension leads when the thermocouple circuit is wired. Parasitic currents are particularly troublesome when several thermocouples are connected to a single indicator through a selector switch, if the latter is not bipolar. The insulation resistance of thermocouples to ground is low at very high temperatures, and well-type elements with a grounded junction yield undefined conditions in the case of single-pole commutation.

Since the current in thermocouple circuits also depends on the resistance of all the leads (x, x', y, y', z, w, etc.) the calibration of a particular indicator (such as a millivolt-meter) is only exact for a certain total resistance value, to which the circuit can be adjusted by means of a trim resistance. This adjustment, however, only holds for the lead temperature existing at the time when the circuit was adjusted. The effect of the lead resistance is negligible in the case of currentless measurements taken with a servoed instrument (see below).

In thermocouple circuits, there are a number of other effects besides the Seebeck effect; e.g., the Peltier effect, the Thomson effect and the Joule effect (cf. Reference 42) are negligible when currents are relatively small. On the other hand, the Becquerel effect can be very large. Local material dissimilarities in the extension leads can generate significant parasitic e.m.f.s in the circuit when there is a temperature gradient (Reference 42) even when servoed instruments are used. For this reason, complete accuracy of the thermocouple circuit can only be achieved when the temperature distribution at the time of the calibration can be exactly reproduced in all the leads. This is almost impossible in aircraft. Errors can reach ±2°C, especially when leads are made of iron or constantan. For this reason iron-constantan thermocouples are rarely used any more in aircraft. Local deformations of thermocouples and leads can also produce heterogeneousness and must be avoided. Welding of thermocouple contacts is therefore better than twisting. When the depth of immersion of the element is changed, heterogeneousness caused by temperature can have a strong influence and for this reason the depth of immersion should not be changed at later stages.

Vibration effects can produce noise voltages of up to 100 microvolts in thermocouple-circuits with frequencies from 10 Hz to 5 KHz, dependent on the rate of rotation of the engines. These noise voltages are therefore equal to or greater than the useful signal at low temperatures (e.g. at the engine air intake point). Servoed indicators are usually equipped with an input filter, which filters out these voltages (and 400 Hz stray voltages). However, servoed instruments utilizing resistance probes are usually more accurate at temperatures below 200°C and cost the same (Section 5.3).

In the case of home-made thermocouples, the aging effect which changes the calibration after a specific period of operation must also be given consideration; thermocouples in commercial probes are usually pre-aged. In the case of open thermocouples, special attention must be given to chemical and physical effects which limit the life of the thermocouple at high temperatures by absorption of gases, reduction or oxidation, and other processes, depending on the material of the elements (cf. manufacturer's specifications).

3.3 Indicators for Thermocouples

3.3.1 Direct Indicators

In the case of all thermocouple indicators, the accuracy is directly dependent on the arrangement of the temperature reference point. In simple on-board aircraft instruments (Figure 44), the reference point is located at the compensated indicator (a moving coil millivoltmeter) so that the thermocouple extension leads run directly to the instrument housing. The positive connection is made of bronze for copper-constantan and chromel-alumel thermocouples, and of iron for iron-constantan thermocouples. The negative connection is always made of constantan. The wiring in the instrument utilizes iron or copper wire, and constantan wire. Since the temperature of the instrument, and thus the reference point, depends on the cabin temperature, the thermoelectric voltage will also change with cabin temperature. A bimetallic spring mounted on the moving coil device, and a resistance with a negative temperature coefficient in series with the copper coil, provide compensation for changes in the cabin temperature. Since the millivoltmeter requires a certain current, different lengths of extension leads would yield different scale calibrations. A trim resistance of constantan wire is connected in series with the negative extension lead, and is adjusted so that the total external resistance is that for which the instrument was calibrated (e.g., 8 ± 0.05 ohms). A measuring device of this type has, in itself, a permissible tolerance of ±10°C for a measurement range of 0 to 1,000°C.

The measurement accuracy of the above arrangement cannot, of course, be very high. Separating the temperature reference point from the instrument can result in considerable simplification of the design (since no special plugs are needed for the extension leads, etc.) and a considerable reduction in cable weight in large aircraft (since copper leads between the reference point and the indicator can be significantly smaller in diameter with the same resistance than, for example, chromel-alumel leads). Instead of a thermostatically controlled reference point (Figure 45), the use of a cold junction compensator (Figure 46) is preferred in aircraft. The latter utilizes a bridge circuit that is supplied with a regulated voltage source and consists of three normal resistors and a temperature sensitive, semiconductor resistor. The bridge output terminals are connected in series with the thermocouple and its extension leads, copper leads and indicator. The bridge output voltage is proportional to the reference point temperature and is used to compensate for the undesirable change in the thermoelectric voltage. Even in this thermometer circuit, the lead resistance must be adjusted to the value for which the indicator was calibrated.

Figure 47a shows a thermometer arrangement with several measuring points and a common cold junction compensator. The latter must be at the same temperature as the terminals of the thermocouples (or the terminals of the extension leads). This usually requires a considerable amount of copper wire between this point and the commutator. In this case especially, a separate trim resistance is needed for each thermocouple if exact measurements are to be obtained.

Figure 47b shows 4 thermocouples in parallel to measure an average temperature (e.g., turbine exhaust temperature). In this case also, separate trim resistances rather than a single resistance should be used for each thermocouple for maximum accuracy, since the internal resistances of the thermocouples can be different.

All methods that measure a current generated by the thermocouple, rather than the unloaded thermoelectric voltage, have the disadvantage that resistance changes in the leads (as a result of lead temperature changes during flight) can produce large measurement errors. The magnitude of the errors depends on the ratio of the trim and lead resistances to the other resistances of the circuit.

3.3.2 Servoed Indicators

The effect of lead resistances can be eliminated if measurements are made when no current is flowing, as in the "potentiometer" circuit shown in Figure 48. In this type of balanced bridge circuit, the thermoelectric voltage is balanced with a variable voltage across the diagonal branch of the bridge, the magnitude of which is manually adjusted by the potentiometer R_p. A temperature indicator is coupled to the wiper of the potentiometer. Since the characteristic curve of the thermocouple, i.e., the thermoelectric voltage as a function of the temperature, is non-linear over a large segment of the range, a non-linear potentiometer is usually used, and given segments of the temperature range can be expanded by changing bridge elements. The thermoelectric voltage and the voltage across the diagonal branch of the bridge are equal when the sensitive, series-connected meter is zeroed. Temperature measurements, however, are only correct if points P_{12} and P_{13} are at the reference temperature. The circuit shown in Figure 48 can be combined with the circuit shown in Figure 46 by replacing the resistance R_3 in the former circuit with a temperature sensitive resistance so that variations of the bridge temperature, which is also the reference temperature in this case, are automatically compensated. Since the type of temperature indicator just described requires a manual setting for each measurement, it is used almost exclusively as a test device. Accuracy is typically ±0.1% of the temperature measurement range.

In the case of a servo indicator, as described in section 2.2.5, the zero indicator in Figure 48 is replaced by a servo circuit (Figure 49). The filtered DC error voltage is converted into an AC voltage by means of a chopper and is then amplified and fed to an AC motor. The motor, through gearing, turns the wiper of the potentiometer R_p until the difference between the thermoelectric voltage and the bridge voltage becomes zero. At this point, no current flows in the thermocouple circuit, and hence, the lead resistances have no effect on the point of balance. The angular position of the bridge potentiometer (which is wound non-linearly to match the non-linear characteristic curve of the thermocouple) indicates the temperature at the thermocouple on a scale. In this type of temperature indicator, the temperature reference point is usually in the indicator housing for practical purposes, and thus, the extension leads run to the instrument. Only newer models with an input filter should be used in order to eliminate 400 Hz pickup and noise voltages caused by vibrations. Servo instruments usually have an accuracy of $\pm 1^{\circ}C$ between 500 and $1{,}000^{\circ}C$.

Figure 50 shows several common types of indicators for thermocouples. The series on the left utilize simple millivoltmeters and the series on the right are servo instruments. The latter are manufactured with expanded partial ranges, with two pointers, or with an additional counter wheel indicator to improve scale resolution.

4. HEAT TRANSFER AND TEMPERATURE RECOVERY

Every type of temperature measurement is based on the heat exchange processes between the temperature probe and the medium which is being measured. In addition, there is also the undesired heat exchange with other media which must be eliminated as much as possible. One characteristic of every type of heat transfer is the heat transfer coefficient. When taking measurements in rapidly moving media, there is also a temperature rise caused by the transformation of kinetic energy into thermal energy, the relative magnitude of which is described by the so-called recovery factor. Since these factors are functions of a large number of parameters, knowledge of the most important processes in heat transfer facilitates selection of the correct measuring points and the optimum type of measurement probe, and also helps in evaluating and interpreting the measurement results. The following brief discussions are intended only to introduce some of these factors. More detailed treatment of these subjects can be found in the following references: 1 through 13, 22 through 24, 44, 46, 48, 99A and 99B.

4.1 Modes of Heat Transfer

The following considerations hold, on the one hand, for the heat transfer between the temperature probe as a solid body and the medium which is being measured, be it a gas, liquid or solid. On the other hand, they also hold for the heat transfer between the medium and the structure of the aircraft, in which case, the temperature of the aircraft structure can influence the temperature of the medium to be measured (e.g. the outer skin can affect outside air temperature and piping can affect oil temperature.)

Heat transfer within a medium, or between two different media, can take place in the following ways (Reference 99B):

(a) Through heat conduction: This is a transfer of kinetic energy from high energy molecules to low energy molecules within a medium or between two media in direct physical contact. There is also a material exchange by molecular diffusion between two gases or liquids which corresponds to Brownian movement.

(b) Through convection: This is a type of heat transport by relocation of microscopic particles and is based, in the case of natural convection, on a difference in density resulting from temperature differences. In the case of forced convection, however, the flow is the result of mechanical influences.

(c) Through radiation: This is a heat transfer process without a material carrier, but by means of wave energy, primarily through infrared radiation. It can pass through transparent media, but can also be reflected at the boundary surfaces, or be absorbed by the media and thus be reconverted into heat energy.

All three types of heat transfer can occur either individually or combined.

Table 5 shows the connection between the most important terms in thermal technology which can be inserted into the formulae with the units referred to the quantity of heat or work (Joules) or referred to the electric power (Watts). In order to convert from the thermal system of units (calories) to the electrical system and vice versa, the thermal equivalent 1 cal = 4.185 Wsec = 4.185 J can be used. In the case of heat transfer, a specific thermal power P equals the heat quantity Q per unit time t that is transported within a substance or from one substance to another. The transferred heat quantity increases when the thermal conductance G is increased or the thermal resistance R (equal to $1/G$) is decreased, for a specific temperature difference ΔT.

The magnitude of the conductance value in this case is not only conditioned by geometrical dimensions (length, area, etc.) and the material properties (thermal conductivity, specific heat, etc.) but primarily by the type of heat transfer and energy increase (energy of motion, heating power, etc). Table 6 gives the general equations for the heat transfer processes.

4.2 Heat Transfer between Immobile Materials

4.2.1 Heat Transfer between Solids

The heat transfer between two solids, which are directly adjacent to each other and are surrounded by a single extremely good layer of thermal insulation occurs only through conduction and is dependent on:

(a) the temperature differential ΔT of both bodies.

(b) the geometric dimensions (length L and diameter d, etc.).

(c) the specific thermal conductivity k of both adjacent surfaces.

Heat transport within a material (e.g. along a metal rod) with a cross area A is described by the equation:

$$Q = \frac{k}{L} \cdot A \cdot t \cdot \Delta T \tag{6}$$

The heat transfer between two solids (1) and (2) on the other hand, must pass through the series arrangement of the thermal resistance of the material (1), the contacting surface (s_b) and the material (2). Thus, we obtain the following equation:

$$Q = \frac{1}{\frac{d_1}{k_1} + \frac{d_b}{k_b} + \frac{d_2}{k_2}} \cdot s_b \cdot t \cdot \Delta T \tag{7}$$

The thermal resistance is heavily influenced by the degree of contact between both solids (e.g. adjacent position, welding, soldering or cementing, possibly with metallic additive to the cement) and also by the presence of some sort of interference, e.g. air, oxide or other intermediate layers. When there is no common insulation around both solids, a heat exchange of each solid with the atmosphere occurs through conduction, convection and radiation, (see below). Calculation of the processes in all these cases is circumstantial and inexact.

TABLE 5. TABLE OF MEASUREMENT UNITS

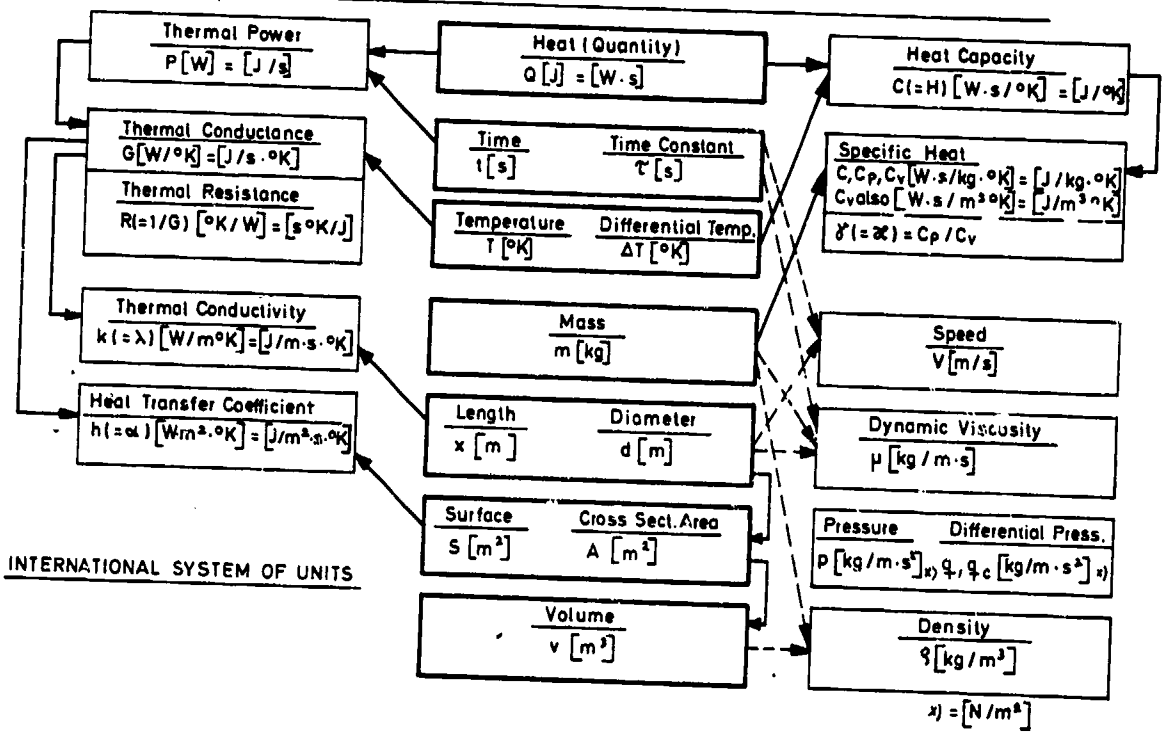

TABLE 6. GENERAL EQUATIONS OF HEAT TRANSFER

Heat (quantity)
$$Q = c \cdot m \cdot \Delta T = c_v \cdot v \cdot \Delta T = C \cdot \Delta T$$
$$= h \cdot S \cdot \Delta T \cdot t = G \cdot \Delta T \cdot t = P \cdot t$$

Thermal conductivity = 1/thermal resistance
$$G = h \cdot S = 1/R$$

Thermal power
$$P = Q/t = h \cdot S \cdot \Delta T = G \cdot \Delta T = (1/R) \cdot \Delta T$$

Differential temperature
$$\Delta T = T_1 - T_2$$

Differential temperature, kinetic heating
$$\Delta T = T_1 - T_2 = r \cdot V_t^2/(2g \cdot J \cdot c_p) = T_2 \cdot r \cdot (0.5)(\gamma - 1) \cdot M^2$$

Conduction, heat transfer
$$Q = h_C \cdot S \cdot \Delta T \cdot t = (k/d) \cdot S \cdot \Delta T \cdot t$$

Natural convection, heat transfer
$$Q = h_n \cdot S \cdot \Delta T \cdot t = [K_n \cdot (\Delta T)^{1/4}] \cdot S \cdot \Delta T \cdot t = K_n \cdot S \cdot \Delta T^{5/4} \cdot t$$

Forced convection, heat transfer
$$Q = h_c \cdot S \cdot \Delta T \cdot t = [(k_f/d_b) \cdot Nu] \cdot S \cdot \Delta T \cdot t$$
$$= (k_f/d_b) \cdot b \cdot Re^n \cdot Pr^{0.31}$$

Radiation, heat transfer
$$Q = h_R \cdot S \cdot \Delta T \cdot t = K_R \cdot \epsilon \cdot \sigma \cdot S(T_1^4 - T_2^4) \cdot t$$

Combined heat transfer (parallel)
$$Q = (h_D + h_c + h_R) \cdot S \cdot \Delta T \cdot t$$

Combined heat transfer (in series)
$$Q = [1/(1/h_1 + 1/h_2 + 1/h_3)] \cdot S \cdot \Delta T \cdot t$$

Time constant
$$T = C/G = R \cdot C = (m \cdot C)/(h \cdot S)$$
$$= (\rho_f \cdot c_b \cdot d_b)/(4 \cdot h_c)$$

Nusselt number
$$Nu = b \cdot Re^m \cdot Pr^{0.31}$$

Reynolds number
$$Re_x = (\rho_f/\mu_f) \cdot V_f \cdot x_b \;;\; Re_d = (\rho_f/\mu_f) \cdot v_f \cdot x_b$$

Prandtl number
$$Pr = c_{pf} \cdot \mu_f/k_f = 4\gamma/(9\gamma - 5) \approx 0.72$$ for air and exhaust gas

Indices:
- $( )_1$ State 1 or Object 1
- $( )_2$ State 2 or Object 2
- $( )_D$ Conduction
- $( )_R$ Radiation
- $( )_n$ Natural convection
- $( )_c$ Forced convection
- $( )_f$ Gas or fluid
- $( )_b$ Solid body

4.2.2 Heat Transfer between a Solid and an Immobile Gas or Liquid

In this case, two types of heat transfer are possible: through thermal conduction and/or radiation. When measurements are taken in immobile liquids or gases (moderate and high density), heat transfer through direct thermal conduction between the probe and the liquid plays a primary role, i.e. this case corresponds to that of heat transfer between two solids (see above). Especially in the case of gases (because of the lower thermal conductivity of the gas) the response time of a probe for temperature variation can reach 15 to 45 minutes!. Furthermore, heat transfer through the support structure and the lead wires of the probe to, or from the wall to which it is attached can assume significant proportions. When measuring temperature in a rarified gas, the heat transfer through radiation from the temperature probe to the adjacent wall assumes major proportions, in which case, the temperature of the gas which is being measured may have little effect. An important characteristic of heat transfer primarily through radiation is the relationship between the temperature level and the temperature differential: probe temperature and wall temperature are in this case both to the fourth power. The heat transfer in a vacuum through radiation alone can be described by the following equation:

$$Q = K_R \cdot \epsilon \cdot \sigma \cdot s \cdot t(T_1^4 - T_2^4)$$

where K_R is a form factor which is dependent on the geometric dimensions, ϵ is the emissivity, σ the black body radiation constant and s the radiating surface. Measurements in immobile gaseous or liquid media in aircraft, however, are an exception, since natural convection automatically is formed when there are appreciable temperature differences.

4.3 Heat Transfer between a Solid and a Flowing Gas or Liquid

The time which is needed for a termperature probe to reach the equilibrium temperature is dependent on the heat transfer between the medium and the probe.

The heat transfer in this case occurs through natural or forced convection, which also includes direct thermal conduction, and also through radiation.

4.3.1 Velocity and Temperature Boundary-Layers

When a gas or a liquid flows over a solid surface (e.g. a temperature probe), a boundary-layer is formed in which the velocity decreases from the undisturbed value to zero at the walls. In other words, velocity gradients occur. This velocity boundary-layer impedes direct contact between the flowing media and the surface of the solid so that a temperature difference between the undisturbed flow and the temperature probe also produces temperature gradients, i.e. a temperature boundary-layer. The conditions are described in the boundary-layer theory of Prandtl and the similarity theory of heat transfer of Nusselt. The Nusselt number (Nu) provides a correlation of different shapes of the solid as a function of the thermal conductivity of the liquid or the gas.

The thickness of a boundary-layer is determined by a series of mutually influential parameters, e.g. local flow conditions and temperature differences. The gradients for velocity and temperature are distributed in the same way only when the Prandtl number (Pr) equals 1. Otherwise, the ratio of the thickness of the flow boundary-layer (D_s) to the temperature boundary-layer (D_T) equals the square root of the Prandtl number.

Medium	Pr	$\sqrt{Pr}$	D_s/D_T
Air	0.72	0.85	0.85/1
Fuel	13.6	3.8	3.8/1
Oil	9000	95	95/1

In air and exhaust gas, the temperature boundary-layer is slightly thicker than the flow boundary-layer. In oil, on the other hand, the temperature boundary-layer amounts to only approximately 1% of the flow boundary-layer. This means that for temperature measurements, the position of the temperature probe within the flow can heavily influence the measurement results.

The heat transfer between a solid body and a flow must occur across the boundary-layer. The thermal resistance of the boundary-layer R_{bl} $1/(h_c \cdot s)$ depends on several parameters, including:

(a) the geometric dimensions of the body in the flow,
(b) the thermal conductivity of the flowing media,
(c) the Reynolds number or the type of flow (laminar or turbulent),
(d) the Prandtl number,
(e) the temperatures of the body and the flowing media,
(f) the type of convection (natural or forced),
(g) pressure (in the case of a gas),
(h) Mach number

4.3.2 Heat Transfer by Natural Convection

In the case of natural convection, a flow is generated in an originally immobile liquid (or gas) at a solid (wall or temperature probe) when there is a temperature difference between both (e.g. self-heating of the probe). This process occurs very slowly, depends on the magnitude of the temperature differential, and seldom reaches a steady-state condition. The direction and rate of the flow and the thickness of the boundary layer are influenced by the configuration of the surrounding walls, their insulation and a number of other factors which are very difficult to determine.

The dependence on the given temperature difference ΔT between the solid and the gas or liquid is characteristic of natural convection. The heat exchange occurs primarily by conduction since the velocity of the gas or the liquid can only reach relatively small values and the flow around a temperature probe can be considered laminar.

The heat transfer coefficient for natural convection is:

$$h_n = K_n \cdot (\Delta T)^{1/4} \tag{9}$$

where K_n is a constant that is a function of the heat transfer value, velocity distribution, geometric conditions, etc. The heat transfer is:

$$Q_n = K_n \cdot s \cdot (\Delta T)^{5/4} \cdot t \tag{10}$$

Temperature measurements with natural convection (e.g. compressor intake or exhaust temperature measurement on an engine which has just been cut) are very rare and are only meaningful if temperature probes with very small time constants are used together with a measuring technique where the self-heating of the probe is negligibly small. For further details about natural convection, the book of McAdams (Reference 12) can be used, which describes the influence of shape, etc.

4.3.3 Heat Transfer by Forced Convection

In the great majority of all temperature measurements in aircraft, heat transfer occurs by forced convection, in which case, the motion of the medium which is being measured is caused by the speed of the aircraft, the engine, a fuel pump, etc. The type of flow can be either laminar or turbulent, or separated from the body, depending on the given conditions. The velocity of the flow sometimes reaches such values that there is a considerable temperature rise when it impacts on a solid, and other effects must also be taken into account, e.g. the effect of compressibility in the case of gas.

4.3.3.1 Heat Transfer in a Subsonic Flow

In <u>laminar flow</u>, all of the flowing particles move longitudinally. The heat transfer in this case (within certain limits) is almost independent of the Reynolds number Re, the total pressure p_T and the Mach number M, and occurs almost completely by thermal conduction. When the rate of flow increases, i.e. when the Reynolds number becomes greater, there is an irregular lateral motion of the flowing particles in addition to the longitudinal motion, so that the adjacent flow layers are mixed. In this type of <u>turbulent flow</u>, the lines of flow only represent the average values of the paths of the flow particles. The transition from one type of flow to the other takes place sharply at a specific rate of flow (critical Reynolds number), but can be influenced by a number of different factors (References 99A and 99B). The local Reynolds number must be referred to as the distance x from the forward edge or to the diameter d of the solid (Re, x or Re, d). As the Reynolds number increases, i.e. as the velocity increases (or in gaseous flows when the temperature drops* and the

* In liquids, when the temperature rises.

pressure increase) the transition point moves forward. Since in the case of a turbulent flow, heat transfer is the result of convection as well as conduction, the heat transfer coefficient increases significantly. An example of this can be seen in Figure 51, where the change in total pressure corresponds at 36,000 ft. to a twelve-fold increase in Reynolds number, or an increase in Mach number of 0.2 - 2.4. In the transitional zone between pure laminar and pure turbulent flows, measurement values deviate greatly; in Figure 51 only the average values have been plotted.

The heat exchange which is high in a turbulent flow is usually calculated by taking into account the "effective values" for the viscosity and thermal conductivity (Reference 4).

The heat transfer coefficient for forced convection is usually described by the following equation:

$$h_c = k_f/d_s \cdot Nu = \frac{k_f}{d_s} \cdot a \cdot Re^m \cdot Pr^{0.31} \tag{11}$$

where the Nusselt number Nu is a function of the Reynolds number Re and the Prandtl number Pr. The factor a which depends on the given conditions (especially on the design of the body and direction of flow) and the exponent of the Reynolds number is best obtained experimentally. Figure 52 shows characteristic relationships between the Nusselt number and the Reynolds number which have been obtained for different thermometers and for different directions of flow (parallel flow or cross flow, References 5, 12, and 44). The given measurement values and formulae hold only under specific conditions.

In addition to the previously discussed ideal flow conditions, there is also an eddy flow aft a flow separation, which occurs for example in the wake of a cylinder hit by a cross flow, or behind solids which project from a plane surface in a parallel flow. Often, small modifications, e.g. changing the angle of sideslip, are enough to significantly alter the type of eddy formation. Heat transfer in these zones is in any case variable; one can obtain a statistic distribution of the measurement values which are often non-reproducible even with respect to their mean value. These conditions are often encountered on the skin of the aircraft and in the engines including intakes and jet pipes.

4.3.3.2 Temperatures of a High-Velocity Flow

The heating effect in a boundary-layer has not been previously discussed. It occurs as a result of the fact that the kinetic energy of directed motion is converted into molecular motion when the flow velocity is reduced, which also means a corresponding increase in the temperature of the decelerated medium.

The following considerations are valid for liquids* as well as for gases, which in contrast to liquids also have the property of compressibility. For this reason, in the case of gases, the true airspeed V_t must be used as a measure for the velocity, which, however, deviates from the value shown by the airspeed indicator depending on flight altitude and temperature, especially at high speeds.

For example, when a flat plate (or even a smooth cylinder) is inserted in a flow parallel to the direction of the flow, there is a temperature rise due to friction T_{kf} in the boundary layer. In a laminar flow (i.e. when the Reynolds number is small), this rise can be described by the following equation:

$$\Delta T_{kfl} = \frac{Pr^{1/2} \cdot V_t^2}{2g \cdot J \cdot c_p} \tag{12a}$$

* At temperature measurements in liquids there are measurable temperature rises due to friction at rates of flow (5 m/sec.max.) which are common in aircraft, only at very high Prandtl numbers, e.g. in the case of oil ($Pr = 9{,}000$). The temperature rise in this case is approximately 0.2°C so that even the large temperature dependence of the Prandtl number in liquids can be ignored.

and in a turbulent flow (i.e. when the Reynolds number is large) by:

$$\Delta T_{kft} = \frac{Pr^{1/3} \cdot V_t^2}{2g \cdot J \cdot c_p} \tag{12b}$$

where g = the gravitation of the earth, J = mechanical equivalent of heat, and c_p = the specific heat of the liquid.

Usually, however, the conditions of flow are not so well defined so that the resulting temperature rise is usually caused by a combination of different processes (friction, compression, etc.). It is usually called the recovered temperature rise and can be defined by the following equations:

$$\Delta T_{kr} = \frac{b \cdot Pr^n \cdot V_t^2}{2g \cdot J \cdot c_p} \tag{13a}$$

$$= \frac{r \cdot V_t^2}{2g \cdot J \cdot c_p} \tag{13b}$$

In other words, in the majority of cases, in contrast to the previously given equations for the frictional heating, the Prandtl number as a function of the probe design and the conditions of flow must be multiplied by a factor b, in which case, even the exponent may change (References 12 and 24). The term $(b \cdot Pr^n)$ in the numerator of equation 13a is usually designated the recovery factor r, since, in addition to the velocity, it usually determines the value of the resulting temperature rise in a specific medium.

The maximum possible value for the temperature rise in a compressible medium, i.e. a recovery factor of $r = 1.0$, is reached when the flow is completely reduced to zero, e.g. in a container open in the direction of the flow. One can then speak of adiabatic compression insofar as there is no addition or dissipation of heat during this process. The resulting full adiabatic temperature rise ΔT_k which occurs when there is complete conversion of the energy of motion into thermal energy and which is measured by a thermometer mounted in the center of the container is defined by the following equation:

$$\Delta T_k = \frac{V_t^2}{2g \cdot J \cdot c_p} \tag{14}$$

By conversion (Reference 101) the adiabatic temperature rise can be obtained as a function of the Mach number:

$$\Delta T_k = T \cdot \frac{\gamma - 1}{2} \cdot M^2 \qquad (\gamma = c_p/c_v) \tag{15}$$

This temperature rise reaches approximately 1.3°C at a speed (V_t) of 100 knots, and approximately 12°C at 300 knots. At 1,000 knots it is approximately 130°C.

According to equations (13a) and (13b), the recovery factor is a function of the Prandtl number which itself is not a function of the temperature and the pressure. The factor b and the exponent n, however, depend on the flow conditions in question. The following equation is valid:

$$r = f(V_t, Pr, Re, k, K \dots)$$

Therefore, for the most part, we have the same parameters as in equation (11) for the heat transfer coefficient for forced convection. The factor K takes into account the structural properties of the probe and the place where it is mounted. Closer investigation shows that correlation of the recovery factor and the Reynolds number is only possible for very simple probes. Figure 53 shows this for a flat plate (Reference 6) and a cylindrical temperature probe (References 5, 10, and 12) where there is a parallel flow which is a function of the Reynolds number with respect to the "characteristic length" x. It can be seen that in these cases, the recovery factor in a laminar flow (low Reynolds number = low speed) is approximately equal to $Pr^{1/2}$. In a turbulent flow, however, r approximates $Pr^{1/3}$ and is almost independent of the pressure and the speed. In the transitional zone, there are significant deviations (shaded areas), see also Reference 113. For probes in a cross flow, the recovery factor is much lower and can drop to 0.5, in which case, the minimum values are reached as the local speed of sound is approached (for cylinders in a cross flow, this corresponds to a Mach number of approximately 0.5). This is a result of the flow separation and vortex shedding. Figure 54 shows the local recovery factor on the surface of a cylinder as a function of the Mach number and the direction of flow. Note that there are negative recovery factors (to -0.2) on the rear of the body (Reference 5). Even more difficult to represent are the conditions when the body in the flow has sharp edges, where the flow separates, or where the flow is discontinuous or oscillates. In these cases, the inaccuracies in measurement are correspondingly greater.

Transferring the results obtained in the wind tunnel to temperature measurements in aircraft is only possible to a limited extent since in the latter case, the flow conditions are much more complex. For simple temperature probes which are built into the aircraft skin, and which take measurements in the flow boundary-layer of the aircraft, it must be taken into consideration that there is a conversion of the unperturbed flow in front of the aircraft into the local flow at the aircraft through a process of adiabatic compression at the nose and along the forward edge of the wing, etc. Added to these are the effects of lateral wind, propeller vortices, etc. Before the air reaches the temperature probe, its temperature is increased by compression, i.e. by the conversion of kinetic energy into thermal energy. A further increase results from the friction on the surface of the aircraft, in which case there is a temperature exchange between the surface of the aircraft and the air. This process can sometimes be counteracted by a drop in temperature when there is a drop in pressure, i.e. a process of expansion which corresponds to an increase in the local speed. There is again heating due to friction on the surface of the probe itself. At higher speeds, the material properties (density, viscosity, etc.) also change depending on the pressure and the temperature so that the flow boundary-layer and the temperature boundary-layer exerts mutual influence on each other. The velocity and pressure gradients which occur in the direction of the flow also have an effect on the measured magnitudes.

Figure 55 gives several mean values from different sources (e.g. References 7, 24, and 41) for the recovery factor of different types of probes in a parallel or cross flow as a function of the Mach number. Minor changes of the external design or measurement processes can result in considerable changes in the magnitude and the curve of the recovery factor. Also very significant are changes in the configuration and/or flight attitude of the aircraft, i.e. changes of the flow form at the probe, in which case inaccuracy in measurement quickly reaches intolerable values as the speed is increased.

In order to increase measurement accuracy, the temperature of a rapid air or gas flow must be measured outside the boundary layer of the aircraft with a method where the temperature rise remains essentially independent of the flow at the measuring element of the probe. This is the case when a housing is mounted in the air flow outside the boundary layer of the aircraft and is open in the direction of the flow and contains a measuring element in its center. This type of temperature probe, where there is (almost) full adiabatic compression of the air, is known as a total air temperature probe (stagnation probe). This designation is based on the fact that the sum of the static air temperature T and the adiabatic temperature rise ΔT_k is called the total air temperature (TAT):

$$T_T = T + \Delta T_k \tag{16}$$

$$= T + V_t^2/(2g \cdot J \cdot c_p) \tag{17}$$

$$= T \cdot (1 + \frac{\gamma-1}{2} M^2) \tag{18}$$

The total air temperature is always used as the reference temperature for all measurements in rapid air or gas flows, since it is directly related to the flight speed so that if necessary, it can also be used to compute the static air temperature.

However, because of the remaining inaccuracies, the recovered temperature rise ΔT_{kr} even in the case of TAT probes, is not exactly the same as the full adiabatic temperature rise ΔT_k, but usually amounts to 0.95 to 1.0 for the ratio $\Delta T_{kr}/\Delta T_k$. Using this ratio, the previously mentioned recovery factor can be more precisely defined:

$$r = \Delta T_{kr}/\Delta T_k \tag{19}$$

Since the recovery factor is that percentage of the value of the full adiabatic temperature rise ΔT_k which the probe can reach, and since a recovery factor of 1.0 is the full value of ΔT_k, its value is always smaller than or approximately equal to 1 in air and exhaust gas where the Prandtl number is approximately 0.7 and cannot exceed this value.

The recovery factor, for this reason, lies between approximately $r = 0.5$ and $r = 0.9$ for simple temperature probes mounted in the boundary layer of the aircraft and usually between $r = 0.95$ and $r = 0.99$ for TAT probes for measuring the outside air temperature. This latter value is reached by the interaction of the almost full adiabatic compression with the individual recovery factors of the element and shield and the extremely small heat losses which result from radiation and dissipation (Reference 44).

Using the recovery factor, on the other hand, a so-called recovery temperature T_r can be defined as the sum of the static temperature T and the recovered temperature rise ΔT_{kr} according to the following equation:

$$T_r = T + \Delta T_{kr} = T + r \cdot \Delta T_k \tag{20}$$

$$= T + r \cdot V_t^2/(2g \cdot J \cdot c_p) \tag{21}$$

$$= T\ (1 + r\ \frac{\gamma-1}{2}\ M^2) \tag{22}$$

The recovery temperature was formerly a very important factor since the low recovery factor of older types of probes was the source of the greatest error in measuring the outside air temperature so that the other sources of error seemed minor. On the other hand, when TAT probes are used, the recovery temperature is only one of several moderate sources of error.

4.3.3.3 Temperature Measurement in a Supersonic Flow

The laws of motion of a compressible gas in a subsonic flow are very diverse in comparison with those which apply to the supersonic range. When the local speed of sound is exceeded, a shock wave forms in front of the tip of the body in the flow (nose, TAT probe, forward wing edge, air intake edge, etc.). The static pressure, the air density and the static air temperature increase behind the shock wave; the total pressure, the Mach number and the airspeed, however, are less behind the shock wave than in front of it. The total temperature, however, does not change. Thus, the equation for the total air temperature remains completely valid for the supersonic range:

$$T_T = T \cdot [\ 1 + (\ \frac{\gamma-1}{2}\ )\ M^2\ ] \tag{23}$$

The distance between the shock wave and the body in the flow diminishes as the speed is increased. The total air temperature increases at the solid top surface of the body as a result of compression, and then decreases along the body in the boundary layer. The values obtained in practice, however, are only 85 to 92% of the calculated values since as the altitude and total air temperature are increased, heat transfer as a result of radiation also increases. The thickness of the boundary layer along the lateral surfaces of the body in the flow increases almost proportionally with the speed, which is primarily the result of the increase in volume caused by heating (Reference 99B).

In the case of simply constructed probes with very small measuring elements (e.g.open thermocouples or open thermistors), it is possible that in supersonic flow, the measuring element may penetrate the shock wave, which forms around its somewhat thicker housing. In an analogous manner, the shock wave can also enter the opening of a TAT probe housing at angles of attack of more than approximately 30°, so that sometimes the radiation shield penetrates the shock wave. A sharp drop in the recovery factor is observed in both cases (cf. Section 5.2.2, Figure 67 and Reference 41).

During supersonic flight, the air enters a TAT probe, through the effect of the linear shock wave, at a subsonic speed, but at an increased temperature (Reference 3). The overall recovery factor which results from the combined effect of the recovery factor of the shock wave ($r = 1.0$) and the corresponding subsonic recovery factor of the probe, asymptotically approaches the value 1.0 as the speed is increased. This, however, only holds true when the probe is not mounted in a poor location, i.e., where the thickness of the boundary layer which increases with the speed, reaches or exceeds the height of the TAT probe. Depending on the aircraft design and the place where it is mounted, the shock waves formed at the nose and at the probe may interfere with each other at a specific speed. Inaccuracy of measurement can also result in this case.

The effect of variable properties of the gas, especially those of (vaporized) atmospheric moisture on the total air temperature, as well as the changes in the recovery factor caused by (liquid) water (in droplets)and ice are discussed in Section 5.2.

5. REFERENCE VALUES, ERRORS AND DEFINITIONS OF TEMPERATURE MEASUREMENT

5.1 Reference Values of Temperature Measurement

5.1.1 Static Temperature

When we speak of the "temperature" of a given material, normally we mean the so-called static temperature. Static temperature is defined as that temperature which the material has in a state of rest, i.e., without disturbing its temperature equilibrium by a probe which can either add or take away heat. It must, however, be noted that in most cases the static temperature of a material is not the same in all places, since the temperature exchange with the environment fluctuates greatly from place to place.

In many cases, differences in the heat exchange of this type are intentional, e.g. in the cooling systems of an engine, where heat is applied in certain places, and removed to as great an extent as possible in others. If measurements are taken in liquids in a so-called "dead" corner where there is practically no flow, one can note temperature layers by varying the depth of immersion of the probe. The individual values for these layers, however, are local temperatures and usually of no importance. What is required is a mean temperature, and for this reason, a point must be sought where there is optimum mixing of the liquid, and where a measurement will be of particular significance. Analogous considerations are also valid for temperature measurements in solids and on surfaces.

A special case is the measurement of static air temperature, which is disturbed by the very existence of the aircraft and by the more important fact that the probe is exposed to a moving air stream. The static air temperature has to be known for flight test and navigational purposes (true airspeed) and is calculated from the total temperature T_T, or the recovery temperature T_r and the Mach number as follows:

$$T = T_T/(1 + \frac{\gamma-1}{2} M^2) \tag{24}$$

$$\text{or} \quad T = T_r/(1 + r \cdot \frac{\gamma-1}{2} M^2) \tag{24a}$$

Normally $\gamma = c_p/c_v$ can be made equal to 1.40 so that the following reduced expression is obtained:

$$T = T_T/(1 + 0.2\ M^2) \tag{24b}$$

$$\text{or} \quad T = T_r/(1 + r \cdot 0.2\ M^2)$$

In large aircraft, the calculation is usually done automatically by air data computers. If there is an air log which directly gives the true air speed, conversion can be done according to the following equation:

$$\left.\begin{aligned} T &= T_T - V_t^2/(2g \cdot J \cdot c_p) \\ \text{or} \quad T &= T_r - r \cdot V_t^2/(2g \cdot J \cdot c_p) \end{aligned}\right\} V_t \text{ in m/sec} \tag{25}$$

Normally, even in this case, c_p can be considered a constant so that we obtain the following expression:

$$T = T_T - V_t^2/7592 \quad (26)$$

$$T = T_r - r \cdot V_t^2/7592 \quad (26a)$$

$\Big\}$ V_t in knots

The given equations are in general use and are sufficiently accurate as long as the air can be treated as an ideal gas, i.e. when the limits given in Section 5.3.1 are not exceeded.

5.1.2 Temperature Definitions Related to Static Temperature

Any accurate measurement requires the qualitative and quantitative coverage of all errors. Only then will it be possible to adjust the measured value by the corrections (these are the error values with inverted sign). Therefore, in order to achieve direct comparability of different measurements, a prescribed sequence of computing and correcting steps and an established designation of intermediate values must be used, meaning a certain systematic order and an established terminology. For all measurements with the exception of fast air and engine exhaust gas flows, temperature definitions are referred to the unperturbed temperature, that is, the static temperature T.

The indicated temperature T_i is the indication of a thermometer that still includes all the errors (see Section 5.2):

$$T_i = T + E_p + E_L + E_I \quad (27)$$

where

E_p = position error;

E_L = temperature lag error;

E_I = instrument error.

The basic temperature T_{ic} is the indication of a thermometer, corrected for instrument error:

$$\begin{aligned} T_{ic} &= T_i + \Delta T_{ic} \\ &= T + E_p + E_L \end{aligned} \quad (28)$$

where

ΔT_{ic} is the instrument error correction.

The corrected temperature T_{icl} is the indication, corrected for instrument error and temperature lag error:

$$\begin{aligned} T_{icl} &= T_i + \Delta T_{ic} + \Delta T_{icl} \\ &= T + E_p \end{aligned} \quad (29)$$

where

ΔT_{icl} is the temperature lag correction.

The calibrated temperature T_c is the indication of the thermometer, corrected for instrument error, temperature lag error, and position error, and hence is equivalent to the static temperature:

$$\begin{aligned} T_c &= T_i + \Delta T_{ic} + \Delta T_{icl} + \Delta T_{pc} \\ &= T \end{aligned} \quad (30)$$

where

ΔT_{pc} is the position error correction.

Since the temperature lag error is often so small that it can be disregarded, the following form can be found also:

$$T_c = T = T_i + \Delta T_{ic} + \Delta T_{pc} = T_{ic} + \Delta T_{pc} \qquad (31)$$

The mean temperature T_{mean} is the mean value for a measured object, found through readings taken at different points, where the object is subject to a predominant addition of heat at one point and to a predominant loss of heat at another point.

The local temperature T_{loc} is the temperature of a measured object that prevails at a certain point of the object.

The probe temperature T_p is the temperature assumed by the measuring element of the probe under given conditions and/or error sources.

The measured temperature T_m is the electrical output signal of a probe, corresponding to the probe temperature, that still contains all the errors with the exception of those components of instrument error which occur only in the connected indicator instrument (or telemetry module or recording instrument).

5.1.3 Total Temperature

As has already been discussed in Section 4, when measuring temperature in fast-moving air and gas flows, i.e., when measuring the outside air temperature (OAT), the compressor inlet temperature (CIT), and the exhaust gas temperature (EGT), sufficient accuracy can only be obtained if the total temperature is calculated using suitable temperature probes with internal adiabatic compression. Only the TAT gives a measure of the total energy level of the gas.

The adiabatic temperature rise ΔT_k of the air or the gas* produced by total conversion of energy of motion into thermal energy as a function of the Mach number M, or the true air speed V_t is obtained from the following equations:

$$\Delta T_k = T \cdot \frac{\gamma-1}{2} \cdot M^2 \quad (T \;\&\; \Delta T_k \text{ in } {}^oK) \qquad (32)$$

$$= T \cdot 0.2 \cdot M^2 \quad (\text{for } \gamma = c_p/c_v = 1.4) \qquad (32a)$$

$$= V_t^2/7592 \quad (\text{for } V_t \text{ in knots}) \qquad (33)$$

$$= V_t^2/26040 \quad (\text{for } V_t \text{ in km/h}) \qquad (33a)$$

The value of the actual recovered temperature rise T_{kr}, which is dependent among other things on the structure of the probe, is the product of the adiabatic temperature rise and the so-called recovery factor r according to the equation:

$$\Delta T_{kr} = r \cdot \Delta T_k$$

$$= r \cdot T \cdot \frac{\gamma-1}{2} M^2 \text{ etc.} \qquad (34)$$

The total temperature T_T, which is the sum of the static temperature T and the adiabatic temperature rise, equals:

$$T_T = T + \Delta T_k$$

$$= T(1 + \frac{\gamma-1}{2} M^2) \quad (T,\ T_T \;\&\; \Delta T_k \text{ in } {}^oK) \qquad (35)$$

$$= T \cdot (1 + 0.2\, M^2) \quad (\text{for } \gamma = 1.4) \qquad (35a)$$

$$= T + V_t^2/7592 \quad (\text{for } V_t \text{ in knots}) \qquad (35b)$$

$$= T + V_t^2/26040 \quad (\text{for } V_t \text{ in km/h}) \qquad (35c)$$

Since in practice the recovery factor is always smaller than 1.0, the following equations are valid for the recovery temperature T_r of the probe in the absence, or after correction, of all other errors:

$$T_r = T + \Delta T_{kr} = T + r \cdot \Delta T_k \quad (T,\ T_r,\ \Delta T_{kr} \text{ in } {}^oK) \qquad (36)$$

$$= T \cdot (1 + r \cdot 0.2\, M^2) \quad (\text{for } \gamma = 1.4) \qquad (36a)$$

$$= T + r \cdot V_t^2/7592 \quad (\text{for } V_t \text{ in knots}) \qquad (36b)$$

The given equations show that the adiabatic temperature rise and the recovered temperature rise can only be directly expressed in ^oK (or ^oC) when they are considered as functions of the true air speed V_t (Figures 56a, 57 and 58), since the latter is already a function of the static air temperature. If the

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* The material values, e.g. Prandtl number and Reynolds number, of air and engine exhaust gases with the same temperature and pressure can be considered to be approximately equal, so that the given equations are valid for both media.

temperature rise is plotted as a function of the Mach number M, a separate value is obtained for each static temperature (Figures 56b and 57). In this case, the temperature ratio T_T/T (or T_r/T) is best selected as the ordinate for precisely this reason (Figures 56b, right scale, and 59).

Figure 60 shows the total temperature as a function of the flight altitude and the Mach number under standard atmospheric conditions. In this diagram, the zones where rain is normally encountered or where icing can occur, are also indicated. When certain (altitude dependent) speed values are exceeded, the temperature of the aircraft approaches the total temperature, so that no more icing can occur, and the ice which has already formed, melts. Since the static air temperature in the atmosphere can deviate considerably from the values for standard atmospheric conditions, Figure 61 gives the possible maximum and minimum values for the total temperature as a function of Mach number and altitude.

When temperature measurements are made in rapidly moving air and gas flows, the total temperature must be used as the reference temperature for all corrections, since only measurements made with stagnation probes yield the required accuracy, and the total temperatures which occur on the surface of faster aircraft or in their engines are decisive for considerations of aerodynamics, maximum operating temperature, air density, mass velocity and other parameters. The recovery temperature in comparison is only one of the resulting intermediate values in the correction of the different errors.

5.1.4 Temperature Definitions Related to Total Temperature

The <u>indicated air temperature</u> IAT, or <u>indicated total air temperature</u> T_{Ti} is the indication of a thermometer that still contains all the errors:

$$T_{Ti} = T_T + E_p + E_L + E_I \qquad (37)$$

where

E_p = position error;

E_L = temperature lag error;

E_I = instrument error.

The <u>basic air temperature</u> BAT or <u>basic total air temperature</u> T_{Tic} is the indication of a thermometer, corrected for the instrument error E_I:

$$T_{Tic} = T_{Ti} + \Delta T_{ic} \; * \qquad (38)$$

$$= T_T + E_p + E_L$$

where

ΔT_{ic} is the instrument error correction.

The <u>corrected air temperature</u> or <u>corrected total air temperature</u>, T_{Ticl}, is the basic total temperature corrected for the temperature lag error, E_L:

$$T_{Ticl} = T_{Tic} + \Delta T_{icl} \; *$$

$$= T_T + E_p \qquad (39)$$

where

ΔT_{icl} is the temperature lag correction.

The <u>calibrated air temperature</u> or <u>calibrated total air temperature</u> T_{Tc} is identical to the total temperature T_T (total air temperature, TAT or stagnation air temperature), and results from the total temperature T_{Ticl} after correcting for the position error:

$$T_T = T_{Ticl} + \Delta T_{pc} \; * \qquad (40)$$

* Correctly, the corrections ΔT_{ic}, ΔT_{icl} and ΔT_{pc} should be written ΔT_{Tic}, ΔT_{Ticl} and ΔT_{Tpc}, but this is not yet common practice.

where

ΔT_{pc} is the position error correction.

Since the temperature lag error can be disregarded in most cases, the following form is also frequently found:

$$T_T = T_{Tic} + \Delta T_{pc} \tag{41}$$

The total air temperature T_T is equivalent to the sum of static air temperature (cf. below) and the full adiabatic temperature rise of the air T_k:

$$T_T = T + \Delta T_k \tag{42}$$

The full adiabatic temperature rise ΔT_k is the temperature rise of a gas flow as a result of the complete conversion of kinetic energy into thermal energy, provided that this process takes place without adding or removing heat, that is, in the form of adiabatic compression.

The static air temperature T (SAT)* is defined as the temperature of the unperturbed air at flying altitude. Since (unlike static atmospheric pressure) it can be measured directly only at very low air speeds, it is normally determined from the total temperature T_T after making the compression correction ΔT_{kc} (where the latter is equivalent in magnitude to the adiabatic temperature rise ΔT_k but has the opposite sign):

$$T = T_T + \Delta T_{kc} \tag{43}$$

$$= T_T - \Delta T_k \tag{43a}$$

The true air temperature TAT is a synonym for static air temperature used only by the U. S. Air Force (caution, normally the abbreviation TAT denotes total air temperature!).

The true total air temperature T_{Tt} is sometimes used to designate the correct value of the total temperature at flight conditions, where the actual γ deviates from 1.40 and γ_{eff} must be used in equation (35), cf. Section 5.2.1.

The local air temperature T_{loc} is defined as the air temperature at a certain point of the aircraft. However, depending on local flow conditions it can assume any value between static and total temperature.

The measured total air temperature T_{Tm} is the electrical output signal of a temperature probe. It still contains practically all the errors with the exception of those components of instrument error that occur only in the user or indicator instrument (scale error and the like).

The computed static air temperature is the electrical output signal of an air data computer. It results from correcting the computed total air temperature for the adiabatic temperature rise, and consequently, is equivalent to the static air temperature T (within the limits of computer accuracy).

Additional temperature definitions can be found in the literature that are used only as intermediate values when computing the corrections for the position error, or have become obsolete entirely. It should be kept in mind that, in the past, it was felt that the only important source of error in measuring air temperatures was the relatively low recovery factor r (cf. below) of the temperature probes** used at that time. All other components were mostly neglected (and unjustifiably so), such as temperature lag error and instrument error. This concept was retained for years after introduction of total temperature probes, although in that case, the individual error components must be assigned different weights.

The recovery temperature T_r is that temperature of a probe element which, in the absence of (or after correction for) all other potential errors, is determined exclusively by the prevailing magnitude of the velocity error E_V*** and the total temperature T_T:

* Also called free air temperature, ambient air temperature, or actual outside air temperature, OAT.

** These probes were generally installed in the boundary-layer of the aircraft.

*** This is the error originated due to a recovery factor not equal to 1.0 (See Section 5.2.).

$$T_r = T_T - E_V \tag{44}$$

Consequently, it is also equal to the sum of static air temperature T and the recovered temperature rise ΔT_{kr} (which is the original type of definition)

$$T_r = T + \Delta T_{kr}$$

The recovered temperature rise ΔT_{kr} is the temperature rise which takes place at a temperature probe upon incomplete stagnation of a gas flow as a result of friction and/or compression (and is always smaller than the full adiabatic temperature rise ΔT_k):

$$\Delta T_{kr} = r \cdot \Delta T_k = \Delta T_k - E_V \tag{45}$$

The factor r is called the recovery factor. In accordance with the stipulation made above (absence of any other error type), these equations in the case of older temperature probes* apply only for wind tunnel measurements, unless the so-called "flight test recovery factor" was used in place of r which allowed for additional errors and was determined through measurements on the aircraft. In these cases, the term recovery temperature was frequently used for the definition of basic temperature T_{Tic}, or indicated temperature T_{Ti} (cf. above).

The term ram air temperature RAT is used with different definitions in the literature. It can designate, for example, the recovery temperature T_r (for instance, ARINC) the indicated temperature T_i, the total temperature T_T, or even the adiabatic temperature rise ΔT_k.

The observed outside temperature formerly was another term used for basic temperature, T_{Tic}.

The indicated total absolute temperature T_{Ti} is defined in the ARINC specifications** as "indication of a total temperature indicator, corrected for all errors (including velocity error), but with the exception of instrument error and read-out error". However, the associated equation will yield the recovery temperature T_r of a widely used total temperature probe which, consequently, includes nothing but the velocity error.

5.2 Errors of Temperature Measurements***

Temperature measuring errors always result in the following cases:

(a) when the ideal heat exchange between the measured object and the measuring element is disturbed as a result of heat transfer to foreign objects or heat transfer from them;

(b) when a certain period of time is required to restore the temperature equilibrium after temperature changes of the measured object;

(c) when the material constants of the measured object show deviations from the customary standards as a result of extreme environmental conditions or admixture of other substances (such as water vapor and rain drops in air);

(d) when instrumentation conditions (such as improper calibration) cause false thermometer indications.

It is true that the conditions named under items a and b are determined in both cases by the type and magnitude of heat transfer, depending on location and design of the temperature sensor. However, a separation is made for procedural reasons between the group of the position errors and the temperature lag error which is encountered when the temperature of the measured object changes. Changes in material constants are usually significant only in air temperature measurements under extreme conditions (high total temperatures), in states near the saturation point of air with water vapor, or during intensive rainfall, and are then called meteorological errors. The errors that are exclusively due to instruments are called instrument errors, and this term is not confined exclusively to the indicator but includes the probe and the leads.

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* These probes were generally installed in the boundary-layer of the aircraft.

** ARINC characteristics 545, 565, 575, and 576.

*** This section is an expanded version of the corresponding chapter in Reference 100.

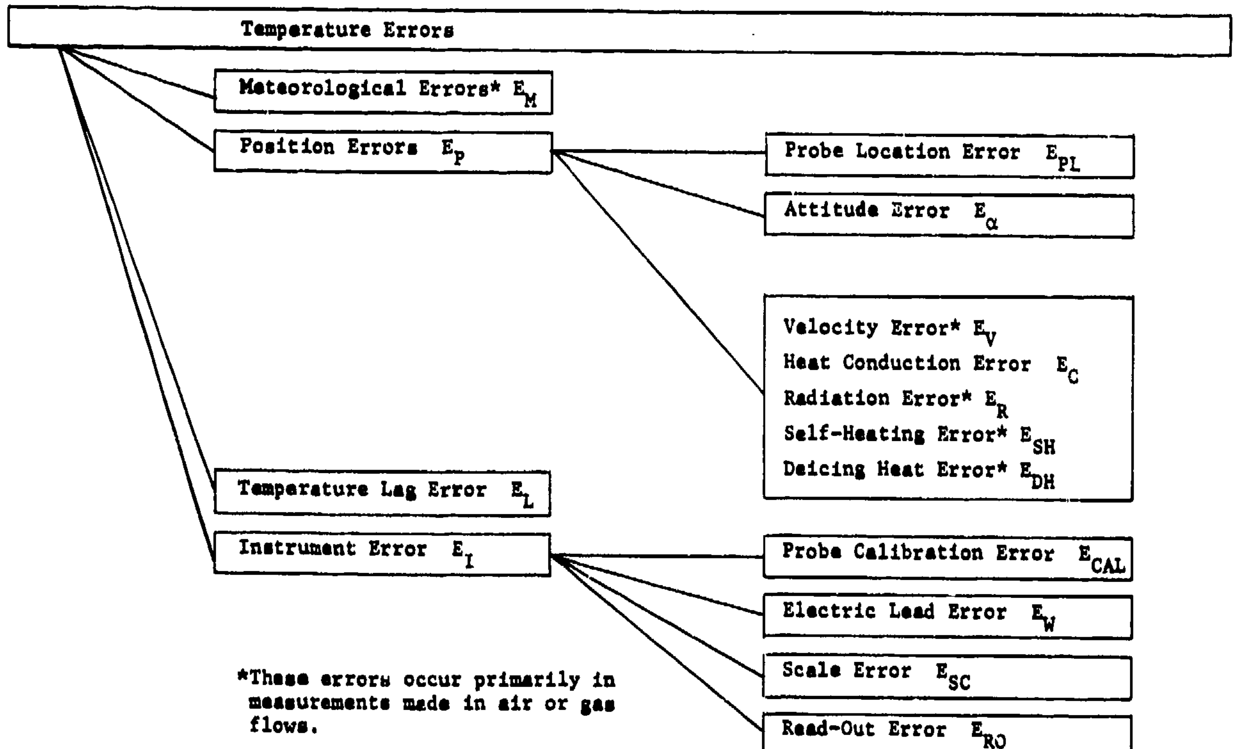
Table 7 provides an overall listing of the most important temperature measuring errors in an aircraft with an organization by error categories, but naming only the most important individual errors. Of course, their magnitude depends on the type of temperature measurement and the prevailing conditions. For instance, the errors designated by an asterisk must be taken into consideration only where readings are taken in fast air or gas flows. Many of the errors named in this table are avoidable errors if a certain expense can be incurred in selecting the equipment, and only a few of them are truly unavoidable errors. Incidentally, measuring errors can be considered to be independent of each other only where each error remains small in itself so that it does not appreciably affect the others (Reference 44). Exceptions to this rule will be pointed out in the text below.

Depending on the causes for their development, measuring errors are either systematic errors (with fixed sign +/-) where, however, a certain additional uncertainty or dispersion range with changing sign is present in most cases, or they are random errors where the sign can change from one to another (cf. the Section on "error analysis"). The corrections needed to correct the instrument readings are exactly equivalent to the values of the corresponding errors (or error categories) but have the opposite sign.

The errors that are encountered during temperature measurements in aircraft will be discussed in somewhat greater detail because little specific literature has become known on this subject. Yet, considerable simplifications have to be made in the presentation because these errors are always a function of a great number of (mutually dependent) parameters. Mathematical determination of these errors is quite difficult, since in most cases the composition of probe materials and the probe design are involved. Therefore, only general functional equations are listed in many instances. This applies particularly to air and gas temperature measurements especially where the measurements obtained from simple probes in the boundary layer are subject to flow phenomena that are difficult to acquire. However, diagrams are provided for some widely used probes* that can be used to achieve a fair estimate of the errors that can be expected for the prevailing flight parameters.

In the text below it was found best to discuss the errors on the basis of an outside air temperature measurement (where the engine inlet temperature and engine exhaust temperature readings represent variants only). In these cases, all errors are referred to the total temperature (T_T). Moreover, appropriate information will be provided on the other types of temperature measurements, but these must be referred to the static temperature (T).

TABLE 7. LIST OF TEMPERATURE MEASUREMENT ERRORS



* Based on manufacturers' information and measured data.

5.2.1 Meteorological Errors

Meteorological effects will be defined here as changes in the actual values of air or a gas, compared to the characteristics that apply for dry air or gas which are normally considered to be constant. However, where high total temperatures are reached this is not always valid. This applies especially where the humidity of the air at flying altitude is high. This error, considering its causes, will not be encountered in readings taken in liquids, at surfaces, and in solids.

Deviations from the ideal gas state* (thermal imperfections) can be disregarded here, since they only occur at altitudes and air speeds that can be reached only by space gliders and hypersonic research aircraft.

On the other hand, deviations of the $c_p/c_v = \gamma$ ratio from the standard value (caloric imperfections) may affect computations of the static air temperature T or the true air speed V_t from the Mach number M and total temperature T_T. Figure 62 shows that the value of γ depends on total temperature as well as total pressure and relative humidity. However, as we already indicated in the preceding chapters, the actual value of γ at a pressure of 1 atmosphere and a temperature of +20°C will be higher than the standard value of 1.40 in dry air, and lower in saturated air (at the same pressure and temperature). Therefore, this value represents a good compromise for normal conditions near the ground. This applies even for high altitudes, i.e., at lower pressures, since only lower temperatures are encountered there (and only very low humidity values).

Since temperature probes for ambient temperature measurements (unlike pressure sensors** for air speed or altitude measurements) are located directly in an air compression zone, we must take into consideration that the value of γ in the event of pressure changes is also a function of the compression or expansion rate, since the air molecules require a certain amount of time (relaxation time) in order to achieve temperature equilibrium. Therefore, the normal value of γ must be replaced by an effective value γ_{eff} whose profile has much less slope than the profile of the static values. As a result, we can use the standard value of 1.40 for practical calculations, as long as the total temperature does not exceed certain boundary values. These are about +160°C in dry air (corresponding to Mach 2.2 at high altitude), and only about +75°C in air with a relative humidity above 80% (corresponding to Mach 0.85 near the ground on a hot and humid summer day). At these boundary values the errors will amount to only about 2 parts per thousand of the prevailing total temperatures (measured as absolute temperatures in °K), but will rise sharply as the boundary values are exceeded.

In order to compute the effect of pressure and temperature on the effective value of γ, we must acquire a mean specific heat on the basis of gas enthalpy ($i = c_p \cdot T$) for the boundary values of T and T_T. Three cases (for dry air) are stated in Reference 2:

Altitude (ft)	Mach No.	γ_{eff}	$T_T - T_{T1.4}$	$M - M_{1.4}$	$T - T_{1.4}$
0	1.2	1.4011	+ 0.23°C	-0.00035	-0.14°C
50,000	2.3	1.3977	- 1.36°C	+0.00145	+0.51°C
100,000	3.0	1.3898	-10.68°C	+0.00815	+3.00°C

In this table the subscript 1.4 indicates that these temperatures and Mach number were determined for $\gamma = 1.40$. This means that, for a Mach number of 3.0 at 100,000 ft., the actual total temperature is 10.68°C lower than the temperature computed using the standard value of $\gamma = 1.4$. Consequently, the total temperature measured under these circumstances is a true value that is due to variable gas characteristics. Errors are encountered only where the static air temperature is computed from the actually measured total temperature with the aid of the standard value of $\gamma = 1.4$, or the abbreviated equation

$$T_{1.4} = T_T'/(1 + 0.2\, M^2) \tag{46}$$

---

* Ideal gas defined as $p = \rho \cdot R \cdot T$

** Diaphragm capsules, etc.

In these cases the prevailing effective value γ_{eff} must be substituted for γ in the comprehensive equation for the true total temperature

$$T = T_T/[1 + (\frac{\gamma-1}{2})\cdot M^2] \tag{47}$$

In our example, using $\gamma = 1.4$ would yield a static temperature that would be too low by 3°C. The profile of the γ_{eff} values is plotted in Figure 62a for two altitudes (near the ground and in the lower stratosphere) and for certain Mach numbers in accordance with the prevailing total temperatures (under the conditions of the standard atmosphere).

Very little clarification has been obtained on the effect of relative humidity on the ratio of specific heats c_p/c_v for the case of temperature measurements at high air speeds. Wind tunnel investigations are fairly difficult since oversaturation and hence condensation occurs even in unperturbed flows at all Mach numbers exceeding a certain value that depends on initial humidity. This value is located near $M = 0.8$.

The γ values for stagnant, saturated air (100% rel. hum.) are plotted in Figure 62a as functions of static air temperature and pressure. Again, γ_{eff} values should be used here because the temperature is increased upon air compression, but the relative humidity is decreased in proportion; their profile would have a much smaller slope. However, this is not confirmed in the only reference (39) accessible to the authors in which this problem is addressed directly.

Although the actual magnitude of the change in the γ_{eff} value is still a matter of dispute in the event of a coincidence of high static air temperatures and very high relative humidity, it can be considered established that, under subtropical conditions, considerable variations of γ will occur even in the high subsonic air speed range. It is true that this type of weather situation is relatively rare in the moderate latitudes, and that near-sonic air speeds at low altitudes are rarely flown by other than military aircraft. Yet it should be possible to acquire the γ variations qualitatively, since otherwise considerable errors would be incurred in the practical calculation of static air temperature. However, these errors will drop sharply as the temperature or humidity decreases. Since, in normal flying operations, high subsonic or even supersonic air speeds are flown only at higher altitudes where static temperature and especially the humidity are relatively low, the effect of humidity can be neglected in most cases.

The humidity content in the air, where the water occurs only in the vapor phase, should not be confused with its liquid water content (LWC) in the form of water droplets, snow, or ice crystals. It is true that drops of water impacting on the temperature sensor element will be deflected in the majority of cases. Yet the moment will occur, depending on the internal design of the probe housing, where large parts or all of the surface of the probe element are covered by a thin layer of water. While the air flow continues to carry new water drops to the element, it will, on the other hand, favor a continuous evaporation. Most of the thermal energy required for this evaporation will be removed from the sensor element, establishing a temperature equilibrium that differs from conditions prevailing in dry air. This so-called "wet bulb temperature" is less than the temperature which would be established in the absence of water but under otherwise equal conditions ("dry bulb temperature"). This magnitude depends on an extraordinarily large number of factors (such as relative humidity, ventilation rate, temperature level, difference between water temperature and static air temperature*, thickness and shape of the water film on the element, and others). If the temperature of the air entering the probe housing drops below the freezing point, the temperature of an element wet, for instance, by wet snowflakes, will also drop to a certain value that can be considerably below 0°C. If subsequently the still liquid (undercooled) water film freezes for any reason (such as shock, vibration, impact of an ice particle, etc.), heat will be released so that the sensor temperature will rise to 0°C and will remain there until all the water has frozen. Only then will the temperature of the element drop again slowly (depending on the ventilation rate). If the air flowing into the housing has a water vapor content that is located near saturation (above water)**, the excess water vapor will condense on

* Rain droplets cannot always assume, rapidly enough, the temperature of the air layers through which they drop.

** The vapor pressure above an ice layer is less than the vapor pressure above a water layer.

the ice layer and will increase it until the saturation level of air above ice is reached. This condensation of water vapor and freezing on the element will release heat that might cause a probe temperature which is located above the drop temperature (References 97 and 98). Since all of these phenomena take place in a compression zone*, and since all transitory states between liquid water and ice can occur, it is impossible to predict the deviation of temperature sensors wet with water droplets or wet snowflakes from its rated value.

The only established facts are that dry snowflakes unlike water droplets or wet snowflakes have only a small effect, and that the error magnitudes in the case of a probe design with internal air deflection are only a fraction of the error magnitudes encountered in probes where the measuring element is exposed directly to the air flow. As a result of this dependence on probe configuration the effect of liquid water is better treated as part of the velocity error (cf. below).

5.2.2 Position Errors

These errors are due to perturbances in ideal heat transfer between the measured object and the probe element and are, therefore, mainly functions of the heat transfer coefficient. This means that, in measurements in gas or air, they are functions of the flow type and the mass flowrate through the environment of the probe. The prevailing values of these errors, being dependent on flight parameters, are generally stated as functions of pressure, altitude H and Mach number M in the text below, since this will yield the simplest evaluation (at adequate accuracy).

(a) Probe location Error E_{PL}

In the case of measurements in gas or air, this error is a function of the enveloping flow at the probe location and will occur in the normal flight altitude (unlike the altitude error which is discussed below). For simple probe types located in the boundary layer its magnitude will depend on the local flow velocity which, as we know, can differ greatly from the air speed, depending on local conditions. This means, of course, that pressure gradients will occur. Consequently, the kinetic temperature increase ΔT_{kr} at the probe is no longer directly dependent on air speed but associated with it through effects that are difficult to acquire. On the other hand, depending on local velocity at the probe, the prevailing flow type (laminary, turbulent, or separated flow) will be subject to change, shifting suddenly from one type to another depending on the local Reynolds number, change in configuration or angle of attack or sideslip. This will have a direct effect on the recovery factor r (cf. Figure 53). These effects are encountered with special severity where, for instance, a surface probe is installed on a flat fuselage belly in conjunction with a close radiation shield. These facts, and the conditions outlined in Section 4.3, are partly responsible for the fact that wind tunnel measurements of simple probes are not valid for probes installed on aircraft, and that in this case it is hardly possible to isolate the probe location error from the other position error components.

In the case of total temperature probes the probe location error will reach a significant magnitude only where the probe is installed in a highly unfavorable location, for instance in the effective range of a propeller or within a thick boundary layer (cf. Figure 63). The location error can be acquired by comparison with a probe of the same type located at a point with ideal flow conditions (for instance on a nose boom).

Surprisingly, the value of total temperature is changed only little if the probe is located in the air intake of an engine** although the relationship between the flight Mach number and the local Mach number in the air intake is not a simple one (it is dependent, for instance, on engine rpm). Yet a thermometer for compressor inlet temperature (CIT) should not be used to compute the static air temperature, since the differences are too great for that. Figure 65 shows the gradual reduction of flow velocity in an air inlet during supersonic flight as a result of oblique and normal shock waves and as a result of the increasing cross-section area of the inlet. Although this is accompanied by increasing static pressure, the internal Mach number differs quite considerably from the flight Mach number, so that it is better to use the air flow-rate (mass velocity, for instance, measured in $kg/m^2 \cdot s$) as a unit of measure. Figure 62c shows an example for the mass flowrate of an engine as a function of the Mach number for given values of total pressure and total temperature.

* On which no specific literature is known.

** The same total temperature prevails at any point of the cross-section of a wind tunnel.

For rapidly moving gases, as in inlets, not only velocity gradients are found but temperature gradients as well, and their distribution will be dependent on a very large number of parameters (such as inlet geometry, air inlet velocity, temperature difference between wall and air, Reynolds number, Prandtl number and others). Consequently, even where a TAT probe (for a CIT reading) is installed on an inlet wall, we must keep in mind that in the event of fast climbing or descending flight the CIT indication may lag considerably behind as a result of the temperature exchange between the air and the inlet wall (lag error) until the wall has adjusted to the new temperature equilibrium. Therefore, in order to acquire accurate readings during flight tests, especially in the air inlets of VTOL aircraft, a number of temperature probes are distributed evenly over the cross-section (cf. Figure 66).

The probe location error can be defined approximately by:

$$E_{PL} = f(P_T, p, K_F, K_{PL}) \qquad (48)$$

meaning that it is a function of total pressure and static pressure at the probe location, the design characteristics of the probe (K_F), and the airflow characteristics at the probe location (K_{PL}).

Sometimes the term installation error is falsely used as a synonym for location error. Actually, however, the installation error designates only the difference between the location errors of identically equipped aircraft of the same production series. It is determined primarily by production tolerances which have the result that aircraft of identical appearance can have different flow relationships at the probe.

Where measurements are taken in liquids a considerable probe location error can develop if the probe is installed, for instance, at a "dead" point of the tank where no motion exists and where a temperature can develop that differs from the temperature of the remaining liquid. Likewise, a probe location error can develop during measurements on surfaces and in solid bodies, if an unfavorable location is selected for the probe where the desired heat exchange with the measured object is obstructed but where the heat exchange with foreign objects is favored as a result of this location.

(b) Attitude Error E_α

This error will occur only when the probe is exposed to a non-parallel flow. Consequently, relatively large angles of attack (α) and angles of sideslip (β) will cause this error in probes for air temperature measurements. The older probe configurations react quite clearly to angles of about $\pm 5°$, and in case of TAT probes with the deicing system in operation, an angle of $\pm 10°$ (often more than $\pm 20°$ if the deicing system is turned off) must be exceeded before the reduced local flow of the probe element appears as an air speed reduction, leading to a reduction in the temperature indication (cf. Figure 64). When the deicing system is turned on this is partially compensated or even overcompensated by an increase of the self-heating error and deicing heat error, cf. below.

The attitude error involved in TAT readings in fast aircraft is usually so small that it ca be neglected. However, in the case of extremely low-speed aircraft a strong crosswind can render the encircling flow of the aircraft highly instable. VTOL aircraft and helicopters frequently have very large angles of yaw and during vertical ascent or descent, even purely vertical local flows, that can have a high degree of vorticity as a result of recirculation effects, rotor effects, and the like. In that case, very large errors and large error dispersion must be expected in outside air temperature measurements unless an indicator with extremely low probe self-heating characteristics is used (and unless any probe deicing system that may be installed can be turned off).

The attitude error can be defined by:

$$E_\alpha = f(\alpha, \beta, K_\alpha, M) \qquad (49)$$

where

K_α represents a factor that is dependent on the probe design.

An attitude error during measurements in liquids occurs, for instance where, in the case of stem probes, the flow encounters the foot of the probe first, changes its temperature, and only then reaches the temperature-sensitive top of the probe. This type of error does not occur during measurements on surface and in solid bodies.

(c) Velocity Error E_V

This error which occurs only during measurements in fast air or gas flows states the amount by which the recovery temperature T_r of a probe remains below the total temperature T_T, which is equivalent to the difference between full adiabatic temperature increase ΔT_k and acquired temperature increase ΔT_{kr}:

$$E_V = T_T - T_r = \Delta T_k - \Delta T_{kr} \tag{50}$$

Before defining the dependence of the velocity error on air speed in more detail some additional terms must be explained:

(c-1) The recovery factor r is nondimensional and designates, as previously mentioned, the ratio of the temperature rise acquired by the probe to the full adiabatic temperature rise.

$$r = \Delta T_{kr}/\Delta T_k = (T_r - T)/(T_T - T) \tag{51}$$

$$\Delta T_{kr} = r \cdot \Delta T_k \tag{52}$$

In air and in engine exhaust gases it is always smaller than 1.0*, even though some instrument readings appear to contradict this - probably because other effects were not fully covered. In the case of simple probes installed in the boundary layer of the aircraft, the recovery factor will be between the limit values of approximately $r = 0.5$ and $r = 0.9$ (cf. Figure 55), where a measuring uncertainty between $\pm 0.03\ r$ and $+0.1\ r$ can occur if the airflow moves parallel to the probe surface. In the case of stem probes placed in a lateral flow, this instrument dispersion can be even considerably greater. The recovery factor for simple probes in the boundary layer of a wind tunnel could be defined by:

$$r = f(V_t, Pr, Re, k, K_K, K_{PL} \ldots) \tag{53}$$

As the local flow conditions in a wind tunnel and on the skin of an aircraft are quite different (cf. Section 4.3.3.2), the term "flight test recovery factor" was created:

$$r_{\text{flight test}} = 1 - (1-r)\frac{V_t^2 \text{ local}}{V_t^2 \text{ aircraft}} \tag{54}$$

Where this factor was acquired through measurements on the aircraft, it automatically incorporates other components of the position error (such as the probe location error) so that a family of curves dependent on altitude or aircraft weight (or angle of attack) resulted in place of a single curve dependent on velocity.

In the case of good total temperature probes the recovery factor is practically only a function of the Mach number M and the design characteristics K_K of the probe:

$$r = f(M, K_K) \tag{55}$$

It is true that this definition, too, incorporates certain components of the heat conduction error and radiation error (cf. below). However, in the case of TAT probes all terms involved in the air speed error or the recovery temperature can be treated as if all errors were independent of each other. It is seen from Figure 68 that the recovery factor of TAT probes for outside air temperature readings in the subsonic range (up to about Mach 0.7) is located near a constant value between 0.96 and 0.99; certain models with open wire elements have even better characteristics. The dispersion between individual probes of the same type should amount to about $\pm 0.015\ r$ in the air speed range below Mach 0.7 (cf. the dispersions of simple probes stated above). Sometimes, slightly lower values of r will be encountered in the case of TAT probes for engine air inlet temperature measurements. This is primarily due to the requirements for low probe height, high vibration stability, and others**. In the

* In the case of other gases and liquids with higher Prandtl number, values above 1.0 are possible.

** One extreme sample is the Type 153 AC probe with recovery factors between $r = 0.7$ and 0.8.

case of TAT probes for outside air temperature measurements the recovery factor increases sharply above approximately Mach 0.7, since the internal flow velocity in the probe remains constant above this value. Under these circumstances it is better to use a different term:

(c-2) The recovery error is nondimensional, too, and states the ratio of velocity error to total temperature:

$$\eta = (T_T - T_r)/T_T = E_V/T_T \qquad \text{or} \tag{56}$$

$$E_V = \eta \cdot T_T \tag{57}$$

Figures 67 and 68 show that, in the case of TAT probes, the recovery error can be considered almost constant in the supersonic range (this is the manufacturer's information). However, a more detailed investigation will show that, while the total temperature ahead and behind the shock wave remains constant in supersonic flight, all other values (Mach number, velocity, pressure, etc.) are subject to change. As a result of shock wave formation in front of the probe inlet, the air flow enters at increased temperature, but with a subsonic Mach number that is a function of the flight Mach number (cf. Figure 69). The shock wave itself can be assigned a recovery factor of $r = 1.0$. The prevailing subsonic recovery factor r_1 of the probe must be added. The supersonic recovery factor r' that results from these two steps can be described as follows (according to Reference 3):

$$\frac{1 - r'}{1 - r_1} = \frac{[1 - \frac{2\gamma}{\gamma-1}(M^2 - 1)] \cdot [1 + \frac{\gamma-1}{2} M^2]}{\frac{\gamma+1}{2} \cdot M^4 \cdot [\gamma \cdot M^2 - \frac{\gamma-1}{2}]} \tag{58}$$

where the supersonic flight Mach number must be substituted for M (cf. Figure 69). Thus assuming a constant subsonic recovery factor, a sharply defined maximum of the recovery error is obtained (and hence of the velocity error) at Mach 1.0 which decreases at first rapidly and then more slowly with increasing Mach number and will start to rise again slowly beyond Mach 2.5. However, assuming that the recovery factor will rise at Mach numbers between 0.7 and 1.0 (at the probe inlet), this stipulation will have an effect in the supersonic range between Mach 1.0 and 1.4 also (where the Mach number drops behind the shock wave from Mach 1.0 to about Mach 0.7). As a result, the maximum of the velocity error is considerably flattened so that the values of the theoretical consideration differ only little from the values that stipulate a constant supersonic recovery error (relative value in percent of total temperature).

Instrument readings where the velocity error increases rapidly with increasing supersonic Mach number are partly due to the selection of an unsuitable probe type, and partly to the fact that the thickness of the aircraft's boundary-layer* at the (poorly selected) location of the probe increases with increasing air speed, and hence are consequences of the probe location error (cf. above).

(c-3) The recovery ratio R is sometimes used in English-language literature. It is defined by the ratio of recovery temperature to total temperature:

$$R = 1 - \eta = T_r/T_T = (T_T - E_V)/T_T \tag{59}$$

The interrelationship between the terms recovery factor r, recovery error η and recovery ratio R is seen from Figure 67, where the following relationships apply:

$$\eta = 1 - R \tag{60}$$

$$\eta = (1 - r) \cdot \frac{\Delta T_k}{T_T} = (1-r) \frac{T \cdot \frac{\gamma-1}{2} M^2}{T + T \cdot \frac{\gamma-1}{2} M^2} \tag{61}$$

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* Therefore, some probe types are supplied with greater probe height (greater distance between inlet and base plate).

$$r = 1 - \eta \frac{T_r}{\Delta T_k} = 1 - \eta \frac{T + T \cdot \frac{\gamma-1}{2} M^2}{T \cdot \frac{\gamma-1}{2} M^2} \qquad (62)$$

Probes with thermocouples to measure the engine exhaust gas temperature (EGT) are included in this figure.

The velocity error E_V (measured in °C or °K) can be defined in different ways, using the term named above (r, η, R). In most instances only the Mach number (from altitude and indicated air speed) and the recovery temperature T_r (from the indicated temperature through correction for all other error types) are available for determining the velocity error. Where a flight log is available the Mach number can be replaced by the true airspeed V_t, which is then directly available. Hence the following equations are obtained:

$$E_V = f(r, M) \qquad (63)$$

$$= T_T - T_r = \Delta T_k - \Delta T_{kr} = \Delta T_k - r \cdot \Delta T_k \qquad (64)$$

$$= (1 - r)\ \Delta T_k \qquad (65)$$

$$= (1 - r) \frac{V_t^2}{7592} \quad \text{(for } V_t \text{ in knots)} \qquad (66)$$

$$= (1 - r) \cdot T \cdot \frac{\gamma-1}{2} M^2 \quad \text{(for } T \text{ in °K)} \qquad (67)$$

$$= (1 - r) \cdot \frac{T_r}{1 + r \cdot \frac{\gamma-1}{2} M^2} \cdot \frac{\gamma-1}{2} \cdot M^2 \qquad (68)$$

$$= \frac{T_r}{(1 - \eta)} - T_r \qquad (69)$$

In the case of TAT probes equation (69) can be replaced by the approximated formula

$$E_V \simeq \eta \cdot T_r \qquad (70)$$

(cf. Figure 111). The error resulting from this approximation will remain below a level of 0.01°C as long as η is smaller than 0.005, since under these conditions T_T and T_r will have approximately the same magnitude.

The changes in gas characteristics, i.e. changes in γ (cf. Section 5.2.1), which take place under certain conditions, will not affect the magnitude of the quantities r, η, and R, unlike the absolute values of total temperature T_T and velocity error E_V.

On the other hand, the liquid water content (LWC) in the air (in droplet form) will affect the velocity error, depending on design characteristics (cf. Section 5.2.1). Probe types with internal air deflection prior to reaching the element (cf. Figure 18) have a much lesser dependence of the measured temperature on the liquid water content in the air than other probe configurations (cf. Figure 17), since practically only very few and extremely small water particles reach the probe surface and, therefore, can have an effect only on relatively small parts of that surface. Figure 70 shows the velocity error components (in °C) for the Type 101 (without air deflection) and the Type 102 (with air deflection), plotted as a function of the water content in the air (kg water per kg air), and determined by wind tunnel measurements at Mach 0.4 and 0.5 (Reference 39). Measurements taken by other authors during flights with different water content in the air yielded the following relationships between the temperature increase ΔT_{kd} measured in dry air and the temperature increase ΔT_{kw} measured in air with water content, acquired for the Type 102 probe with enclosed element:

$$W = \Delta T_{kw}/\Delta T_{kd} \left.\begin{array}{ll} = 1.0 & \text{in dry air} \\ = 0.98 & \text{in granular snowfall} \\ = 0.93 & \text{in intensive rainfall} \\ = 0.91 & \text{in very intensive rainfall} \end{array}\right\} \begin{array}{l}\text{constant from} \\ \text{Mach 0.5 to} \\ \text{Mach 0.95}\end{array} \qquad (71)$$

It is natural to designate the ratio $\Delta T_{kw}/\Delta T_{kd}$ the liquid water influence factor W by which the prevailing recovery factor r of the probe type in question must be multiplied:

$$E_V = (1 - W \cdot r) \cdot \Delta T_k = (1 - W \cdot r)(T \cdot \frac{\gamma-1}{2} \cdot M^2) \tag{72}$$

On the other hand, heat is released at the probe element during ice formation, and is indicated as a temperature increase.

(d) Radiation Error E_R

In the case of TAT probes for outside air temperature measurements that have at least one radiation shield, this error will be significant only at high total temperatures (above about M = 3) and at very high altitudes (above about 40,000 ft). As a rule, it has a negative sign. Figure 71 shows the radiation error for the Type 101 probe, plotted as the relative value of total temperature vs. the Mach number. The dependence on altitude is indirectly included in this plot, since high Mach numbers are flown only at great altitudes. The radiation errors for the Type 102 series tend to be slightly smaller than the data shown in this figure (exact measurements are still lacking). TAT probes for engine inlet temperature measurements, in most cases, show a slightly greater radiation error that may have the effect of a decreasing recovery factor r with increasing air speed (similar to the curve for the Type 154 F in Figure 68).

In the case of simple probes without a radiation shield, considerably higher values for the radiation error at medium air speeds and high altitudes must be expected. In this type of probe it is also possible to encounter radiation errors of several degrees centigrade with positive sign, when it is exposed to direct sunlight, or on the ground, when the probe is exposed to solar radiation reflected from the runway.

Radiation errors can be described by:

$$E_R = f(T_r^4 - T_w^4, K_R, 1/p_T) \tag{73}$$

Again, T_w denotes the temperature of the probe housing, and the factor K_R represents the radiation characteristics of the element that are due to its design, i.e., emissivity, the probe area, and additional parameters; p_T is the total pressure.

In the case of compressor inlet and turbine exhaust gas temperature measurements, the radiation error deserves special attention when the engine is operated under partial load (with low air flowrate) at high altitudes, and/or when the temperature of the engine casing differs greatly from the gas temperature (for instance, when a cold engine is started). In these cases it is difficult to isolate the radiation error from the heat conduction error (refer to Figure 72).

For temperature measurements on surfaces, the outward side of the probe should have the same emissivity as the remaining surface so that both are subject to the same radiation losses.

The radiation error encountered during measurements in liquids is almost always so small that it can be disregarded.

(e) Conduction Error E_C

This error is due to heat transfer from the element via its mountings and electrical leads. In the case of TAT probes for outside air and compressor inlet temperature measurements, the probe housing and the aircraft fuselage are normally at a slightly lower temperature than the element. However, the opposite can also take place, for instance when the probe deicer system is turned on, or in rapidly climbing flight. In the latter case the temperature of the fuselage drops at a slower rate, because of its large heat capacity, than the air temperature. Consequently, the conduction error can have either a negative or a positive sign. However, it will have a major effect only if there is little convective heat exchange with the air. This means that it will rise with increasing altitude*

* At very high altitudes the boundary layers inside the probe (at the element, at the radiation shield, and at the housing walls) become very thick so that they will affect each other with the result that extrapolations based on data acquired at low altitudes will become highly inaccurate.

and decreasing gas pressure. In the case of modern probe types, it is so small that it can be disregarded, and is difficult to isolate from other errors such as the radiation error, even in extreme cases. At low air speeds it can possibly have the effect of reducing the recovery factor r (which is otherwise constant in this air speed range) (cf. Figure 68, "cylinder probe").

The conduction error can be defined by:

$$E_C = f(1/p_T, T_r, T_w, K_C) \tag{74}$$

where

T_w is the temperature of the probe housing, and

K_C is a factor to allow for design features.

The conduction error of simple (wall-type) resistance probes or thermocouples is a hyperbolic function of immersion length in the gas flow. (Figure 73 shows the resulting ratio of temperature differences $(T_r - T_m)/(T_T - T_W)$, as a function of the probe length/diameter ratio and the flowrate or mass velocity). The conduction error can assume appreciable values if the probe diameter is relatively large, compared to immersion depth, and also at small flow velocities.

In the case of temperature measurements in <u>liquids</u> the heat transfer between the probe and the measured object is considerably better than in gases, and in most instances the temperature difference between the liquid and the container or tube wall into which the probe is screwed is relatively small. Consequently, appreciable errors can develop only in extreme cases (for instance, where a probe of very short immersion length is used)(cf. Section 6.1.3).

In the case of temperature measurements on <u>solid objects</u> (such as cylinder blocks, etc.) or on <u>surfaces</u>, the probe is mounted directly on the object to be measured so that heat conduction could take place only via the electrical leads, if these are routed unfavorably (cf. Section 6.1.3), so that they cannot assume the temperature of the solid body at their ends near the probe.

(f) Self-Heating Error E_{SH}

This error is encountered only in resistance thermometers. It is due to the fact that a current must flow through the element in order to determine its resistance which, in turn, is a function of its temperature. During this process a heat energy P, (cf. Figure 74) is generated that increases with the square of the measuring current, $(I^2 R)$. This thermal energy is transferred to the air flow via the thermal resistance R_{bl} of the probe element's boundary layer. As altitude is increased, and/or airspeed is reduced, the element temperature must increase before equilibrium is reached. As a consequence, errors of several degrees or more can occur, if indicators requiring large currents are used.

The self-heating effect of resistance thermometers is one of the most important, and <u>frequently one of the largest errors</u>, unless modern servo indicators are used. It is a systematic error, always of positive sign, and can be defined by:

$$E_{SH} = f(P, 1/p_T, 1/M, T_T, K_{SH}) \tag{75}$$

meaning that this error is a function of electrical power, P, total pressure p_T, Mach number M, total temperature T_T, and a factor K_{SH} that is due to the probe design. Another form of definition is:

$$E_{SH} = P \cdot R_{bl} = P/(h_c \cdot S) \tag{76}$$

In the case of <u>simple probes</u> used to measure the <u>outside air temperature</u>, wind tunnel calibrations cannot be applied directly to conditions prevailing in the aircraft, since the boundary-layer at the element differs considerably in both cases. Unfortunately, exact data on this problem are not available, probably because it used to be the prevailing opinion that the self-heating error was so small compared to the velocity error that it could be disregarded.

The relationships are much more clearly defined in the case of <u>TAT probes</u> for outside air and compressor inlet temperature measurements. In this case the self-heating error varies approximately inversely to the square of total pressure, meaning that it is dependent on altitude and Mach number (and on total temperature to a degree that can be disregarded in most instances). Therefore, in accordance with the constant air flowrate above a range of about $M = 0.7$, the self-heating error will remain constant, too. As a result of the effect of design features, TAT probes with open wire elements have almost twice the self-heating error of enclosed elements at almost all air speeds.

Figure 74 represents the electrical power P (in mW) dissipated in the probe, as a function of the measuring voltage U applied to the probe's terminals, and the current I flowing through the probe, plotted for the prevailing resistance value R_X. For measurements in air, a value of 5 mW should not be exceeded to avoid excessive self-heating. Figure 75 shows the relative self-heating SH of the probe, in $^\circ$C per mW, for different probe types with open-wire and sealed elements, as a function of altitude and Mach number. Using these data the self-heating error E_{SH} (absolute value in $^\circ$C or $^\circ$K) is obtained from:

$$E_{SH} = SH \cdot P \quad [P \text{ in mW}] \tag{77}$$

Two auxiliary scales for instruments with 4.5 mW and 40 mW probe load are shown on the right margin of these figures so that the self-heating error for the majority of galvanometers and ratio meters can be read directly. The values plotted in Figure 75 for zero air speed ("still air") and for measurements in stagnant water ("still water") are approximate, for information only, since they cannot be measured exactly (cf. Section 4). Figure 76 shows the self-heating of some CIT probes as a function of the air flowrate (mass velocity) of the engine.

Again, it should be pointed out that modern servo instruments subject the probe to a load of only about 0.04 mW, so that no measurable self-heating develops. However, this does not apply to the older types, nor to the majority of air data computers.

The self-heating error reaches much smaller values during temperature measurements in liquids. Therefore, a simple cooling water thermometer, for instance, cannot be used to measure cockpit air temperature because the calibrations are only valid for self-heating in water and for a certain flow velocity.

The self-heating error assumes even less significance in most instances during temperature measurements on surfaces or in solid bodies, provided that the probe has a very good heat contact to these bodies, meaning that it should be completely embedded in the body, if possible. The amount of the remaining self-heating error under these circumstances depends on the current through the probe, on the heat transfer resistance, and on the volume and thermal characteristics of the body.

Thermoelectric elements have practically no self-heating error, because the generated measuring currents are far too small.

(g) Deicing Heat Error E_{DH}

This error can develop only during outside air and compressor inlet temperature measurements, if probes with deicing systems are used. As we indicated earlier, ice formation on the aircraft and hence on the temperature probe is improbable after the aircraft has exceeded an air speed of about Mach 0.8 (cf. Figure 60). However, it is not acceptable to depend on this possibility since, on the one hand, not all aircraft reach this air speed range and, on the other hand, an unacceptably long period of time is required for "natural deicing", if there was an ice built-up at lower speeds*. It must be kept in mind that ice formation on the probe will considerably increase the response time to temperature changes and the conduction error, and that the ice temperature will be measured instead of the air temperature. The ice temperature depends, in part, on the temperature of the aircraft fuselage and the heat balance** during the process of freezing or thawing ice particles. These measuring errors can exceed 10° to 20°C. Where an air data computer is used, this can result in false values being used over extended periods of time.

For these reasons, most of the TAT probe housings are provided with deicing heaters (cf. below). Unfortunately, despite boundary layer control (cf. Section 2.1) this will result in a certain temperature

* In the case of probes with open-wire elements the wire can break as a result of the vibration of ice particles adhering to it.

** Cf. meteorological errors, Section 5.2.1.

increase of the air in the probe housing that results in a temperature measuring error, the deicing heat error E_{DH}. This error increases with decreasing air flowrate through the probe, i.e., with decreasing air speed and/or with increasing altitude. At zero air speed (for instance in hovering helicopters and VTOL aircraft) the temperature increase can reach such large values that open-wire elements will be damaged. The deicing heat error still ranges between 0.2 and 0.5°C at Mach 0.3, depending on altitude. Errors in probes with two measuring elements* can reach approximately twice these values. Unfortunately, very little measured data is available (at Mach 0.3 and 0.8, Reference 32B). Figure 77 shows the values extrapolated by the manufacturer from these measurements for two of his models. The measured points for the remaining models are of about the same order of magnitude.

Therefore, for conventional aircraft, it is recommended that the deicing system be turned on only after takeoff. In the case of helicopters and VTOL aircraft this does not apply to periods of hovering flight. Any ice layer that may have formed during a zero air speed period (for instance during extended hovering flight) will be thawed in not less than 45 seconds, in accordance with test conditions for probes. The deicing heat error is relatively small at higher air speeds (for instance, between 0.12° and 0.3°C at Mach 0.8). Since most of these data apply for dry air, we can assume that these magnitudes are smaller under icing conditions. The relationship between this error and the attitude error under extreme angles of attack or angles of yaw was already discussed in the chapter devoted to that error.

The deicing heat error always has a positive sign and could be defined by:

$$E_{DH} = f(H, M, K_{DH}, P_{DH}) \tag{78}$$

where

P_{DH} is the electrical power devoted to deicing, and

K_{DH} represents the design features.

The deicing heater, like the heater of the pitot tube, is self-regulating.

Probes with bleed air deicing systems can be supplied for measurements of compressor inlet temperature (cf. Figure 22, right). In that case any deicing heat error that may develop will be a function of the flowrate, the bleed air temperature, and the probe design.

Since the deicing heat error will occur only in heated probes for TAT and CIT measurements, it is not necessary to consider it for any other type of probe.

5.2.3 Temperature Lag Error E_L

The temperature lag error encountered in measurements of air and gas temperatures (outside air, compressor inlet, engine exhaust gas) is an aerodynamic error, and occurs only when there are changes in total temperature (when the static air temperature and/or the air speed change). In that case, a period of time is required for the element to reach a new state of temperature equilibrium (cf. Figure 79).

If the response of the probe corresponds to a differential equation of the first order, the relationships represented in Figure 78 (solid curve) will result when a certain temperature value T_A changes abruptly to another value T_E. The measured temperature T_x will change as an exponential function of time t (in seconds), in accordance with the equation:

$$\frac{T_E - T_X}{T_E - T_A} = e^{-t/\tau} \tag{79}$$

In this equation, $T_E - T_x = E_L$, which is the absolute value of the temperature lag error at time t_x. The time constant τ (measured in seconds) is the time required for the measuring instrument to reach 63.2% of the temperature difference in the event of an abrupt temperature change. In that case the indicated error will be (100% - 63.2%) = 36.8% of the temperature change. A value of 99.33% of the step change will be reached only after five times the time constant (cf. Figure 84).

In the case of constant rate of temperature change, the indicated temperature will always lag behind a time period exactly equal to the time constant. The temperature lag error is equivalent to the

* One for the TAT indication, the other for the air data computer.

product of time constant and temperature gradient.

$$E_L = \tau \cdot \Delta T/\Delta t \tag{80}$$

The time constant τ of a temperature probe can also be defined as the product of heat capacity C and thermal resistance R_{bl} in the boundary layer on the probe element:

$$\tau = C \cdot R_{bl} = f(p_T, M, K_L, C) \tag{81}$$

Consequently, it is a function of total pressure and Mach number (or altitude and Mach number), heat capacity C of the probe, and other design features. The latter factor, represented by K_L, also takes into consideration the flow conditions, the heat transfer coefficient, and other effects. The time constant remains independent of the temperature values (T_A, T_E, T_x, etc.) as long as the physical characteristics of the flowing medium are independent of temperature. In the case of probes located in engine inlets the air flowrate is used instead of altitude and Mach number. In that case the time constants τ_I and τ_{II}, respectively, for the flowrates Φ_I and Φ_{II} are inversely proportional to the square root of both flowrates:

$$\tau_I/\tau_{II} = \sqrt{\Phi_{II}/\Phi_I} \tag{82}$$

The temperature indication lag of sealed elements (cf. Figure 80) is up to 100 times greater than that of open-wire types, where the mass is considerably smaller and the surface of the resistance wire is in direct contact with the air (cf. Figure 83). However, open-wire types almost always have multiple time constants, cf. below. Figures 81 and 82 show the time constants of some CIT and EGT probes as functions of the flowrate (mass velocity).

It is true that a certain portion of the probe time constant can be compensated by simple means (Reference 52). However, since the time constant can change by a ratio of three to one over the performance range of an aircraft, this possibility has been exploited only in special cases to date. However, it can be of interest for flight testing purposes, since it is relatively inexpensive.

The time constant of simple probes installed in the aircraft's boundary layer is dependent on the installation since the air that contacts the probe has already experienced a temperature exchange with the aircraft fuselage. Also, the fuselage requires much more time to follow temperature changes than the small probe. Consequently, the time constant of the probe in this case is increased. To a smaller degree, this also applies to a TAT probe if it is installed in an inlet wall for CIT measurements (cf. the corresponding paragraph in the discussion of probe location error, above).

If a probe reacts faster to temperature increase than to the same magnitude of temperature decrease, this is an indication of a large radiation error (Reference 47), typical for older types of temperature probes.

The spatial resolution of temperature measurements, that is the distances covered at different air speeds, vs. three times the value of the time constant - corresponding to a relative temperature lag error of 5% of the temperature change - is illustrated in Figure 85.

Consequently, assuming that the probe response corresponds to an exponential function (or a differential equation of the first order), the absolute value of the temperature lag error E_L (measured in °C or °K) is:

$$E_L = T_E - T_x = (T_E - T_A) \cdot e^{-t/\tau} \tag{83}$$

The left-hand diagram of Figure 84 shows the acquired portion of a temperature change, or the relative temperature lag error (both stated in percent of the temperature change), as a function of the ratio of the selected time t_x to the time constant τ. Shifting to the right-hand diagram the absolute value of the temperature lag error in °C (or °K) is directly obtained as a function of the acquired absolute value of the temperature change.

The response of many resistance probes to abrupt temperature changes no longer corresponds to a single exponential curve; rather, it can be subdivided into several exponential portions (shown as dotted curves in Figure 78). The first, rapidly reacting portion is frequently interpreted as the behavior of the resistance wire, and the additional ("slower") portions as the behavior of the radiation shield, the mountings, and the probe housing. In these cases several time constants must be used. One example of this type is shown in Figure 83 for the Type 102 E 2 AL probe with open-wire element. The reaction behavior of this probe can be described by two time constants, where the following equation applies for the absolute value of the temperature lag error E_L:

$$E_L = T_E - T_X = (T_E - T_A) \cdot (0.8 \cdot e^{-t/\tau_1} + 0.2 \cdot e^{-t/\tau_2}) \tag{84}$$

The time constant cannot be measured in stagnant air (this holds also for self-heating) but only in an exactly defined flow velocity, since otherwise the heat transfer would be dependent on too many parameters in too complex a relationship, and the resulting instrument readings would show too great an error spread. Therefore, the measurement must be performed in an air flow of constant velocity but abruptly variable temperature. At least four values for the temperature change of the probe must be measured (for instance, 20%, 50%, 63.2% and 90%), in order to be able to plot a curve of sufficient accuracy. Only if the times for 50% temperature change are equal to 0.693 τ and for 90% equal to 2.303 τ (Reference 13), will the curve represent a simple exponential function. This is the only condition for which the time constant τ is in itself sufficient to determine the indication time lag.

In a thermometer array a time constant is not only associated with the temperature probe but also with the connected unit (for instance, the indicating instrument). This results in a thermometer time constant τ', defined by the equation:

$$\tau' = \tau + 2F/w_n \tag{85}$$

Here τ is the time constant of the probe (with simple time constant), F is the damping ratio (in most cases between 0.7 and 2), and w_n is the (undamped) natural frequency of the indicating (or recording) system (in radians/sec). Since the natural frequency of these systems is far greater than 10 Hz in most cases, the contribution of the indicating instrument to the time lag of the indication or recording is so small in most instances that it can be disregarded (exception: where the instrument is connected to open-wire probes)(Reference 3).

Since temperature changes are individual, short-time phenomena whose exact profile (abrupt, sinusoidal, etc.) is usually unknown, the observer will usually delay his reading until the indication has stabilized after a certain period of time (corresponding to three to five times the time constant). This means that the temperature lag error (like the lag error in the pitot-static system) is normally disregarded. For the reasons named above, a true correction could be made, at best, at a later time to the recorded values. However, this problem is easy to deal with only where the temperature changes are step changes from one constant temperature level to another constant value, or where they are changes with constant temperature gradients (^{o}C/sec). All other changes require elaborate mathematical treatment (Reference 3), particularly where probes with several time constants were used.

The same considerations apply to the resulting time constant when measuring temperatures in liquids. In its expanded form the equation $\tau = C \cdot R$ reads:

$$\tau = \frac{c_p \cdot v}{S \cdot H} \tag{86}$$

where

c_p is the specific heat, Wsec/cm^3 ^{o}K, of the liquid;
v is the volume of the probe, cm^3;
S is the effective surface of the probe, cm^2, and
h is the heat transfer coefficient, W/cm^2 ^{o}K .

Since the heat transfer coefficient is much greater for liquids than for gases, the time constants will be much smaller so that they can be disregarded, in most instances, compared to the rate of change of the temperature.

Where temperatures are measured on surfaces and in solid bodies the time constant of the measuring element is practically insignificant.

5.2.4 Instrument Error E_I

This category is composed of the errors involved in the indicating instrument, recording instrument, or telemetry system, as well as certain errors involved in the probe and in the electrical leads. The information provided below applies for all types of temperature measurements. The instrument errors are always a function of the measured temperature, while the majority of the errors in air or gas flows that were discussed earlier are primarily a function of air speed, which of course makes corrections much more complicated (cf. Chapter 6).

(a) Probe Calibration Error E_{CAL}

The calibration error of a resistance measuring element depends on the type of material used and on the quality of the instrument. The following tolerance limits are customary:

Platinum elements	standard	= $\pm 0.25°C + 0.5\%$ of the indicated temperature (°C);
	special	= $\pm 0.1°C$, for instance from $-50°C$ to $+150°C$
Nickel elements	standard	= $\pm 1.2°C$, in the range from $-50°C$ to $+150°C$
	special	= $\pm 0.1°C$, in partial ranges.

This calibration error can be encountered as a parallel displacement of the standard calibration curve (resistance/temperature or voltage/temperature), as deviations in the slope of the curve (but retaining the primary calibration point at 0°C), as an irregular, positive and/or negative deviation from certain parts of the standard calibration, or as a combination of these. The elements listed as "special quality" above are types with closer tolerances (such as the so-called PCI elements*) and built-in auxiliary resistances. These elements have two calibration points (cf Figure 86) and are directly interchangeable without recalibration of the thermometer. However, since their calibration curve has a little less slope than the curve of standard elements, they must be used in conjunction with specially calibrated indicating instruments (for instance bridge circuits, etc.).

The tolerance limits of the most important resistance temperature probes are shown in Figures 87a and 87b. The actual calibration errors of these elements are located within these boundaries.

In the case of thermistors and all other types of semiconductors the dispersion of the calibration values of individual instruments is usually very large. However, some companies market types with narrower tolerances that are directly interchangeable. This is achieved through series and parallel resistances built into the probe base, so that standard calibration curves apply for these types. The calibration accuracy in the desired temperature range is typically about $\pm 0.1°C$.

The calibration errors of thermocouples are within the tolerance limits of 0.5% to 0.75% of the indicated temperature (cf. Figure 88), meaning that the voltage deviations caused by the calibration error correspond to a maximum temperature error of $\pm 5°C$ to $\pm 7.5°C$ at a probe temperature of 1,000°C.

A calibration change as a function of service life is mentioned in many performance specifications of resistance probes. In most cases it is so small that it can be neglected. However, calibration changes can take place suddenly, for instance when the recommended maximum temperature values are exceeded, when "contamination" occurs as a result of chemical effects, or when the element was mechanically damaged. Since effects of this type frequently remain unnoticed, it is advisable to verify the calibration curve from time to time, especially where the probes do not remain in place permanently - for instance, in flight testing programs. Special care is indicated in the case of thermocouples made in-house, since these are subject to aging effects. Mass-produced thermocouple probes are supplied in aged condition and retain their calibrated values over relatively long periods of time. This applies especially to sealed probes with welded contact points. The service life of thermocouples depends on factors such as the type of material used, the thermal load exposure, the design, any chemical effects that may be encountered (oxidation, reduction), and mechanical stress (especially where the contact leads are twisted). Also, the

* PCI = precision calibration interchangeability.

so-called "restructuring effect" should be mentioned. It occurs when mechanical stress on thermoelectric conductors results in an "amplification" of the inhomogeneities of the crystalline structure of the conductor material and hence in a change of calibration, for instance when the immersion depth of a thermocouple probe is changed. Therefore, the calibration of thermocouples should be subject to continuous verification at not too infrequent intervals.

(b) Electric Lead Error E_W

In the case of resistance thermometers the leads between probe and indicating instrument have a certain resistance (called "line resistance") which, for instance, amounts to about 0.5 ohm in the case of a 16 m length of AN 20 = 0.6 mm^2 cable (Figure 89, lower diagram). As a result of the allowed production tolerances (±5%), this resistance can vary by about ±0.025 ohm between two lines using the cable mentioned above. Additional differences result from the intermediate plugs and terminals, and from the different lengths and different temperatures of the cables. Consequently, in most instances the individual probe leads have different resistance values which, moreover, can be subject to change in the same direction or in the opposite direction.

Where a probe is connected as shown in Figure 89, circuit type (a), which is a two-wire circuit, the resulting line resistance R_W will equal the sum of R_{WA} and R_{WB}. The resistance value of the probe R_X increases, resulting in an erroneous temperature indication. The magnitude of the resulting lead error E_W (in °C) depends on the magnitude of the resistance R_X and on the slope of the R/T characteristic. It is seen from the upper diagram in Figure 89 that a value of (0.5 + 0.5 + 0.025) = 1.025 ohm for R_W in the case of a 50-ohm Pt probe would result in a lead error of about +5.3°C, but only in an error of +0.53°C in the case of a 500-ohm Pt probe*.

On the other hand, where the three-wire circuit of Figure 89 (b) is used, the resulting resistance R_W will be approximately equal to the difference between R_{W1} and R_{W2}. In the example mentioned above, this would correspond to a value of about 0.025 ohm, resulting in an error of only about 0.13°C for a 50-ohm probe (Pt), and in an error so small that it can be disregarded for a 500-ohm probe*. Therefore, low resistance probes should always be operated in a three-wire circuit**.

As a rule, the lead error has a positive sign. However, indicating instruments exist whose scale is already calibrated for a certain amount of lead error. In that case the actual lead error must be adjusted to this rated value through calibration resistances. A better system is represented by instruments with built-in calibration resistances for lead calibration at two points of the scale. The lead error can be defined by:

$$E_W = f(R_W, R_X, K_W) , \tag{87}$$

where the factor K_W represents the type of circuitry (two-wire or three-wire) and the slope of the probe characteristic.

As a rule, supplying resistance probes with alternating current (400 Hz) instead of direct current will not in itself be a source of error. Unshielded leads can cause indication errors (periodic fluctuations and the like) in the indicating instruments.

Parasitic thermo-voltages in resistance thermometers can only occur as a result of gross infractions of the customary cable-laying rules (to test: interchange the leads to the probe and compare the indications).

Parasitic noise voltages can develop at locations with high vibration levels (for instance, at the engine inlet), if unsuitable probe types (or plugs with oxidized contacts) are used. These are often caused by the so-called strain effect, i.e., by a resistance change in the wire as a result of mechanical strain (as in strain gauges).

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* These values apply only for bridge circuits where R_X and R_4 (or R_x and R_1 in the case of the circuits in Figure 25a) have approximately equal magnitudes.

** Four-wire circuits are used for special requirements in flight testing (References 18 and 21) that are even better; however, they require special bridge circuits, cf. Figure 31c.

The error caused by so-called internal line resistance (i.e., by the resistance of the lead elements between measuring element and the plug connection of the probe) is so small in the types customarily used in aviation that it can be disregarded, since the elements are wired in three-wire or even four-wire circuitry inside the probe housing.

Several types of line error are encountered in the connecting leads between thermocouples and indicating instruments. One of these is due to the ohmic resistance of the extension leads which is much greater than that of copper leads of the same diameter. Therefore, where simple indicating instruments are used (millivoltmeters), a calibration resistance must be provided which is used to adjust the overall resistance (of thermocouple plus leads and calibrating resistance) to a certain rated value RL (for instance 8.0 ohms). Resistance changes ΔRL in the conductor loop will enter into the indication, depending on the internal resistance Ri of the instrument. In that case, the ohmic component of the lead error (in ^{o}C) will be

$$E_W[^oC] = \frac{\Delta RL}{RL + Ri} \cdot Ti \tag{88}$$

where Ti is the indicated temperature* of the instrument (^{o}C). Self-balancing compensation bridge circuits do not have any current flow so that this error component is eliminated.

Additional components of the electric lead error,that occur in the leads of thermocouples as a result of the Seebeck and Becquerel effects (cf. Chapter 3), are the parasitic thermovoltages which cannot be compensated. Parasitic thermocouples result, for instance, in the extension leads where they are passed through other materials, (normal connecting plugs or terminal boards), especially where a large temperature drop exists at the lead-in point. The opposing thermovoltages that are generated at these points will cancel each other but only if these two connecting points have the same temperature. Therefore, special-purpose plugs with pins and sockets of the same material as the extension leads are supplied for the points where these lines must be passed through engine casings and bulkhead frames. However, temperature gradients along the balancing circuits are unavoidable. Consequently, the calibration of a thermoelectric thermometer will be accurate only if the temperature distribution along the extension leads is reproduced in the exact pattern that existed at the time of calibration. But this cannot be achieved in flying operations. This error can reach values of several degrees centigrade, but it is impossible to compute it.

The strain effect is primarily responsible for the development of parasitic noise voltages** in the presence of vibration. These voltages can amount ot more than 100 microvolts. In the case of thermocouples with non-welded contacts (such as twisted wires) the strain effect is further "amplified" by changes in contact pressure during vibration and temperature increase so that elements with welded contacts will always yield more stable data.

Electric lead errors in all types of electrical thermometers can be further generated by externally induced voltages in the leads (primarily 400 Hz cf. Chapter 3). In this case the probe leads would have to be twisted wires placed at a distance from all other lines, or shielded cable which will cause considerable weight (and cost) increases.

Consequently, the electric lead error is greatly dependent on the selected thermometer type, the selected circuit type, and additional parameters. The magnitude and sign of this systematic error can differ greatly even between systems of a similar appearance.

(c) Instrument Error of the Indicator

The most important systematic error is the calibration error or scale error which is caused primarily by inaccuracies (during the calibrating operation), in adjusting and reading the instruments, and also through geometrical inaccuracies of the scale. It is determined under laboratory conditions (so that small lead errors can be disregarded) and is stated as the deviation of the temperature indication from the resistance/temperature curve or voltage/temperature curve of the associated probe

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* In the case of thermoelements whose reversal point Tu of the direction of the output current is not located at 0^oC (but at $+20^oC$, for example), the expression (Ti - Tu) must be substituted for Ti in the equation.

** The possibility of noise voltage demodulation at non-linear circuit elements cannot be excluded completely so that not even filters will be able to separate it from the useful voltage. This applies even for RF injections;for instance, one indicator always indicated temperature changes during VHF voice radio traffic!

element (cf. Figure 90, a and b). For this purpose the element and its leads are replaced by a variable standard resistance or standard voltage source. This error can be incluenced by additional errors such as the aging error that causes a gradual change in the calibration curve as a result of a change in the characteristics of electrical components or as a result of a compensation of mechanical strains in the metal of the instrument's mechanical elements.

The group of statistical errors includes both elastic errors and electrical errors, such as drift and hysteresis errors. In addition, mechanical errors such as transmission and friction errors are frequently found. A more detailed discussion of all the errors encountered in indicators is not necessary here, since these problems have been covered exhaustively in the appropriate literature.

Frequently, the term instrument calibration error or instrument scale error is understood as including the sum of all the errors of the instrument. It is then stated as a systematic error with a statistical component, for instance, in the form of a calibration curve with additional parallel boundaries indicating the measuring uncertainty (cf. Figure 90a). The overall accuracy of the instrument states the tolerance limits where the indication must be located under the stated environmental conditions, and includes all the errors of the instrument. It is given either in percent of full scale or in absolute values at certain points of the scale. Typical values for outside air temperature indicators with resistance probes are shown in Figure 90, a and b, and for engine exhaust gas temperature indicators with thermocouples in Figure 91.

Two additional types of errors must be mentioned here: the so-called environmental errors that can develop as a result of external effects on the instrument, such as temperature, attitude, acceleration, vibration, or magnetic field effects, but they are either not present in the normal flying state or they will occur only with improper instrument installation. On the other hand, the read-out error is dependent on the type of indication, that is, on the number of scale marks or needles in the case of dial-type scales, on the subdivision of the last numeral drum in the case of digital drum indicators, on the size of the numerals, on parallax, and the like. Since these are avoidable errors they require no detailed discussion here.

6. PRACTICAL TEMPERATURE MEASUREMENTS

This chapter summarizes and supplements the relationships discussed in the preceding sections that are observed during measurements in air and other gases, in liquids, on solid bodies, and on surfaces, and explains the different aspects that must be taken into consideration for the selection of the suitable thermometer type, probe design, and indicator. The following paragraphs discuss the calibration of the measuring array and the systematic order and practical examples for error correction.

6.1 Different Types of Temperature Measurements

6.1.1 Measurements in Outside Air

Basically, two methods are discussed here:

(a) Temperature measurements using simple probes installed in the aircraft's boundary layer;

(b) Temperature measurements outside the aircraft's boundary layer.

Simple probes used to measure temperature in the boundary layer of the aircraft are relatively inexpensive. Since the dependence of local flow and temperature relationships at the probe on the characteristics of the unperturbed flow around the aircraft is highly complex (cf. Chapter 4, above), large measuring uncertainties arise that increase rapidly with increasing airspeed. Since in this case the wind tunnel measurements cannot be transferred to the probe installation in the aircraft, time consuming and costly measurements must be taken in every individual instance. Moreover, cost effectiveness demands that a cheap probe be coupled only with inexpensive indicator instruments (or appropriate adapter circuitry for recording or telemetry purposes) that have a low accuracy level (such as 1 to 2% of the instrument range). Finally, these instruments require a large measuring current which causes a self-heating error, the magnitude of which increases with decreasing airspeed.

Consequently, probes installed in the boundary layer have a number of disadvantages:

(a) Measuring uncertainty increases sharply with Mach number; the upper limit of feasibility is located between Mach 0.3 and 0.8, depending on design. Where simple measuring instruments (or measuring circuits) are used, the lower limit is located between about Mach 0.15 and 0.2, because of the self-heating error.

(b) The selection of a probe location is about as difficult as it is for static pressure sensing, and must be governed by the same principles.

(c) An appreciable time lag in the event of a temperature change is caused by the temperature exchange between the air and the aircraft surface forward of the probe location.

(d) The majority of simple probes do not have a radiation shield so that direct (or reflected) solar radiation can result in considerable measuring errors (+4°C, and more). However, where radiation shields are used, they will sometimes affect the flow to such a degree that even small directional changes in the air flow can cause abrupt changes in flow velocity at the probe and, consequently, a large dispersion in the measured data.

(e) This type of probe cannot be protected against icing.

(f) The errors of simple temperature probes must be determined through measurements on every individual aircraft type, since wind tunnel measurements cannot be transferred.

Where this measurement procedure is used, probe configurations should be used exclusively where the inflow is parallel to the aircraft surface (such as the L-shaped rod probes illustrated in Figure 12c or flush bulbs as illustrated in Figure 14), since these types have smaller airspeed errors and the errors are more evenly distributed over the airspeed range than probes with lateral inflow. The probes should be mated only to galvanometer indicating instruments in order to keep the self-heating effect small, however, these instruments should still be calibrated individually. The probe location should be at approximately the same point on the airframe forward of the cockpit where the static pressure sensor for the pilot static system is located* (or might be located). If feasible, the probe should be slightly displaced toward the fuselage belly so that in the normal flight attitude the direct solar radiation exposure of the probe is reduced as much as possible, but not too far in order to avoid an excessive effect of the angle of attack and of aircraft configuration changes (due to landing gear, flaps, etc.).

The most accurate technique currently used for the measurement of air temperatures in flight utilizes total temperature probes (TAT probes, stagnation probes) located outside the boundary layer of the aircraft. Modern TAT probes with enclosed platinum elements are applicable over a wide range of velocity which extends from zero airspeed to beyond Mach 3.5**, and up to altitudes of 100,000 ft. For flight testing purposes the relatively low price increase for elements with more stringent calibration tolerances is always worth-while, since these elements simplify error correction considerably (cf. Section 6.2). For the same reason, only good servo instruments should be used as indicators which, moreover, afford a capability for the installation of independent recording outputs (cf. Section 2.2.6). The special calibration curve of elements with close tolerances must be taken into consideration for the selection of the indicator instrument.

When the probe is selected, types with internal air deflection in front of the element should be preferred, since they show only a fraction of the error caused by the liquid water content of the air during rain or snowfall, or flight through clouds, common to all other types of probes. In the case of straight TAT probes with lateral exits where the air is not deflected until it has passed the element, a considerable increase of the heat conduction error, resulting from water accumulation about the leads, can be expected.

In the case of measurements where a short indication time lag is especially important (for instance, measurements during accelerated flights), TAT probes with open-wire elements can be used whose time constant is up to 20 times smaller than the time constant of hermetically sealed elements. These elements, however, are too fragile for general flying operations, and a probe deicing system turned on at zero airspeed may overheat and damage the element. It is true that any ice which may have formed during a period on the ground will melt in flight. However, this may result in ice bead formation on

* At these points where the measured static pressure is approximately equivalent to the non-perturbed static pressure, the local flow velocity will also be approximately equivalent to the true airspeed or, in the same sense, the local Mach number will be approximately equivalent to the true Mach number.

** At even greater airspeeds, where the total temperature exceeds values of 500°C, TAT probes with thermocouples should be considered (cf. Figure 40).

the wires that cause vibration in the air flow, which could lead to wire breaks. In fact, such failures have been experienced using elements with a nominal resistance of 100 ohms (meaning very fine wires) in supersonic flight, and were probably due to shock wave effects and/or resonance effects.

The combination of a TAT probe with a good servo instrument, although relatively expensive, has practically no self-heating error even at the extremely low airspeeds typically encountered during helicopter or VTOL aircraft flights. In addition, the time constant of these probes is practically insensitive (slight increases) to the extremely high angles of attack sometimes experienced in these aircraft during vertical take-off or turning maneuvers. It should be noted that these characteristics are only applicable when the probe deicing system is turned off. In helicopters* and VTOL aircraft this system should be turned on only in cruising conditions. Otherwise the deicing heat error would rise greatly at low forward speeds, especially when combined with climbing or descending flight (where the probe is exposed to a large angle of attack). Similar considerations apply also for conventional aircraft, if precise temperature measurements are required during low airspeed conditions or during the takeoff and landing phases.

In the <u>supersonic range</u>, TAT probes, unlike the pitot tubes, are largely insensitive to normal shock waves since there is no change in total temperature across the shock. Another advantage is the characteristic, inherent in the design of various types, that the recovery factor which remains constant in the subsonic range (up to about Mach 0.7) rises gradually in the range from Mach 0.7 to 1.0, and then rapidly approaches a value of 1.0 in the supersonic range. The recovery error (stated in % of total temperature), and hence the measuring accuracy of a good TAT probe, remains approximately constant above about Mach 1.0**. This assumes that gross errors were not made in the selection of the probe location. Wind tunnel calibrations of TAT probes are also directly applicable assuming, again, correct probe installation on the aircraft.

The selection of a <u>probe location for TAT probes</u> is not very critical. Any location is appropriate that would be suitable for a pitot tube (total pressure). The probe inlet aperture should point exactly in the flow direction when the aircraft is in its normal flight attitude, and should be located outside the aircraft's boundary layer. This means that an especially favorable area is the aircraft nose, in the zone where the flow is still attached. Vorticity zones should be avoided (behind propeller, antenna boom and the like).

Table 8 shows a compilation of the error types that can occur during OAT and CIT measurements, and the conditions under which these errors are avoidable or unavoidable.

TABLE 8 ERRORS IN OAT AND CIT MEASUREMENTS

Error due to	Countermeasures	Error not avoidable for	Error excessive for
Probe location, Attitude	Selection of location	Hovering or vertical flight	Hovering or vertical flight
Velocity	TAT probe	(Old probe types)	Trans- and supersonic speeds
Conduction, Radiation	TAT probe	Mach > 2.5 at altitudes > 60 000 feet	Low speeds at high altitudes
Self heating	Servoed indicator	Ratiometer-indicator	Low-cost instruments at very low speeds
(a)Deicing heat	(b)Switching out of heater at low speeds	Low speeds and/or high altitudes	Low speeds at high altitudes

(a) If deicing heater provided

(b) For helicopters, use of reverse-flow probes

* For helicopters the use of new reverse flow probe should be considered, which is fairly insensitive to icing (cf. Section 2.1.5).

** Cf. Section 5.2.

TABLE 8 ERRORS IN OAT AND CIT MEASUREMENT (CONTINUED)

Error due to	Countermeasures	Error not avoidable for	Error excessive for
Lag	Selection of element (Open-wire element)	Rapid variations of speed and/or altitude	Fully iced probe (Without deicing)
Lead	Three-wire connection	Lower resistance elements	Low resistance elements with two-wire connection and long leads
Vibration	Selection of probe type and location	(Old probe types)	Some engine stations
Probe calibration	Selection of type	(Low-cost types)	(Low-cost types)
Scale	Servoed instrument	(Low-cost types)	(Low-cost types)

6.1.2 Temperature Measurements in Air and Gases of Engines

In the case of reciprocating engines the only temperature measurements which are usually of concern are the carburetor temperature and possibly the air inlet temperature. Well-type probes (cf. Figure 12a, b) are preferred for these measurements. As a rule, simple probes are sufficient in these applications, since a sharp local increase of air velocity and hence an expansion and decrease of static air temperature is frequently encountered at the measuring point. Condensation and icing effects must be expected on the probe at low air temperatures combined with relatively high humidity, of the type described in Section 5.2.1. This means that the determination of exact temperatures can become very difficult.

In the case of jet engines, the most important temperatures are the compressor inlet temperature (CIT) and exhaust gas temperature (EGT), in addition to other measurements that may be required in flight testing. These temperatures must be acquired as total temperatures, since only these will yield a true measure of the total energy level of the gas. TAT probes should be used in this application in order to obtain reasonable measurement accuracy. Data obtained with simple probes require corrections based on the simultaneous measurement of the local Mach numbers and/or the air or gas flowrate (mass flowrate) at the point at which the temperature is measured. In most instances, these data are not available.

6.1.2.1 Important Engine Temperatures

The compressor inlet temperature (CIT) which depends on static air temperature, inlet geometry, and engine air flowrate, is an important operating parameter and is therefore, determined separately (possibly at multiple points) for high-performance aircraft. In the case of supersonic aircraft, engine life varies directly with the CIT (due to heat stagnation in the compressor), and this in turn determines the maximum permissible airspeed for the given situation. Moreover, in many engine types the CIT is needed as an input to the fuel control system, the compressor vane control system or the air inlet control system. At the present time the maximum acceptable CIT value is about 100°C , in special cases (SST) at about 160°C.

Additional measuring points on the compressor may arise in flight testing, for instance at approximately the middle of the compressor (intermediate compressor temperature) and at its outlet (compressor discharge temperature), where the temperatures are considerably higher but still within the normal application range of platinum probes fitted in special housings.

The exhaust gas temperature (EGT) is measured on a continuous basis in order to achieve the desired thrust within the safety limits of engine temperature. In steady-state flight the EGT immediately behind the turbine is usually about 800°C (in special cases more than 1200°C), and can be exceeded only for short periods of time. In some jet engines the EGT is used as an input for the automatic control of jet nozzle cross-section area. Where an afterburner is provided behind the engine, the temperatures behind the afterburner (when it is in operation) will reach extremely high values (up to 1800°C). The EGT magnitude and local distribution, however, depend on a great number of parameters (such as gas and air flowrate of the engine, distance of measuring points from the afterburner plane, etc.) and can be evaluated only after taking all these effects into consideration. Therefore, temperature measuring points behind afterburners are not customary in normal flying operations but may become necessary in engine and flight testing.

6.1.2.2 Selection of Temperature Probes

The expected temperature ranges, the high flow velocity, and the available installation possibilities must be taken into consideration when selecting the temperature probes for CIT measurements. It was formerly the practice to employ thermocouples exclusively since, on the one hand, less significance was attributed to exact inlet temperature measurements and, on the other hand, thermocouples are less sensitive to vibration because of their small size and moreover, are relatively inexpensive. More recently, however, special-purpose resistance probes have been developed that are insensitive to vibration and yield a voltage change of about 4 mV/^{o}C at a resistance change of about 0.4 ohms per ^{o}C temperature change and a current of about 10 mA, compared to about 0.011 to 0.045 mV/^{o}C for thermocouples. Of course, this almost 100 fold increase in output voltage provides better resolution and measuring accuracy. However, a much more important consideration is the noise voltage produced in thermocouples due to intense engine vibrations which can be in the order of 100 μV or more. This phenomenon will impede the execution of measurements where high accuracy is of paramount importance. The noise voltage is dependent on engine rpm (but is not proportional to it) and has a frequency spectrum ranging from a few Hertz to several kHz (Reference 54). Therefore, a special-purpose TAT probe with resistance element should be used in every instance where the relatively low compressor inlet temperature, CIT, must be measured. Simple and inexpensive resistance probes designed for minimum sensitivity to vibration are available for this purpose. As examples, one design utilizes an open-wire nickel element (temperatures range from -70^{o}C to +260^{o}C with a claimed accuracy of $\pm$0.1 ohm, corresponding to $\pm$0.1^{o}C at temperatures near 100^{o}C, cf. Figure 20b), and another designed with a sealed platinum element (temperature range from -260 to +600^{o}C, cf. Figure 20a). Special configurations of these probes are especially suited for measurements in the compressor (intermediate compressor temperature) and behind the compressor (compressor discharge temperature). Special-purpose types are available for combined pitot-static pressure and inlet temperature measurements where the probe must be able to sustain mechanical damage without shedding parts upon deformation*; some of these can be protected against icing with the aid of bleed air. An important consideration for the selection of a probe is the problem of the time constant, since electronic fuel control systems require a much shorter probe time constant than hydromechanical fuel control systems where excessively short time constants would result in unpleasant control oscillations, especially in the vertical lift engines of VTOL aircraft.

On the other hand, thermocouples are the only recommended probe type for <u>exhaust gas temperature measurements</u>, since almost all presently available resistance probes would suffer damage at this location during the short-time temperature peaks that develop frequently at this point. TAT probes with exposed elements are shown in Figure 39. The exposed and shielded configurations of simple probes shown in Figure 38 are not as desirable since their recovery factor is flow-dependent. The computation of corrections for different states of operation becomes very complicated. The immersion depth L of thermocouples into the gas flow should be about 12 to 15 times the diameter of the cylindrical part of the probe, in order to keep the radiation error at a low level. Also refer to Section 5.2.2 and Figure 73.

In the case of exhaust gas temperature measurements behind afterburners, we must take into consideration that the maximum temperatures experienced by the probe greatly exceed the temperature range for which the calibration of most thermocouples is stable. Therefore, their life will be short, even in the case of special-purpose types, and the probe must be recalibrated at the maximum permissible temperature (to avoid destruction of the element) after every flight or series of measurements. It is unknown whether probes of tungsten rhenium are available for applications in aircraft. This material, which is used in space technology, permits the measurement of temperatures up to 3100^{o}C with a measuring uncertainty of about $\pm$7^{o}C.

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* Danger of engine damage.

6.1.2.3 Selection of Probe Locations

The cross-sections just ahead of the compressor and just behind the turbine have proved to be good probe locations for CIT and EGT. It is much more difficult to determine the number and arrangement of probes at the selected station. Radial velocity and temperature gradients occur in air or other gases moved at high velocity through a duct, depending on the types of flow that develop, but these do not differ excessively since air and other gases have a Prandtl number close to 0.7. It is quite difficult to determine the temperature gradients in the flow through computation, since inlet and exhaust ducts deviate greatly from the "ideal" cylinder with respect to their length and geometry, and the instantaneous or effective values of too many parameters would have to be included. Moreover, the engine stations where temperatures are usually measured are located at the transition points between the circular cross-sections of inlet and exhaust ducts and the annular flow cross-sections of compressor and turbine. The conditions prevailing in the air inlet of a supersonic jet were already described in Section 5.2.2 and illustrated in Figure 65. Therefore, the frequently used procedure is to measure the radial velocity distribution first, (possibly as a function of angle) using pitot tubes (or a "comb array"), in order to obtain an idea of the temperature distribution to be expected.

When it is found that there will be no asymmetrical pressure or temperature distributions over the entire operating range of the engine, it is theoretically sufficient to use a single probe per cross-section where the temperature is measured. If the pressure distribution is asymmetrical,at least four to eight probes must be provided in different sectors. The conventional, fixed array is one to four resistance probes in front of the compressor, and four to twelve thermocouples behind the turbine. The latter are frequently designed as dual probes (with two elements). One set of these elements is switched parallel for an average temperature measurement, while the other elements are each individually connected to a test plug in order to measure, for instance, the temperature behind each individual combustion chamber. For flight testing purposes, considerably more probes are usually provided in front of the compressor, in order to monitor the temperature distribution over the cross-section.

The exact location of the probes in the cross-section to be measured must be selected so that the temperature-sensitive elements are placed neither too near the tube or engine side, nor exactly in the center of the flow, since it is the mean value of temperature that is desired. A "rule of thumb" is to place the probe in the zone between 25% and 75% of the radius in the case of a circular cross-section, or into the zones between about 12% and 36% and/or 64% and 88% of the ring width in the case of annular cross-sections. Several concentric rings of probes with separate outputs are always required where it is intended to acquire not only the temperature relationships during flight but also the conditions prevailing during start-up of a cold engine. This is the only possible method to acquire the distribution of the temperatures that are entirely different in this state of operation (or to acquire the temperature differentials between tube wall and the gas), in order to form conclusions as to the most advantageous probe location for the acquisition of the mean temperature. Also, conditions in the air inlet of vertical lift engines in VTOL aircraft are especially critical. An inhomogeneous air mixture of continuously changing temperatures and different temperature distribution over the cross-section area is encountered at these inlets due to exhaust gas reflection off the ground and recirculation, together with the colder external air.

A typical probe array in the inlet of vertical lift engines, using the resistance probes shown in Figure 21, was presented in Figure 66, where the array parallel to the flow (a) is the better solution. In order to determine the mean temperature directly, a possible array places three probes in series and three such groups in parallel so that the resultant total resistance will be equal to the resistance of a single probe. Another possibility is to place four probes (each with a nominal resistance of 50 ohms) in series, and to place two of these groups in parallel so that a total resistance of 100 ohms will result. In that case a mass-produced instrument (for 100-ohm probes) can be used.

6.1.3 Temperature Measurements in Liquids

As a rule, measuring the temperature of liquids in containers does not involve any difficulties since the flow velocities are low and the temperature changes very slowly. The probe must not be installed on the topside but on a side wall or on the bottom of the container. If it is installed on a side wall the location must be low enough so that it will always be wetted by liquid as the level in the container drops. If it is installed on the container bottom, and if there is a possibility of a sediment forming, the probe must be long enough to pass through the sediment (cf. Figure 92) without penetrating the surface of the liquid. In all instances it must be kept in mind that the immersion depth of the probe (which generally will be of the well-type) should not be less than six to eight times the diameter, in order to keep the heat conduction error small. Also, "dead spots", i.e., areas where there is no flow, should be avoided since liquid layers of different temperatures may form these locations. For the same reason, the indication can vary in aircraft on the ground, for instance when wind gusts set the liquid into slow motion.

Where temperatures of liquids are measured in tubing it must be kept in mind that the Prandtl number of liquids differs greatly from one, so that the velocity and temperature distribution over the cross-section area differ greatly. If the Prandtl number is very high (for instance in lubricating oil) the liquid temperature will be uniform over most of the tube cross-section and will show a high temperature gradient only immediately adjacent to the tube wall,if the wall and liquid temperatures differ greatly as a result of heat conduction or radiation. A good compromise for the liquids used in aircraft (as it is for air) is to place the temperature probe outside the tube axis (in the zone from 0.25 to 0.75 r).

Figure 93a shows how relatively long probes can be accommodated in a bend of a liquid line. Care must be taken that the flow encounters the temperature-sensitive part of the probe first and the base last, since the latter will have essentially assumed the temperature of the tube wall. Also, the line can be insulated externally to reduce the conduction error (unless the line is extremely short), so that the tube will approximately assume the temperature of the liquid. Possibly the use of appropriately designed struts may be worthwhile (Figure 93c) which increase the distance between threaded socket and tube,and also permit the employment of relatively large probes in very narrow tube cross-sections. For special applications, i.e., for lines with relatively high internal flow velocity, cylindrical probes for installation in tubing are available (Reference 25). Another frequently used procedure is to measure the temperature indirectly through the tube wall (cf. the next Section).

The usual temperature sensors are well-type resistance bulbs, usually with a nickel element, of the type illustrated in Figures 12a and b. Where these are combined with inexpensive indicating instruments that require a very large measuring current, a self-heating error must be expected. This error is greatly dependent on flow velocity so that the indication may require appropriate correction.

6.1.4 Temperature Measurements on Surfaces and Solids

The measurement of surface temperatures on aircraft is often very difficult. Some of the factors to be considered in the design of an installation include: (a) the installation of a temperature sensor with leads always causes a disturbance of the surface temperature;(b) aerodynamic conditions prohibit the use of some probe locations which otherwise may appear to be desirable; and (c) measuring the internal temperature of solid bodies requires special bores that may adversely affect the strength of the body in question.

Particular care must be exercised in the design of the probe installation in order to obtain the best heat transfer possible between surface and temperature probe. This is done using large-area screwed, soldered, or welded contacts as shown in Figure 94, or by the extremely sparing use of an adhesive of good heat conduction characteristics. In short, there must not be any air, paint or oxide layer, etc., that would provide thermal insulation, nor should it be possible for such a layer to form after probe installation. Especially good conditions exist where a recess can be drilled or machined into the surface in order to embed the temperature probe flush with the remaining surface (Figure 94g) This will avoid any major perturbance of the boundary layer even on surfaces with a high-velocity inflow (such as aircraft skin). The emissivity of the probe surface should be matched with that of the surface to be measured (for instance through blackening, surface treatment, and the like) especially where measurements are performed in rarefied gases (high altitudes). This avoids any perturbance of the temperature equilibrium as a result of differences in radiation. Of course, the thermal capacity (and hence the volume) of the temperature probe should be kept as small as possible compared to that of the surface to be measured so that the probe will follow any temperature change rapidly. Yet the presence of the probe can disturb the temperature field considerably due to heat conduction along the electrical leads as seen from the profile of the isotherms in Figure 95a. The type of lead configuration illustrated should also be avoided because of the danger of a lead breaking at the element. Rather, the leads should be placed as shown in Figure 95b, if possible along an isotherm and with good heat conduction to the surface, and the minimum length L should be about 50 times the lead diameter d. For measurements on the outside of the aircraft skin the leads are best placed parallel to the air flow until they reach the nearest frame, stringer, or web, to reduce the perturbation of the flow; from there they are passed inside the aircraft, unless it is possible to measure the temperature indirectly as described below. Unfortunately, it is rarely possible to place the leads parallel to isotherms on the outside of the skin, as it is seen from the examples of surface temperatures shown in Figures 97 and 98.

In the case of direct surface temperature measurements the actual temperature on the outside (for instance, of an item of equipment or a pipe) is measured by placing a temperature probe on the outside. However, if the probe and its immediate vicinity are covered by an insulating layer, a temperature will be measured that is very close to that of the opposite side of a metal skin. This will permit indirect surface temperature measurements, i.e., a measurement of the surface temperature of the aircraft skin in an air flow by means of a probe installed on the inside of the skin (cf. Figure 95c). The measured value would be too small unless sufficient insulation were placed over the probe location. However, if this insulating layer were to extend over too large an area the reduced loss of heat would excessively change the conditions compared to the normal state, and the measured temperature would be too high. Consequently, most of the heat conduction error caused by the leads of the temperature probe can be compensated via the area and thickness of this insulating layer*.

A very popular procedure is the indirect temperature measurement of a liquid flowing through a pipe, measuring from the outside of the pipe. Figures 96a through 96c show the temperature relationships in the event of unfavorable heat conduction by the leads, the conditions where the leads close to the probe are placed properly parallel to the pipe, and the relationships prevailing when an additional insulation is wrapped around pipe and probe. It is seen that in the latter case the temperatures of liquid and outside tube wall (and the probe) are almost equal. However, this applies only where the length of the pipe is at least 20 times the pipe diameter, and where the probe is located at an adequate distance from the ends of the pipe. In other words, the pipe must be able to assume the temperature of the liquid. Short and thick-walled tubing located on large aircraft elements will tend to assume the temperature of that element (engine and the like) and are, therefore, unsuited for this measuring technique.

Similar considerations apply for temperature measurements in solid bodies, where the measurements must take place with a minimum perturbation to local temperature relationships. Where the geometry of the body permits, the bore for the temperature probe should be as narrow and deep as possible, and the ratio of bore depth to diameter should be greater than five to one where the body has good heat conductivity, and greater than about 15 to 1 where it has poor heat conductivity. If the electrical leads are placed along an isotherm on the body and can be secured to the body with good heat conductivity, the heat removed from the leads (heat conduction error) will not be removed from the probe location but from the environment in which the leads are embedded.

Both resistance elements and thermocouples (for higher temperatures) are used as temperature probes. Figure 94 shows some typical resistance probes for surface temperature measurements, and the mounting technique. Figure 23 shows resistance probes for measurements in solid bodies: tip-sensitive well-type probes in Figures 23a and b, and a special design with a spring-loaded element for cylinder head temperature measurements in Figure 23c which can be exchanged at any time for the thermocouple shown in Figure 41a (provided that the leads and the indicating instrument are switched, too). Figure 41 also shows additional probes with thermocouples, including probes screwed to pipes, screwed under spark plugs or bolt heads, and blind rivet probes. Finally, Figure 95 shows a very small thermocouple that can be made from thermocouple materials (for instance AN 30 wires or from a thermocoaxial line) and is economical especially when made in large numbers. The twisted end of the lead is spot-welded to the aircraft skin. Where the surfaces or bodies have poor heat conductivity the thermocontact is welded to a metal plate which is then attached to the body with the best heat conductance possible. Where the temperature of solid bodies is measured, this array must be embedded into the body. It must be kept in mind that newly made thermocouples should be aged prior to use by glowing them several times, since

* This can be determined through comparison with a contact thermometer, making sure that the conditions are as similar as possible to normal measuring conditions.

otherwise the calibration will change too rapidly. The use of series connected thermocouples for obtaining measurements of mean temperature is not possible, if the thermocouples are grounded.

Incidentally, special resistance probes with extremely small temperature lag errors are available for measuring temperatures on surfaces with rapidly changing temperatures (for instance, fuselage skin of VTOL aircraft in the take-off or landing phase). This is achieved through a special array of different resistances in the same element, using a balancing circuit: "Compu-Therm" principle, References 86 and 87. Special thermocouple types are available for measuring the heat flux (the so-called "heat flux" transducers, References 92 and 93), whose description would exceed the scope of this paper.

Additional information on surface temperature probes, the associated adhesives, and the like, can be obtained from the appropriate company literature.

Temperature-sensitive paints shall be mentioned here that are applied to the surface of the body in question and will show a color change when a certain temperature is exceeded, permitting the indication of temperatures in the range between about 37° and 670°C. These so-called "thermocolors" are supplied in the form of powder that is suspended in denatured alcohol and thinly applied with brush or spray gun. The drying time is about 30 minutes. The temperatures must prevail for a period of 30 minutes in order for the full color change to take place. If the time is only 5 seconds, the indicated temperature will be about 10 to 40% lower than the actual temperature. The color change will be retained for a certain period of time if the surface cools slowly*. The original color will be restored when the surface is wet. These colors can be used several times. Also, there are types with multiple color change (for instance, initial color pink, changing to light blue at 65°C, to yellow at 145°C, to black at 174°C, and to olive drab at 340°C). A similar behavior is characteristic for "Detectotemp paints" and "Thermochrome crayons"; however, the former cannot be used in the open air, i.e. on the outside of an aircraft skin (Reference 96). The Thermochrome crayon must be applied to a hot body, otherwise the color change will take place too early. If the body is above the temperature for which the paint was selected, the color change will take place immediately. Temperature indicators in the form of self-adhesive ribbons are especially well-suited for monitoring purposes, i.e.,for long term observations to determine whether a certain temperature has been exceeded. Different types of the so-called "temp-plates" (Reference 95) are available that provide from one to eight calibrated color points which assume a black color permanently when the indicated temperature is exceeded.

Finally, there are mechanical temperature indicators, for instance the so-called "templugs". These resemble grub screws (about 5 mm long, 2.5 mm diameter) which are screwed into appropriate bores in the object whose temperature is to be measured (for instance, a turbine housing). The hardness of these indicators will change as a function of temperature and exposure period. The templug is subjected to a hardness test in the laboratory after completion of the test in question which permits subsequent temperature determination with an accuracy of ±5° (Reference 13). Another comparable technique has recently been reported which utilizes capsules filled with diamond powder that has been exposed to radiation in a nuclear reactor. The change in the density of the powder is proportional to the temperature reached, and is subsequently determined through X-ray photography with an accuracy alleged to be ±3°C (Reference 102).

6.2 Selection of Thermometers

The aspects which are important in the selection of a thermometer include:

Type of medium to be measured: ambient air, engine exhaust gas, fuel, aircraft skin, equipment, etc.

Temperature range to be covered.

Measuring purpose: permanent installation or test rig.

Type of aircraft: airspeed range, space and weight considerations, etc.

Type of measurement: individual, multiple, mean value, differential, or temperature rate measurement; point or area measurement.

Type of data acquisition: instrument reading, telemetry, airborne recording, or combinations of these techniques.

* Exercise caution in the treatment of objects that are exposed to an air or gas flow!

Combination with other airborne systems such as air data computers, engine control systems; interfaces required for this purpose.
Required measuring accuracy, indication resolution, recording accuracy, measuring duration and sampling cycles in the case of intermittent measurements.
Required response: time or local resolution as a function of airspeed.
Available equipment and modules.
Available funding.
Price and availability of the required equipment.
Available capabilities for in-house fabrication.
Available calibration and testing facilities.
Type of data acquisition and data processing.
Requirements imposed on the measured data by other users.
Available time for preparations.

After considering these points the design type of the thermometer to be used is determined first: resistance thermometer, thermoelectric thermometer, temperature-sensitive paint, and so on. Since electrical thermometers are used in the majority of cases, the advantages of the two types shall be compared again at this point:

Advantages of the resistance thermometer:

Greater sensitivity to small temperature changes, resulting in a much greater output voltage.
Less complicated and inexpensive equipment for indication, recording, or signal processing for telemetry purposes.
Greater accuracy and stability (up to 30 times better over limited temperature ranges).
Failures (short circuit or breakage) are easier to discover.
Less sensitivity to noise voltages (resulting from engine vibration), parasitic thermo - voltages, and the like.
The leads weigh less and are cheaper than compensating lines; longer leads are possible for high-resistance probes.
The output voltage per degree of temperature change can be varied within broad limits through appropriate design of bridge and supply voltage, and it is easy to set up computing bridges. (However, doubling the bridge and supply voltage will result in four times the self-heating error of the probe!).
A temperature reference point or compensating device is not required.
Probes with close tolerances can be supplied so that recalibration is not necessary when probes are exchanged.

On the other hand, thermometers using thermocouples have the following advantages:

The temperature range that can be covered is greater.
An external current source is not required if directly indicating instruments are used.
Thermocouples for many applications can be made in-house at low cost.
Thermocouples have greater stability than resistance elements at high temperatures.

Consequently, these two types supplement each other, the resistance thermometer having a clear advantage in the temperature range from about -100° to $+250^\circ$C in most instances, while only the thermocouple has the necessary characteristics in the range from about $+700^\circ$ to 1500°C*.

Having decided on a certain thermometer type it is necessary to select the probe and the indicating or interface system for telemetry or recording facilities, in accordance with the considerations outlined above. Where it is necessary to use some equipment that is already available, consideration must be given to the fact that the system can be no better than the component which contributes most to the measurement error. The cause of errors named in Section 5.2 must be considered when new equipment is procured. This means, for instance, that in the case of a resistance thermometer

* Tungsten - Rhenium thermocouple up to $+3100^\circ$C.

the probe with the highest possible resistance must be mated to an instrument that will not cause excessive self-heating in the probe. Moreover, manufacturer's information must be critically examined to fully understand its applicability to the measurement requirements for which the component has been or will be procured. Caution is indicated in the case of companies that are very sparing with data and documentation or cannot state the test conditions under which a certain characteristic (such as self-heating) was measured*. This type of advertising may be indicative of an effort on the part of the contractor to expand into new areas, such as sensors, indicating instruments, and the like, without specialized knowhow and suitable test equipment, nor even an in-house development activity that performs measurements in aircraft itself. "Cheap" products can turn out to be very "expensive" when the measurements are finally obtained.

In this connection we must also discuss the in-house fabrication of probes, bridge circuits, and the like. This used to be a common practice since, among other reasons, suitable signal conditioners and indicating instruments with recording outputs were not commercially available. The most frequent shortcomings of this class of equipment are: unacceptably high self-heating of the probe as a result of improper dimensioning of the bridge circuit, the temperature dependence of the dc amplifiers used, inadequately stabilized supply voltages, and the like (cf. Section 2.2 and 3.3). It is recommended therefore that modern servoed instruments with independent measuring outputs (potentiometer, encoder, etc.) be employed in every instance where exact temperature measurements in air and other gases are required. In-house development of comparable equipment requires years of specialized experience and is time-consuming and, therefore, in most cases more expensive.

Prior to the acquisition of probes, signal conditioners, and the like, consideration should be given to the cost effectiveness of procuring expensive components, with more stringent specifications, in the interest of lowering the cost of other operations. As an example, one would expect that in a better system, the number and complexity of the corrections required would be reduced, and this in turn should lower the cost of post flight operations. In addition, the results should be more accurate and easier to evaluate. (cf. Section 6.4)

6.3 Calibration of Thermometers

The characteristics of the thermometer and all the associated recording and/or indicating devices must be known before any accurate temperature measurements can be made. This means that all errors must be identified and defined for all flying states that are of interest. Measures required for this purpose can be subdivided into three stages:

(a) Error determination on individual components in the laboratory.

(b) Error determination of the entire thermometer after installation in the aircraft.

(c) Error determination while airborne.

As a result of the interdependence of many errors and the influence of many parameters that are difficult to evaluate, all experiments and measurements must be planned and executed so that the errors are acquired individually wherever possible. The measured data must be available in a form that can be evaluated directly, for instance, as a function of Mach number and altitude, so that time-consuming conversion routines will be eliminated.

Test facilities available to instrument engineers during flight programs are normally rather limited. Therefore, the following discussion neglects those measurements that would require special facilities such as wind tunnels, etc. However, decade resistance boxes, temperature calibration baths with precision thermometers, digital multimeters, and a temperature test chamber with a range from, for instance, -80^{o} to $+80^{o}C$ (but at least from -20^{o} to $+80^{o}C$) must be available. A thermocouple thermometer tester, a small electric furnace and a "Jetcal analyzer" with all accessories should be available for testing thermocouples.

6.3.1 Calibrations in the Laboratory

As a rule, only the following activity can take place in the typical laboratory or test shop:

* This applies also to information such as "recovery factor = 1.0", "self-heating so small that it can be disregarded", etc.

(a) Measurements of Resistance Probes

The probe calibration error E_{CAL} is determined for the temperature range of interest in a calibration bath agitated at constant rate, by comparing the acquired resistance values to the appropriate table of standard values (for platinum, nickel, etc.) and to the liquid temperature determined with a precision thermometer. Although the self-heating effect is much smaller in liquids than in air, the probe should be operated at the smallest measuring current possible (for instance, 1 mA for Ni and Pt probes, and about 0.1 mA for thermistors), in order to be certain that the self-heating error will remain below a level of 0.1°C.

Additional measurements can be performed in the laboratory on resistance probes used to measure temperatures in liquids. The prerequisite is that the same liquid be used that will be measured by the probe installed in the aircraft and that the liquid flow velocity be the same as prevailing at the intended probe location. The heat conduction error E_C can be determined, for instance, by completely immersing the temperature-sensitive part of the probe in an agitated liquid bath at, for instance, +100°C, and packing the head of the probe in ice. The difference between the measured temperature and the temperature of the liquid will yield the heat conduction error for a temperature difference of 100°C between liquid and probe head. Repetition of this test at several lower temperatures (such as 80°C, 60°C, 40°C, 20°C and 10°C) is advisable because the heat conduction error is a non-linear function of the temperature differential. Detailed instructions for determining the heat conduction error are provided in Reference 25.

The temperature lag error E_L can be determined by rapidly alternating the probe between two agitated baths of different temperatures, measuring the time required to indicate 20%, 50%, 63.2% and 90% of the temperature difference (cf. Section 5.2.3).

The self-heating error E_{SH} can be determined for a given type of liquid and liquid velocity at a desired temperature provided the resistance of the probe is first measured with no significant self-heating (using, for example 1 mA for Ni and Pt probes, and 0.1 mA for thermistors). The current is then increased to ten times the initial value or more. A milliammeter is placed in series with the probe, and an electronic millivoltmeter of very high input resistance is placed in parallel. Assuming the temperature increase was 1.50°C and took place at a current of 10 mA and a voltage of 5.00 V, i.e. at a power of 50 mW, the self-heating effect will be 1.50°C/50 mW, or 0.03°C/mW. Data obtained for a given liquid and a given liquid velocity are almost impossible to extrapolate to other liquids (or gases) at the same or different velocities, since they are also dependent on the probe design and will yield different factors for each probe type.

A rough estimate applicable to certain well-type probes is that the time constant can be about 30 times greater in air, about 2 times in oil, and about 1.5 times greater in liquid oxygen than in water; under otherwise equal conditions the self-heating effect in °C/mW can be about 100 times greater in air, and about 2 times greater in oil and in liquid oxygen than in water, provided a liquid velocity of 5 ft/sec is assumed in every instance. The actual values of other probe types can deviate quite considerably from these relationships.

The calibration of probes for surface and solid body temperatures can be acquired in liquid baths, too; however, measurements of self-heating, temperature lag, or heat conduction are meaningless if performed on individual probes before they are installed, and in most cases, difficult to obtain when the probes are installed on the solid body.

In the case of probes for air and gas temperature measurements little more than the calibration error can be determined in the laboratory. All other errors must be determined in the aircraft (unless suitable wind tunnels are available), since they are dependent on flight conditions (cf. below).

(b) Measurements of Thermocouples

This discussion of the measurement of thermocouple characteristics will be confined to the determination of the probe calibration error E_{CAL}. In using furnaces special care must be taken to assume that the immersion depth is adequate and that attention is given to any effects caused by the high temperature gradient in the region where the leads are passed through the furnace wall (cf. Section 3).

Calibrations using the "Jetcal analyzer", which is discussed below, may prove to be considerably better. A detailed description of the calibration of thermocouples is found in References 13 and 42.

(c) Calibration of Instruments, Signal Conditioners, etc.

The following information on the measurement of instrument characteristics also applies to signal conditioners, telemetry, and recording apparatus. It should be mentioned at this point that these devices, unlike the usual practice with probes, are not always supplied in aged condition. Therefore, it is advisable to operate new equipment for an uninterrupted period of 24 hours, if possible at an increased ambient temperature, before proceeding with its calibration*.

Laboratory tests that can be performed on this equipment include the following:

(a) The instrument scale error E_{SC}, at room temperature (approx. +20°C) and with the specified lead resistance, can be determined by substitution of a precision decade resistance box for the probe**. A warm-up period of at least 30 minutes must be allowed before this measure is made.

(b) Simulate a step change in the temperature at the probe by switching between two resistances that correspond to the probe resistances at the lower and upper limit of the measuring range, to determine the lag of the instrument (or the follow-up time in the case of servoed instruments).

(c) Simulate very small temperature changes at the probe by introducing very small changes in the connected resistance to determine the resolution of the instrument. Repeat this test while lightly striking the instrument to determine the effect of friction inside the instrument.

(d) Turn the equipment off for a period of not less than 30 minutes (until the instrument has returned to room temperature), and measure the time required to achieve a constant indication for three different inputs, perhaps 30%, 70%, and 100% of full scale, to determine the warm-up time. Let the instrument cool down between each of the three measurements.

(e) Repeat the measurements described under item (d), above, at -20°C and then at +60°C, after placing the instrument in a temperature-controlled chamber***. Leave the resistance box outside. These tests can be used to determine both the maximum warm-up time that must be allowed before measurements can begin, and particularly the instrument temperature error. This error is not normally specified, rather, the errors are determined for a constant input at the lowest, intermediate, and highest temperature for which the instrument is designed. These are combined and presented as a mean scale error so that this error appears as a systematic error with a certain dispersion range (cf. Section 6.4.1).

(f) If all components, i.e., the actual wiring, plugs, circuit breakers, etc. are available prior to installation in the aircraft, it is also possible to determine the electric lead error E_W. This error is variable, depending on the indicated temperature, especially in the case of simple instruments. More details are provided in the following chapter.

6.3.2 Flight Line Calibrations

Unlike the individual measurements in the laboratory that were discussed in the preceding section, the thermometer system as a whole can be tested "in situ" on the flight line.

(a) Resistance Thermometers

The final calibration of the total thermometer system can be performed by replacing the probe with a decade resistance box. In this procedure, the following comments may prove helpful. Care should be taken to use connections which are as short as possible and which are made using AN16 gauge wire, or larger****. This precaution will help avoid introducing additional electrical lead errors. In

* It was found in the case of certain types of servoed equipment that this break-in period reduced the warm-up time from 40 minutes to 8 minutes.

** Its loading capacity should be greater than 1 watt, if possible, to avoid self heating of the resistances.

*** These values depend on the ambient temperatures the instrument is expected to experience in the aircraft.

**** If the probes have very low resistance, the resistance of these lines must be deducted from the value of the decade resistance.

selecting resistance box values, if the actual probe calibration is used in lieu of a standard calibration curve, then probe replacement will require a re-calibration.* This procedure will, however, reduce the corrections necessary when the raw data are processed. As an example, some indicators are equipped with externally-accessible trim pots which are used for line compensation and/or scale adjustments. When these are properly adjusted, the values of indicated temperature are as close to the corresponding values of actual temperature as the probe and indicator calibration anomalies will allow. This combined correction may not be suitable over the full range of the scale, because, for example, of different non-linearities in the calibrations of the probe and indicator. In this event, a decision must be made regarding the maximum allowable deviations between indicated and actual values over the portion(s) of the scale which are of prime interest. This procedure is illustrative of the steps that can be taken in setting up a system to facilitate the evaluation of the data acquired during test runs. Ideally, in this instance, for example, the only additional errors that would have to be accounted for would be those dependent on the flow velocity.

In the case of thermometers used for ambient temperature or compressor inlet temperature measurements (OAT or CIT measurements), additional testing on the flight line is hardly possible, except for functional tests in which case the probe deicing heater must be turned off. On the ground, the indication will almost always deviate by some amount from the static air temperature, partly because the time constants in this state are very long, perhaps 15 minutes and more. If the probe heating system (deicer) were turned on, the temperature indication would rise to more than 100°C, and in probes with open-wire elements it is likely that the element would be damaged.

A functional test, in addition to the calibration described above, can be performed on thermometers for measurements in liquids, on surfaces**, and in solid bodies.** In some cases, the engines must be running for this functional test in order to simulate flight conditions. Also, comparison measurements might be taken in this state in order to determine the effects of self-heating or heat conduction errors. Many inexpensive thermometers designed for measurements in liquids allow for the self-heating error of the probe in the calibration of the instrument. In that case, the scale calibration will apply only for the liquid in question and for a certain velocity of the liquid flow. When the engines are turned off, the indication will fluctuate and become unreliable, especially when natural convection sets in, because the probe is subject to appreciable self-heating. A number of good thermometer testers are ailable, but the operating instructions which are provided do not always adequately discuss all potential sources of error (especially self-heating).

(b) Thermometers with Thermocouples

The most important application for thermometers of this type is in the measurement of the engine exhaust gas temperature (EGT). In this case, it should be possible to simulate most of the operating conditions existing in flight by running the engines on the ground. However, the determination of the actual measuring accuracy of thermometer arrays in an engine is very difficult under these conditions.

A good aid in this case is the so-called "Jetcal analyzer" or comparable apparatus. This set permits, among other procedures, a functional test of the entire EGT system with non-running engines, testing each thermoelement and the whole array for continuity, short circuits, etc. Generally, the entire EGT system can be tested with an accuracy of about $\pm 4^{\circ}$C. The different capabilities of the set can be seen from the operating manual supplied with it.

6.3.3 In-Flight Error Calibration

6.3.3.1 General

In the majority of temperature measurements, e.g., in liquids and solids, in-flight error evaluation is confined to a check against the results obtained in the laboratory and on the flight line, and/or to the determination of that probe location which will result in the minimum position error. Optimum probe locations can only be determined in-flight since these conditions cannot be adequately simulated on the ground. The procedure generally used involves the installation of thermometers at different locations to measure the same parameters. The outputs are read out either individually or

* This does not hold for probes with close tolerances.

** Provided the air flow about the aircraft that exists in flight does not play any decisive role.

commutated onto a single channel and read sequentially. The probe that shows the least position error is used in succeeding flights.

The determination of the position error of probes for OAT and CIT measurements using in-flight techniques is, at times, justifiable because the resulting flow, pressure, and temperature relationships at the probe or at the measuring element can be created only in flight. Since position errors are dependent on altitude and airspeed or Mach number, the errors in the aircraft's pitot-static system must first be determined in special instrumentation flights. Techniques for obtaining these calibrations are described in the conventional flight test manuals. However, if a reference thermometer that could be used as a "standard" (cf. below) is not available, then only those procedures wherein the static air temperature is determined by a thermometer on the ground or by a radio sonde at flight altitude can be used. To perform thermometer calibration runs at different airspeeds and the same pressure altitude, or correlate calibration measurements obtained in different flights and in different flight regimes, it is very important that accurate measurements of the static pressure position error as a function of airspeed are available. In the presence of unfavorable meteorological stratification in the troposphere, even small deviations from equal pressure altitude can result in large changes of static air temperature. The differences between the true Mach number and the indicated Mach number (which is subject to pressure errors) as a rule have a minor effect on the accuracy of error determination. However, they cannot be disregarded, especially in the near-sonic airspeed range, since the pressure error in this flight range can assume respectable magnitudes. Figure 99 shows the effect of a relative error of $\Delta M/M = 1\%$ on the determination of static temperature as an absolute magnitude and as a relative temperature error.

All calibration flights to determine the pressure and temperature error of sensors on the aircraft should be carried out with due attention to suitable meteorological conditions. This will pay off in terms of less dispersion of measuring errors. Instrumentation programs of this type can be accomplished quicker at test sites having favorable climatic conditions.

The evaluation of weather situations is based primarily on the vertical temperature stratification and, secondarily, on the vertical structure of the winds. The data required for this evaluation are supplied by the radio sonde stations of the national weather service located in or about the test area; these are acquired at the standardized world-wide times at 00.00 GMT and 12.00 GMT. Where this information turns out to be inadequate, special radio sonde or aircraft missions must be accomplished to determine the temperature as a function of pressure altitude.

In any event it is better to determine the meteorological stratification beforehand, and then carry out the instrumentation flights in the suitable strata, rather than perform the instrumentation flights at preplanned altitudes without meteorological information. The risk inherent in the latter approach is having to discard the data because of excessive meteorological errors which, of course, can render the data useless for error determination.

The following types of vertical stratification are suitable for the determination of pressure and temperature measuring errors under in-flight conditions:

(a) Temperature structure: Attempts should be made to locate areas where the isothermic vertical stratification, i.e., the vertical temperature gradient, is $0^{\circ}C/100m$, or as close as possible to this value. Measurements should never be performed at gradients in excess of $-1^{\circ}C/100$ m (dry adiabatic temperature gradient) or $+1^{\circ}C/100m$ (inversion). In the presence of stratification of this magnitude, an altitude change of $\pm 10m$ (± 30 ft) will cause a temperature change of $0.2^{\circ}C$, which is in the order of measurable errors. It requires special skill on the part of the pilot to maintain altitude with an accuracy of this order, especially where steady flight conditions must be established at relatively high airspeeds.

Thermoconvective weather situations are unsuitable. These are situations with vertical interchange caused either by radiation via a heat increase of the ground (boundary or exchange layers) or by a mixture of air masses of different energy levels. Periods of minimum exchange are favorable for measurements, especially at low altitude. These are the periods where the direction of radiation is changing, as well as all weather situations with dense cloud stratification and no convection phenomena.

Moreover, there must not be any precipitation, unless it is intended specifically to test the effect of the liquid water phase on the airspeed error.

Where a radio sonde is used as a reference for evaluating the stratification, it must be kept in mind that the sonde in most cases will rise at a vertical velocity of about 6 m/s (17 ft/sec) and that the probes in this sonde have a time constant of a few seconds. Therefore, the temperature boundaries will be "blurred" considerably, which complicates the evaluation. This applies especially where radio sonde data are used as reference values to calibrate the aircraft instruments. Under these conditions it is advisable to consult a meteorologist for an evaluation of the true stratification.

The radio sonde itself must be compared to a precision thermometer prior to takeoff. This thermometer should be exposed to a constant air flow of, for instance, 6 m/sec, which corresponds to the climbing rate of the radio sonde, by means of a built-in blower. Also, the time interval between the radio sonde measurement and the in-flight measurements should not be excessive (if possible less than 30 minutes), and the launch site of the radio sonde should be located within the test area, if possible, to minimize the effects of temporal and spatial variations.

(b) Wind field structure: The tests should be conducted in strata of constant wind speed and wind direction, if possible. A vertical wind profile can be plotted from rawin sonde measurements, and this will show immediately which strata have acceptable shear values. Pronounced vertical or horizontal shear lines with associated curvature of the wind profile will result in wave, crest, and rotor formation. In the presence of these boundary area phenomena, the isothermic layers also have wave or crest structures, depending on temperature stratification and shear characteristics of the wind, with wave lengths ranging from about 0.5 km to 40 km. These structures will sometimes occur even in a cloudless sky and as high as the tropopause, however, they can be detected from the characteristic wave structure of the temperature record.

Figure 103 shows a schematic diagram of the strata that are suitable for temperature calibration flights in the lower atmosphere, referred to the vertical temperature stratification (A), wind velocity (B), and horizontal wind direction (C). It is relatively rare to find good conditions with respect to all three effective components in one and the same altitude layer, and this can be expected only near the center of high pressure areas.

The different effects must be considered to determine if existing weather conditions are acceptable.

6.3.3.2 Explanatory Examples

In order to show the difficulties encountered, we shall first describe the "simple" case of determining the position error of a TAT probe without deicing heater, combined with an indicator or signal conditioner that causes probe self-heating power to be 40 mW. The separation of the individual components of the position error will not be considered in this example. It shall be assumed that the maximum altitude is 40,000 ft. and a proven type of TAT probe is installed at a highly favorable location (nose-boom). Then the only effective components of the position error should be the velocity error and the self-heating error, which partially cancel each other. In order to reduce the number of measuring points and the duration of the calibration flight, data will be taken only at 3 different altitudes with increments in airspeed of about Mach 0.1. A radio sonde must be launched concurrently with the flight program. Its temperature readings will provide the static air temperature at the altitude where the tests are accomplished*. The position error is obtained, for instance at Mach 0.5 and an altitude of 20,000 ft., as shown below:

Static temperature T (from the radio sonde)	249.00° K
Total temperature $T_T = T(1 + 0.2\ M^2)$	
$= 249\ (1 + 0.2 \cdot 0.5^2)$	$= 261.45^\circ$ K
Temperature measured in the aircraft T_{Tic}	
(corrected for instrument error)	261.68° K
Position error $E_P = T_{Tic} - T_T$	
$= 261.68 - 261.45$	$= +0.23^\circ$ K

* Example: According to the standard atmosphere, a pressure altitude of 20,000 ft. corresponds to an atmospheric pressure of 465 mb. Consequently, that temperature value must be used which was measured simultaneously with a pressure of 465 mb.

The theoretically expected result of such a series of measurements is plotted in graph form in Figure 100. The practical result of a single series of instrument readings would not, as shown in this figure, yield the approximately parallel curves but curves that appear to be crossing each other at random. There are several reasons for this. Even in the absence of measurement errors, local temporal variations of static temperature may amount to about $\pm 0.5^{\circ}C$, especially in the lowest altitude strata (Reference 104). This means that this effect alone, can be several times greater than the difference between the measured values at a certain Mach number. Therefore, any attempt to accomplish such calibration flights under unsuitable meteorological conditions would be meaningless. Although modern radio sondes have reliable tolerances as narrow as $\pm 0.5^{\circ}C$ to $\pm 0.7^{\circ}C$, some of them are subject to considerable temperature lag errors. If the data from several calibration flights were combined*, a considerably better picture could be obtained. However, it should be kept in mind that such calibration flights will take about two hours. Even if the radio sonde were launched in the middle of each flight, appreciable temperature errors can occur due to changes in the air temperature during the flight. With a very limited number of measuring points in the area of greatest interest for the relevant aircraft under normal flight operations (shaded area in Figure 100), exact test results are difficult to obtain, as atmospheric temperature variations exceed the error to be measured.

Figure 101 shows the acquisition of the deicing heat error as an individual component of the position error through six calibration flights performed at low altitude. These must be repeated at different altitude levels, of course, because of the altitude dependence of this error. Simply connecting the measured values obtained in a single flight (corrected for probe calibration error) would yield a fairly irregular line. The same would still apply for a mean of two flights so that the practical application of these curves requires considerable smoothing.

This example was selected in order to demonstrate how one individual error can be isolated from the others. The test can be performed by using two TAT probes (A and B) of the same type and connecting them alternately via a commutator to the same servoed indicator. Probe A is continuously operated with probe deicer on during the first half of the flights, while probe B remains unheated. During the second half of the calibration flights, probe B is continuously heated, while probe A remains unheated. If both probes are subjected to the same flow (for instance on the underside of the nose at a lateral distance of 25 cm from each other), and if the leads between the probes and the commutator have been adjusted to exactly the same resistance values, all errors will have the same effect on the measurements taken with probes A and B. These errors then cancel out when the difference between the measured values of A and B is taken. This difference then is the deicing error. It should be noted that this does not allow for the individual calibration errors of probes A and B. These corrections must be applied before plotting the measured data.

If only one probe were available, it would be necessary at each increment of altitude and airspeed, after stabilizing the corrected airspeed and altitude indications, to make 60-second measurements with and without the deicer. Sufficient time should be allowed for the probe housing to cool off completely before attempting runs at the next airspeed. Where this method is used, changes in the air temperature during the calibration run appear directly in the measured data. These measurements then become meaningless if the atmospheric temperature changes exceed the deicing heat error. In these cases the use of a radio sonde would not yield any advantages.

The determination of the velocity error shall be discussed as another example. In earlier times this was done by measuring the recovery factor r and plotting the corrected temperatures measured at different airspeeds as a function of the square of the Mach number (Figure 102a). Since the recovery factor is a relative value, a single flight at different airspeeds and constant altitude, where a uniform temperature over the calibration run can be expected, will be sufficient. In many instances this was accomplished by increasing the airspeed at constant acceleration (such as 1 g) until the aircraft's maximum airspeed was attained, and then decreasing the airspeed at idling engine

* Cf. Section 6.4.

speed until the initial value was reached. Since the lag error E_L in the accelerated and decelerated parts of the flight has opposite signs, the mean of the two series of measurements was used to determine the recovery factor. The static temperature was determined by extrapolating the measured data to zero airspeed. Then through this point a line was plotted corresponding to the total temperature T_T in question ; this line represents a recovery factor of 1.0*. The ratio of the temperature increase for $r = 1.0$ to the actual temperature increases at the same Mach number will yield the recovery factor of the probe. Subsequently the velocity error can be computed in every individual instance with the aid of equation (64) (cf. Section 5.2).

Obviously, in the majority of cases this method provides a means of finding the velocity error which is at best, only moderately accurate. More accurate data would depend upon satisfying all of the following conditions:

1. The recovery factor must be constant over the airspeed range under consideration;
2. The self-heating error must be so small that it can be disregarded,** or it must have been acquired as a function of altitude and airspeed in the previously described calibration flights, and corrections for such factors as the instrument error must be applied before plotting the measured data;
3. Where data from acceleration flights are used, the acceleration and deceleration must be equal at the individual Mach number increments, which is hardly possible;
4. The dependence of the static pressure position error on the airspeed must be taken into consideration with respect to maintaining constant pressure altitude;
5. The altitude should not exceed 40,000 feet because of the possible radiation error (especially at supersonic flights).

Apart from the fact that the first condition will never be obtained for TAT probes above a certain Mach number (such as 0.7), it is not satisfied by many probe types in the lower subsonic range, either. Additionally, if only the values at two airspeeds, or for a narrow range of airspeeds were used, the self-heating and the deicing errors would not be properly accounted for.

Figure 102a shows the measured data of a TAT probe that has a constant recovery factor of $r = 0.97$ below Mach 0.7, and the measured data of a cylindrical probe with recovery factors between 0.52 and 0.74, located in flow perpendicular to its axis. If, in the case of the latter probe, only the measured data between Mach 0.7 and 0.85 were used, the resulting static temperature would be 246.5°K instead of 253.0°C, and, at Mach 0.8, the resulting recovery factor would be 0.90 instead of 0.72. Figure 102b shows a phenomenon found in many calibration records where the measured data rises above the line of $r = 1.0$ at small Mach numbers. This infers that the recovery factor can become greater than 1,however, in this instance, the data have not been corrected for self-heating errors. Further, in the range from Mach 0.1 to 0.7 the probe used to acquire the measured data does have a constant recovery factor of 0.97. Consequently, if the plotted measured data are not corrected for self-heating error, and if the line placed through the measured data is extended, the static temperature between Mach 0.2 and 0.5 will show a false value of 249°K instead of the correct value of 248.0°K in this case. If the line for $r = 1.0$ is placed through this false value also, the uncorrected measured data will yield an apparent value of $r = 0.91$ for the recovery factor which is also false.

Consequently, this method to determine the velocity error can be recommended only if all the prerequisites outlined above are truly applicable.

In another method of experimentally determining the velocity error, temperature measurements are made at least five times at each of three airspeeds. These must be accomplished at the same pressure altitude. These data are compared with the total temperature calculated using radio sonde measurements. The procedure, then, is as follows:

From Pressure Altitude and Calibrated Airspeed:	Mach Number M
From Mach Number and Static Temperature (Radio Sonde):	Total Temperature T_T [°K]
T_T [°K] - Mean Value of Measured*** Temperatures [°K] =	Velocity Error $E_{V\,mean}$ [°K]

* Cf. Figure 110, for $r = 1.0$.

** Through using a modern servoed instrument.

*** Corrected for self-heating and deicing effects.

The prerequisites for the application of this technique are the same as described above with the supplementary condition, that the flights be performed under ideal weather conditions.

Considerably better accuracy can be achieved through wind tunnel measurements. In most cases it is worthwhile to acquire a probe which has been calibrated in the wind tunnel for use as a reference probe for in-flight calibrations, since the implementation and evaluation of calibration flights will cost time and money. If the recovery error and the velocity error* are accurately known (equations (56), (57), and (66) to (69)) acquisition of the other errors will no longer pose any major problem (cf. below), since the accurate value of static air temperature is not necessary for their acquisition.

6.3.3.3 Practical Implementation

The following conclusions, among others, can be drawn from the examples discussed above for the acquisition of the position error of OAT and CIT probes:

(a) Where good TAT probes are used, the magnitude of the position error will have approximately the same order of magnitude as the measuring uncertainties associated with the inaccuracy in determining the Mach number (for the computation of T from T_T), by inhomogeneities in the temperature stratification of the atmosphere, by difficulties in maintaining constant altitude at different airspeeds, by inaccuracies of the radio sonde used as a reference, and by other causes**. Therefore, accurate determination of the position error is extremely difficult to obtain experimentally.

(b) Some of the instructions for flight calibration of ambient air thermometers that can still be found in flight test manuals, which appear to be fairly simple, are based on the state of the art as it existed 25 years ago, and in addition, contain unacceptable simplifications (neglecting the self-heating error, deicing heat error, etc.).

(c) Simultaneous data acquisition from a thermometer used as a standard and the measuring system to be used in the aircraft will permit the elimination, to a large degree, of most of the measuring uncertainties mentioned in paragraph (a) of this section (inhomogeneities in the atmosphere, dispersion of radio sondes, and the like).

(d) The most accurate information possible on the magnitudes of the individual components of the position error of the TAT probe used as a reference is required for flight testing purposes, and in this respect the acquisition of the velocity error involves the greatest difficulties under practical circumstances, since the exact value of static temperature is needed for every measurement here. For this purpose it is usually worthwhile to procure a probe calibrated in the manufacturers'*** wind tunnel. Sources for this type of instrument can be only those companies that perform this type of calibration as a matter of routine, publish their data, and can guarantee the performance of their products.

(e) Where it is necessary to depend on company information, it must be kept in mind that the stated data were acquired only in the wind tunnels available at that time, and on normal

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* In the case of TAT probes (η<0.005) the approximate formula $E_V = \eta \cdot T_r$ applies and is presented in graphical form in Figure 111. In this graph, the velocity error can be determined directly from the recovery temperature T_r and the value of η, which is a function of Mach number.

** Where simple probes are used in the aircraft's boundary layer, the position error at high airspeeds is relatively large and, therefore, apparently easy to acquire from the point of instrumentation; however, this also means that its fluctuation range is very large, depending on the flow about the aircraft, which is in turn dependent on the angle of attack, gross weight, configuration, altitude, airspeed, and other parameters. Therefore, using these probes is not meaningful at high airspeeds, since suitable correction procedures are too complicated for practical use, in addition to being excessively inaccurate (cf. Figure 106).

*** Subsequent calibration of a probe in a wind tunnel requires considerable experience and in turn demands the availability of a reference probe to determine the true temperature relationships in the test section.

aircraft. Information on in-flight data outside the normal flight corridor, for instance at Mach 0.1 and 80,000 ft., could at best be acquired through computation and must, therefore,be considered with great caution. On the other hand, information for the normal flight corridor is usually conservative, since compliance with these data must be proved to certain consumers in very stringent test procedures.

(f) For flight testing purposes the position error of the temperature probe is of interest with respect to all those combinations of altitude and airspeed within the capability of the test aircraft. However, in practical flight operations many aircraft in most cases remain within a narrow range about a nominal flight profile (cf. the shaded area in Figure 100) which makes it possible to use greatly simplified corrections for the indicated data.

A suitable reference thermometer, which has been mentioned several times is a TAT probe, Type MS 27 188 - 1 (Model 102 CD 2U), calibrated for the recovery error*, with a hermetically sealed Pt element of 500 ohms having close tolerances, and a servo instrument with recording outputs (such as BH 180 or BH 187 "Auto-Temp" by Howell Instr. Co., or the Temperature Servo Indicator 25001 by Betzelt GmbH Muenchen,Germany). They have an indicator accuracy of $\pm 0.2^{\circ}C$ and a self-heating effect so small (less than 0.04 mW) that it can be disregarded**. Consequently, the instrument error (including probe calibration error) is about $\pm 0.23^{\circ}C$, provided that the system was carefully corrected for lead resistances after installation into the aircraft (cf. Section 6.3.2). When this type of probe can be installed at a favorable location (such as a nose boom), and when the probe deicer is turned off, only the velocity error is significant so that the indication corresponds directly to the recovery temperature, T_r. The associated absolute value of the velocity error E_V is easily determined from the values of the recovery error, and the recovery temperature, which are dependent on the Mach number ($E_V \approx \eta \cdot T_r$ [$^{\circ}K$], cf. Figure 111). Consequently, the temperature values read, or recorded, using this thermometer, require only a correction for the velocity error in order to acquire the associated total temperature as a reference value for other thermometers that must be calibrated.

Subsequently, during calibration flights, direct comparison of the reference thermometer with the thermometer to be calibrated will yield the overall position error of the latter as well as its individual components. The tabulation below illustrates a distinction between two cases of interest. In case (a) the stipulation is that the probe to be calibrated has the same nominal resistance (for instance, 500 ohms) and the same characteristic*** as the reference probe. In that case as shown in Figure 104a, a commutator should be used. In case (b) it is assumed that different probe resistances or characteristics exist so that it is not possible to commutate. In both cases, it is stipulated that the measurements are performed at constant airspeed and altitude**** so that the lag error can be neglected. Practically, its magnitude could be determined with adequate accuracy only in an appropriately equipped wind tunnel.

Overall position error*****	(1a) as in (1b), no commutation. (1b) Comparison of the indication of the reference thermometer, corrected for the velocity error (= total temperature), with the indication of the thermometer to be calibrated, corrected for instrument error.

* Less favorable than wind tunnel calibration is in-flight calibration performed with the aid of a radio sonde, as described above.

** The Betzelt indicator also has provisions for switching to a Mach-function potentiometer for direct indication of Static Air Temperature (SAT) (cf. Figure 33).

*** Caution! Normal platinum probes and platinum probes with narrowed tolerances (PCI elements) have different characteristics; the same applies, for instance, to 100 ohm Pt probes and 100 ohm Ni probes.

**** These altitudes are selected after evaluation of the radio sonde mission that is nearest with respect to time and location.

***** Deicer turned off on both probes.

Recovery error * ** (from the velocity errors at one and the same altitude)	(2a) Reference probe and calibrated probe connected alternately to the reference indicator. Comparison of the indication of the reference probe, corrected for velocity error (= total temperature), with the uncorrected indication of the probe to be calibrated (= recovery temperature). (2b) Comparison of the indication of the reference thermometer, corrected for velocity error, to the indication of the thermometer to be calibrated, corrected for instrument error. If the latter causes appreciable self-heating, the sum of velocity and self-heating error will be obtained.
Self-heating error* **	(3a) Through a combination of methods (2a) and (2b): the self-heating error will be the difference between the two measurements on the probe to be calibrated. (3b) The self-heating error of the probe can be determined by itself with the aid of a special "test thermometer"***. The values for the thermometer to be calibrated at all altitudes and airspeeds are obtained from the determined self-heating (SH) in $^{\circ}C/mW$ and the probe loading P (in mW) of the thermometer to be calibrated, which are multiplied (SH • P).
Deicing heat error	(4a,b) Alternating operation of the probe to be calibrated without and with deicer. The difference between the two indications directly yields the desired error for the altitude and airspeed in question.
Probe location error*	(5a) Reference probe and calibrated probe connected alternately to the reference indicator. The difference between the indication with reference probe, corrected for velocity error, and the uncorrected indication with the probe to be calibrated, yields the sum of velocity error and probe location error (possibly preceded by method 2a; caution, different probe locations!). (5b) (perhaps possible through a combination of the methods previously named).

* Deicer switched off on both probes.

** The probe to be calibrated must be installed at an ideal location, too, in order to avoid probe location errors.

*** A test thermometer for self-heating consists of a bridge circuit where a double-pole commutator can be used to connect either a large series resistance to the bridge and a high-resistance shunt parallel to the instrument, or a small series resistance and a low-resistance shunt. These resistances are so selected that the same scale calibration results in both switch positions (testing with maximum load test resistances!), but that the current flowing through the probe will change at a ratio of, for instance, 1 to 40 and hence a self-heating effect of 1 to 1600 (Reference 31A) will develop. Different indications will be obtained when a probe is connected, depending on the commutating switch position (without or with self-heating). For instance, an increase of the indication by 2.0°C for a given situation under a probe load of 200 mW, will result in a self-heating effect of 0.01°C/mW in this case.

If the procedures described above are used to determine the velocity error at high altitudes or at high supersonic airspeeds, the measured data will also include the components of the conduction and radiation errors. In that case the latter cannot be separated from the velocity error. The results of these measurements, that is the position error or its individual components, are plotted in Figure 107 for two different altitudes as a function of the Mach number.

New developments of flight logs (Dornier AG, Germany) have recently become available for flight testing purposes that produce the true airspeed (TAS) directly, with high accuracy. Error acquisition on ambient air thermometers using the TAS as the airspeed parameter instead of the Mach number might afford some advantages. This subject was omitted here for reasons of space, among others, and only two diagrams were included. Figure 112 shows the interrelationship between pressure altitude, indicated airspeed, Mach number, and true airspeed at standard atmospheric conditions. Figure 113 shows the absolute and relative error of true airspeed as a function of prescribed errors of the Mach number, static pressure, and static air temperature. This diagram shows, among other features, that a 1% error in Mach number or an 0.5% error in static temperature (in ^{o}K) will result in a 1% error when computing the true airspeed.

6.4 Data Evaluation

6.4.1 Error Handling

The acquired data are best processed in the systematic order explained in Section 5.3, using a specific sequence of steps and the standardized designations for measured data, subject to certain types of errors or corrected for certain types of errors. This systematic order is repeated in Figure 105 for the case of an ambient temperature measurement. The left-hand column indicates the build-up of the temperature indication as a result of the combined action of the existing physical situation (static temperature and kinetic self-heating) and the measuring errors (instrument error, temperature lag error, and position error). The center column indicates the correction routines taking place in an air data computer, and, in the right-hand column, the correction routines performed manually with the aid of tables, diagrams, and the like. The same pattern applies to the correction of recorded data. This figure also indicates the interrelationship between true values, errors, and data acquired through measurements that are subject to error:

Erroneous value = true value + errors

true value = erroneous value − errors

= erroneous value + corrections.

Consequently, the correction value is equivalent to the absolute magnitude of the corresponding error value, but has the opposite sign.

Other frequently used error magnitudes include the following terms and definitions:

(a) Accuracy is defined as the degree of agreement between the measured value and the true value. However, in most cases it is stated as "inaccuracy", that is, the difference between measured value and true value (or absolute value), or as the percentage ratio of inaccuracy to true value or to the maximum value of the measuring range (that is, a relative value).

(b) Resolution (or precision) is defined as the magnitude of the minimum change to which a measuring device clearly reacts or the minimum change of the scale indication (for instance, between two scale marks) that can be read without interpolation.

(c) Repeatability designates the ability of a measuring device to produce the same output signal if the same input signal is applied repeatedly.

Where a systematic order is used, the following must be kept in mind at all times: In certain cases, some of the errors named above are so small that they can be disregarded and, consequently, their corresponding correction values will be practically zero. Therefore, in these cases the acquired numerical value, for two or three different correction stages will be equal. Special care is indicated where a correction step (such as correction for indicator time lag) was not performed for any reason, although the associated error has an appreciable magnitude. In that case, the designations for the subsequent intermediate values of the calculation must be clearly distinguished from the normal case (for instance, stating "total temperature, not corrected for indicator time lag error"). Only the most important error categories (position error, temperature lag error, and instrument error) are named in the systematic order shown in this paper. In certain boundary cases, the group of meteorological errors can be added, which is best processed in conjunction with the position errors. Since each error group consists of individual errors, their overall value must be determined from the individual error values in a secondary calculation not included here, before these are entered into the system (cf. below). This is the reason why the term "recovery temperature" does not appear in Figure 103, since it is merely an intermediate value for the calculation of the position error.

Since any computation routine is a time-consuming nuisance during the measurement period, in addition to introducing new sources of error, attempts must be made beforehand to reduce the number of error types and their magnitudes as much as possible. All avoidable errors, such as read-out errors, errors due to recording paper shrinkage, errors due to using instruments with excessive self-heating characteristics, probe location errors, and the like*, must be avoided using appropriate instrumentation techniques. Also, every care must be taken to ensure that the absolute magnitudes of the remaining unavoidable errors are kept as small as possible. The reader is referred to the appropriate discussion in the preceding chapters and to Table 8 in Section 6.1.1.

The errors encountered in accomplishing the different calibrations (according to Section 6.3) can be generally subdivided as follows:

Random errors. These are errors with a statistical distribution of values and indefinite sign (positive or negative).

Example: $E = \pm 0.23^{\circ}C$.

Systematic errors have a certain magnitude and definite sign for a given set of circumstances.

Example: $E = + 0.46^{\circ}C$ (at scale point $+ 25.0^{\circ}C$).

However, because of the limited measuring accuracy, they are frequently encountered as systematic errors with uncertainty, that have a definite sign but a magnitude which is variable within certain limits.

Example: $E = + 0.52^{\circ}C \pm 0.26^{\circ}C$.

Systematic errors can be corrected, since they are determined with respect to sign and magnitude. Several individual systematic errors can be combined through addition into a single magnitude for an error group. Random errors, like the uncertainties of systematic errors, have alternating signs and therefore partially cancel each other, so that they must be combined on a statistical basis. A good approximation for this purpose is provided by the square root of the squared absolute magnitudes of random errors (RSS summation):

$$\pm E = \pm\sqrt{E_a^2 + E_b^2 + E_c^2 + \ldots} \qquad (89)$$

where E_a, E_b, etc are random errors, generally expressed in standard deviations (cf. page 82).

Systematic errors with uncertainty can develop, for instance, as follows: The error E is normally defined as the deviation of the measured value X_m from the true value X_C. Even with careful execution of a series of measurements specifically designed to determine, for instance, the scale error E_{SC}, the readings taken at a certain point on the scale (such as $+25^{\circ}C$) during twenty repetitions of the measurement will yield 20 different measured values ($X_1, X_2, \ldots X_{20}$), which will show a certain amount of dispersion in their values**.

Consequently, we obtain for n measurements of a point n measured values: $X_1, X_2, \ldots X_n$.

The true value is X_C.

The measured error values are:

* e.g. temperature-sensitive DC-amplifiers or oscillators in telemetry equipment.

** This is due, among other causes, to the fact that both the instrument and the test equipment connected to it have a finite accuracy. In addition, there are environmental effects, read-out errors, etc.

$$E_1 = X_1 - X_C$$
$$E_2 = X_2 - X_C$$
$$\vdots$$
$$E_n = X_n - X_C .$$

The <u>arithmetic mean value of the error</u> is:

$$\bar{E} = (E_1 + E_2 + \ldots + E_n)/n \tag{90}$$

The <u>mean error spread</u> α is:

$$\alpha = [(E_1 - \bar{E})_{abs} + (E_2 - \bar{E})_{abs} + \ldots + (E_n - \bar{E})_{abs}]/n \tag{91}$$

Thus we can obtain a systematic error with uncertainty in the following form:

$$E = \bar{E} \pm \alpha \text{ (for example, } E_{SC} = +\,1.97^\circ C \pm 0.045^\circ C).$$

Where it is intended to add this error to another systematic error with uncertainty (for instance, $E_W = -\,0.47^\circ C \pm 0.050^\circ C$), the following procedure is followed:

$$E_{overall} = +\,1.97^\circ C - 0.47^\circ C \pm \sqrt{0.045^2 + 0.050^2}$$
$$= +\,1.50^\circ C \pm 0.067^\circ C$$

Mostly, the standard deviation σ is used instead of the mean error spread α, in which case:

$$E = \bar{E} \pm 1\sigma \,[\text{for example, } E_W = +\,0.90^\circ C \pm 0.017^\circ C \;;\; (1\sigma)].$$

The standard deviation σ is defined as follows:

$$\sigma = \sqrt{\frac{(E_1 - \bar{E})^2 + (E_2 - \bar{E})^2 \ldots . + (E_n - \bar{E})^2}{n}}$$

Generally, measurement errors can be represented by a gaussian distribution. In this case the probability that an error of a particular measurement falls in the interval $\bar{E} \pm 1\sigma$, $\bar{E} \pm 2\sigma$, or $\bar{E} \pm 3\sigma$ is 68.3%, 95.4%, or 99.7%, respectively.

For a gaussian distribution the value of the standard deviation σ can be expressed in terms of the mean error spread α by:

$$\sigma = \alpha\sqrt{\pi/2} \approx 1.25\,\alpha \tag{92}$$

Now we can transform the form of the error E_{SC} mentioned above through multiplication ($\pm 0.045^\circ C \cdot 1.25$) into*:

$$E_{SC} = +\,1.97^\circ C \pm 0.056^\circ C; \quad (1\sigma).$$

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* If needed we can also obtain the form for 3σ by multiplication ($\pm$ 0.056 · 3): $E_{SC} = +\,1.97^\circ C \pm 0.168^\circ C$; ($3\sigma$)

The combination of several individual errors into one overall error is achieved, in the examples used here, through direct addition of the systematic components and forming the square root of the sum of the squared statistical error components (RSS summation of the uncertainties), as follows:

$$E_{overall} = E_{SC} + E_W$$

$$= + 1.97^\circ C + 0.90^\circ C \pm \sqrt{0.056^2 + 0.017^2}\ ^\circ C$$

$$= + 2.87^\circ C \pm 0.0575^\circ C;\ (1\sigma)$$

It is seen that, in the combination of uncertainties or standard deviations, as well as statistically distributed errors (random errors), it is always the maximum value that governs the overall value. In most instances, all values can be disregarded if they are smaller than one fifth of the maximum value. Moreover, it is impractical to calculate with a greatly different number of integers behind the decimal point, since this will not increase the accuracy but merely the computation effort. Therefore, numerical values with too many decimals should be rounded off.

In the calculation routine described above, the overall error for a certain point of the instrument scale (indication + 25°C) was determined. If this routine is repeated for a number N of additional scale points, a different mean for the overall error will be obtained for every point on the scale, where the standard deviation can assume different values. If, as in the example of Figure 101b, the variation of the means is small, an <u>overall error with uncertainty</u> can be derived from the number N of the overall errors that are each valid only for one single point. Here, the sum of the means and the separate standard deviations can be combined as follows:

$$E_{overall} = [(\bar{E}_I + \bar{E}_{II} \ . \ . \ . + \bar{E}_N)/N] \pm \sqrt{(\sigma_I^2 + \sigma_{II}^2 \ . \ . \ . + \sigma_N^2)/N} \qquad (93)$$

Thus we obtain a single value that is valid for the entire range with sufficient accuracy. The same procedure would be possible for an error profile of the type shown in Figure 101a, for low Mach numbers, but it would yield a false statement on the characteristic error profile over a large part of the range.

Practical examples for the acquisition of the overall error as a function of different parameters are provided below. However, it should be noted here that, in addition to the <u>linear error distribution</u> discussed above, where the uncertainty boundaries are located symmetrically to the mean value, there are also <u>nonlinear error distributions</u> where the uncertainty boundaries are located asymmetrically to the mean value. One typical example of this situation is the velocity error. For instance, if the recovery factor fluctuates by ± 0.1 in the subsonic range, the velocity error will have an asymmetrical error distribution. Figure 106* shows a velocity error of $E_V = -2.3^\circ C + 1.45/-1.05^\circ C$ for Mach 0.5. Conversion into a symmetrical form (by forming a simple mean of uncertainty), $E_V = -2.3^\circ C \pm 1.25^\circ C$, is not correct, but in this case, is acceptable in place of cumbersome computation routines, because the error value so obtained is located "on the safe side" (conservative), meaning that it is more likely too high than too low. This form of a systematic error with symmetrical uncertainty can be combined subsequently with other errors of any type, by the procedure described above.

In our discussion of the individual components of the position error in Section 5.2, we mentioned only a few systematic errors that always have a positive sign (such as the self-heating error E_{SH} and the deicing heat error E_{DH}), or always a negative sign (such as the velocity error E_V and the attitude error E_α). Other systematic errors can have different signs, depending on prevailing

* These figures will be discussed at greater length in the section which follows.

conditions (heat addition greater than heat subtraction, and vice versa), for instance, the heat conduction error E_C and the radiation error E_R. When all of these components are combined into the position error, a systematic overall error will arise that has a positive sign below a certain Mach number, passes through zero at this Mach number, and then has a negative sign (cf. Figures 108 and 109)*. The uncertainty or standard deviation of the position error will be very great in the presence of a self-heating error and/or a deicing heat error at extremely low airspeeds, but will decrease very rapidly with increasing airspeed, and rise again gradually above about Mach 0.2. Where these two errors are absent, the uncertainty of standard deviation will be almost equal to zero at very slow airspeeds, and will rise slowly with increasing airspeed, as a function of the velocity error (cf. Figures 106 and 107)*.

In most instances, the temperature lag error can be disregarded, since the measurements are normally performed in a steady-state flight (that is, at constant altitude and airspeed). Where the probe has a simple time constant, and in the event that temperature jumps or temperature changes at constant temperature gradient occur during the measurement, these could be corrected as described in Section 5.2.3. Since it is very difficult, in most cases, to determine whether temperatures belong to one of these two types, or whether they have a sinusoidal pattern, rather elaborate mathematical treatment will usually become necessary (cf. Reference 3). These are subject to great uncertainty, especially in those cases where measurements acquired in accelerated flights are to be evaluated**. Measurements using probes that have several time constants are normally even more difficult to correct. However, in the case of the probe shown in Figure 83, it can be expected that, at airspeeds above Mach 0.2 (that is, where the time constant has values of $\tau_2 < 0.4$ sec), the error will be less than 1% of the temperature change within one second after its occurrence, so that it can be disregarded for practical purposes.

The components of the instrument error E_I (probe calibration error E_{CAL} and scale error E_{SC}) are dependent on the temperature measured in every instance, and no regular interrelationship exists between error magnitude and temperature. These errors are due to random variations in the production process. These must be measured individually for every probe and instrument (cf. Figures 106 and 108)* , unless units with close tolerances and servoed indicators, or signal conditioners with extremely small errors, are used (cf. Figure 107)* . The electric lead error E_W is dependent on the indicated temperature also, but it can be made so small, through use of three-wire or four-wire leads if low-resistance elements are used, that it can be disregarded.

6.4.2 Data Evaluation and Error Correction

Here, data evaluation and error correction are discussed using the example of <u>ambient temperature measurement</u>, and the values illustrated in the form of tables and diagrams.

As a rule we must distinguish between two cases:

(a) If the data are needed over the entire flight envelope of the aircraft, the dependence of all errors on Mach number must be plotted for the different altitudes at increments of typically 10,000 ft. This rather voluminous effort will be required in most instances for flight tests of civil as well as military aircraft.

(b) In the case of aircraft used for civil transportation or similar use, a certain range about a standard flight profile will be exceeded only under exceptional circumstances. Possibly, two such flight profiles can suffice, where an aircraft is employed in both short-range and long-range flights.

In the first case, five diagrams may be required. This could even double in number where both operations with and without probe deicer are to be covered. However, in the second case, the probe deicer system will always be turned on at take-off, so that one or two diagrams will suffice. In the case of VTOL aircraft or helicopters, relationships will possibly range between these two boundary cases.

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* These figures are discussed in more detail in the section which follows.

** The time lag error of the recorded pressure values must first be corrected also.

As a rule, it will be sufficient to plot the appropriate values of the standard atmosphere as static temperatures for the different altitudes. However, it should be pointed out that, near the ground, the static temperature can deviate considerably from the standard value. In those cases where the instrument errors that are dependent on the indicated temperature reach large values, it may prove necessary to prepare two or three diagrams for the lowest altitude level (for instance, 1000 ft)*, since the indicated temperature is a function of at least static temperature and Mach number. For practical purposes (meaning the correction of the indicated values in the aircraft), the usual procedure is to establish a small correction table from the graph presentations, possibly stating the applicable range of the table.

The first type of data evaluation is shown in Figure 106 for a 1200 ohm flush-bulb probe with ratio meter at an altitude of 100 ft and a static temperature of + 16°C. The same presentations would have to be prepared for the altitudes of 10,000 ft, 20,000 ft, and 30,000 ft (depending on the ceiling of the aircraft). In the upper part of Figure 106, the self-heating error E_{SH}, and the airspeed error E_V with its dispersion range, are plotted versus the Mach number. Since the components of the instrument error depend on the temperature of the probe or on the prevailing indication of the instrument, the probe calibration error E_{CAL}, the scale error E_{SC} and their sum, the instrument error E_I (each measured individually in the laboratory) are plotted in the lower part of this figure as a function of temperature. The electric lead error E_W was considered negligible, since the probe has very high resistance and is operated in a three-wire configuration. In order to be able to transfer the instrument error to the upper part of the figure as a function of Mach number, we must first determine the recovery temperatures T_r associated with the individual Mach numbers, assuming a mean recovery factor of $r = 0.85$** for the altitude in question and for the static temperature prevailing there, using Figure 110. The values of the instrument error that are associated with the recovery temperature values found by this procedure can be plotted subsequently in the upper diagram versus the corresponding Mach numbers. Addition of self-heating error, instrument error, and velocity error (mean) will yield the solid line of the overall error, by which the mean of the indication deviates from the total temperature. The uncertainty of the velocity error would have to be displaced in accordance with the overall error of -2.8°C with an uncertainty of about ± 0.8°C*** at Mach 0.4. To be quite precise, specific uncertainties would have to be taken into consideration for the other error values as well, but in the case shown here these are very small and can therefore be disregarded. The uncertainty would increase rapidly with increasing airspeed and would reach a value of about ± 4°C at Mach 0.85, for instance, making the measurement useless.

It is seen from this example that, in the case of simple temperature probes located in the aircraft's boundary layer, the airspeed error, and dispersion, is so large at intermediate Mach numbers that it overshadows all other errors. Further, it is seen that the effort involved in the calibration and in the summation of the instrument errors that are dependent on the indicated temperature, and the position errors that are dependent on the Mach number are extremely laborious and time-consuming if an acceptable accuracy is to be obtained.

The opposite example is illustrated in Figure 107, where the error components for a total temperature probe of 500 ohms and a servoed indicator are shown. In this system, the replacement of the probe or the instrument requires only the adjustment of the line compensation for the instrument at two measuring points. In that case, the instrument error, like its individual errors, is defined by its tolerance limits:

$$\pm \Delta E_I = \pm\sqrt{\Delta E_{CAL}{}^2 + \Delta E_{SC}{}^2} = \pm\sqrt{0.1^2 + 0.2^2} = \pm\ 0.23^\circ C \qquad (94)$$

(cf. the upper part of the figure). This value is not dependent on the indicated temperature, since the

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* This corresponds approximately to minimum altitude, except for take-off and landing.

** To simplify the presentation, a very favorable probe location was assumed; this may result in an almost constant recovery factor.

*** In this case, dispersion range should be asymmetrical: +0.9°C, -0.7°C; it can be rounded off with little error to ± 0.8°C, cf. above.

limits for E_{SC} and E_{CAL} apply for the entire temperature range. Where probes with normal tolerances are used (cf. the dotted curves), this would not be the case. However, during normal climbing and let-down flights the temperature in the subsonic airspeed range would be displaced to such a degree that the probe calibration error should not exceed the limits of $\pm 0.3^{\circ}C$. The result would be an instrument error of $\pm 0.36^{\circ}C$ over the entire subsonic range that would include the scale error limits. The center diagram shows the deicing heat error E_{DH} of a probe housing with one element*, and the velocity error E_V with uncertainties up to an altitude of about 1000 ft. The lower diagram shows the corresponding values for the stratosphere. This type of a TAT probe will also yield an asymmetrical measured value distribution, where the mean will be located nearer to the smaller values. Consequently, if the probe deicer were turned off, only the velocity error E_V would have an effect, and its uncertainty would have to be summed (after conversion to symmetrical values) with the uncertainty of the instrument error (RSS summation). If the probe deicer were turned on, the velocity error E_V would be compensated by the deicing heat error E_{DH} at about Mach 0.35 in sea level flight, and Mach 0.55 at an altitude of 40,000 ft. At greater airspeeds, the velocity error would be predominant, and at slower airspeeds the deicing heat error predominates. The resulting overall error** was not plotted in this figure, but the self-heating error E_{SH} when using galvanometers and ratio meters is shown for comparison purposes. This error is hardly noticeable in the case of good servoed instruments.

Figure 108 shows an example for case (b) identified above, where the resulting overall error is shown for a flight profile with Mach 0.2 at sea level, Mach 0.85 at 40,000 ft, and Mach 2.0 at 50,000 ft, using servoed instruments with enclosed and exposed elements, each set with and without probe deicer system.

Figure 109 shows a combination of a probe having an enclosed 500 ohm platinum element with a galvanometer and with a ratio meter, for the same flight profile. The means and the associated uncertainties are represented for the conditions with and without probe deicer. In this case the tolerance limits of the instrument error and probe calibration error were entered into the calculation of the uncertainty, since the stipulation had been made that the probes and the indicators should be interchangeable without individual recalibration. It is seen from this figure that this requirement, which is made by some aircraft operators, results in uncertainties or tolerance limits ranging from $\pm 1.5^{\circ}C$ to $2.5^{\circ}C$, in spite of using TAT probes. Under these conditions the employment of probes having close tolerances would hardly yield an advantage, since the instrument tolerances are the governing factor. However, where simple probes are used in the aircraft's boundary layer ***, the uncertainties would increase very rapidly at airspeeds in excess of about Mach 0.4. The case of a combination of a TAT probe (with normal calibration tolerances) with a galvanometer is shown for comparison purposes, both elements being calibrated in the laboratory. This results in a considerably narrowed uncertainty, cf. the shaded zone, of about $\pm 0.5^{\circ}C$. However, the mean can have such an unfavorable profile versus the Mach number that a strange correction table would result.

The following conclusions can be drawn from these figures:

(a) The customary combination of TAT probe with enclosed element (normal tolerances) and galvanometer or ratio meter will permit only <u>moderate accuracy</u> with a dispersion of $\pm 2.5^{\circ}C$ in the case of the galvanometer or $\pm 1.5^{\circ}C$ for the ratio meter (applicable for measuring ranges from $-50^{\circ}C$ to $+70^{\circ}C$), unless correction tables are used.

(b) If an instrument error correction (composed from individual calibrations of instrument and probe), which is dependent on the indicated temperature, is allowed for the equipment array named above, supplemented by an additional correction for the position error (velocity error + deicing heat error), which is dependent on the Mach number, <u>quite reasonable accuracy</u> will be achieved with an uncertainty of about $\pm 0.5^{\circ}C$. However, the correction tables must list a large number of values and become too cumbersome to handle in practical application.

* Deicing heat error is considerably greater in housings with two elements (by more than $1^{\circ}C$ at slow airspeeds).

** Cf. Figure 108.

*** As is the case in Figure 106.

(c) The <u>best accuracy</u> (0 to $\pm$ 0.25°C with an uncertainty of $\pm$ 0.25°C) is provided by a probe with an open-wire element and close tolerances, combined with a servoed indicator. In that case the correction tables may become entirely superfluous. However, because of their fragile nature, these open-wire elements are hardly feasible for sustained use in supersonic aircraft.

(d) Very good results (0 to -0.6°C $\pm$ 0.32°C) are achieved with probes having sealed elements and close tolerances, combined with a servoed indicator. In this case, the correction tables could be eliminated for many types of applications, and where it is needed, the table will be much simpler than it is in the case of item (b) above.

An additional inaccuracy of $\pm$ 0.5°C will develop during the <u>conversion</u> of total temperature to <u>static temperature</u> (using tables or Figure 110, line $r = 1.00$), increasing the uncertainty to a maximum of $\pm$2.6°C or $\pm$ 1.6°C, respectively, in case (a), and to about $\pm$ 0.6°C in case (d).

In an <u>air data computer</u>, the corrections for the systematic error component of the output signal (that is fed to other systems and indicators) should be done automatically. In that case, the theoretical uncertainty for the static temperature in a computer for the subsonic airspeed range (at a mean error of $\pm$ 0.01 Mach in the acquisition of the Mach number) would have a maximum of $\pm$ 0.8°C*. However, the "guaranteed" values are $\pm$ 3.0°C from Mach 0.3 upward for the older systems (referred to the indication on the instrument), because of the difficulties involved in acquiring the Mach number at the low airspeeds. However, in this respect, the following should be noted:

The "guaranteed" values of air data computers apply almost always with the stipulation that the position error at the inputs for static and total pressure are present only above Mach 0.4, and that the position error table acquired during the flight test program is valid for all aircraft of the same type and permits 100% correction. Also, it is usually stipulated that the velocity error be the only appreciable error of the temperature probe (which in the case of TAT probes has been reduced to the magnitude of the other errors). Moreover, in the majority of cases the computer manufacturers prescribe neither the selection of the TAT probe (for instance, types with close tolerances and two elements), nor that the deicer be turned on (while taxiing and in flight). And in any event, the temperature probes are in most cases operated at greatly excessive voltages, so that self-heating will assume considerable values. Consequently, it is worthy of examination whether the "guaranteed temperature values" are maintained only with a substitute resistance (that can be loaded as desired) or whether they can be maintained with a probe exposed to an air flow, taking into consideration whether the probe deicer is turned on or off.

Consequently, there are the following uncertainties as absolute and relative values for the determination of the static air temperature, the relative values referred to a static air temperature of 216.7°K = - 56.5°C in the stratosphere* (for airspeeds up to about Mach 2.3 and altitudes below about 60,000 ft):

Conversion from the TAT indication	Galvanometer	$\pm$2.6°C = $\pm$1.2%
	Ratio-meter	$\pm$1.6°C = $\pm$0.74%
	Servoed instrument (with PCI)	$\pm$0.6°C = $\pm$0.28%
Air data computer	New (manufacturer's data)	$\pm$1°C = $\pm$0.46%
	Old (manufacturer's data)	$\pm$3°C = $\pm$1.4%
Radiosonde	New (calibrated prior to take-off)	$\pm$0.7°C = $\pm$0.33%
	In general, maximum	$\pm$5°C = $\pm$2.3%

When the <u>compressor inlet temperature</u> CIT is measured, the data are evaluated by the same procedure, except that they may be plotted versus the flowrate (mass velocity) instead of the Mach number. In the case of the OAT measurement, it was sufficient to plot the airspeed as a function of

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* Provided that Mach numbers of greater value are flown only at greater altitudes.

** Values are slightly more favorable at lower altitudes and airspeeds, if no flights in clouds, dense fog, or intense rain are made.

Mach number and altitude, since low Mach numbers at high altitude (such as Mach 0.2 at 40,000 ft) is a situation outside the flight regime of normal aircraft. Since it is quite possible that low air velocities at relatively low pressures are encountered in engine inlets at throttled engine output, the values shown in the diagrams for the recovery error η will no longer be valid in such cases. An additional recovery correction factor must be applied that is a function of probe design and can assume values between about 0.4 and 4.7. Probe designs of the type shown in Figure 20b have an appreciably greater recovery error compared to the TAT probes used for OAT measurements, but have a recovery correction factor instead, that is close to 1.0. In that case, the correction values for the recovery error that must be applied will be appropriately greater, but the additional correction for the correction factor will be eliminated. These problems are discussed in detail in Reference 60.

Similar considerations should be valid for measurements of the engine exhaust gas temperature EGT. Here we are concerned with very high temperatures, where we will always encounter relatively high radiation and heat conduction errors whenever there are major temperature differences between the gas and the wall of the jet nozzle (which occurs especially after a change in engine output). Under these conditions, it would be very difficult to isolate the radiation and heat conduction error from the velocity error or its change as a result of a change in the recovery error. Also, it should be pointed out again that the general practice in computing the total temperature is to use a value of $\gamma = 1.33$ in place of the γ value of 1.4 that is customary for OAT measurements. This is because of the high static temperature of the gas. Publications available to date on EGT measurements (References 41 and 48) do not yet cover all the conditions that are possible for the OAT measurement.

In the case of temperature measurements in liquids, in solid bodies, and on surfaces, the only error in addition to the instrument errors is usually the heat conduction error E_C, which is the only component of the position error of concern here. It is dependent on a great variety of parameters, such as flying time, engine operation temperature, altitude and airspeed, and the like. Very frequently the data evaluation is limited to the addition of probe calibration error E_{CAL} and scale error E_{SC} as functions of the indicated temperature. Any self-heating effect that may have to be taken into consideration, for instance, where the temperature of solid bodies is measured with tip-sensitive probes, can be included implicitly through an appropriate offset of the values of the scale error. Position errors that must be processed, as in an OAT measurement, will be encountered only in surface temperature measurements on the aircraft skin or in the air inlets or exhaust gas tubes. In that case, special attention must be devoted to the radiation error E_R unless the temperature probes have the same emissivity as the surface concerned (Reference 111).

CONCLUSION

The first temperature measurements in aircraft were probably made more than 50 years ago. Perhaps this period should have been sufficient to clarify all the open questions in spite of the advances that have been made in the performance of aircraft. That this is not at all true, and that the number of unsolved problems has increased rather than decreased, shall be discussed briefly in this section.

It was seen from the preceding sections that considerable progress has been achieved only in the last decades, especially with respect to ambient temperature measurements, but this progress has certainly not been fully exploited. The development trend in modern avionic systems shows that the importance of air data, and hence of temperature measurements, will continue to increase in the future, since a continuously rising number of black boxes demand more and more inputs of high accuracy that must be derived from air data. With the aid of appropriately designed modern air data computers, it should be possible to limit the error in temperature values to about three to five parts per thousand, provided that the boundaries that correspond to the flight regime of present-day conventional aircraft are not exceeded:

Sea level: approximately Mach 0 to 1.2;
40,000 ft: approximately Mach 0.7 to 1.7;
60,000 ft: approximately Mach 1.5 to 2.3.

This range is also characteristic for the validity limits of the simplified equations using $\gamma = 1.40$ and of the diagrams listed in this book. That is, provided the air has a limited content of water vapor and liquid water.

However, the list of flight regimes mentioned above must be supplemented by at least one additional range:

80,000 ft: approximately Mach 2.0 to 3.0.

At this altitude the density of the air is very low so that a sharp rise of the otherwise negligible heat conduction error must be expected if the air speed is considerably below the minimum named above. On the other hand, very high total temperatures will develop in this airspeed range (according to Figure 61, up to a maximum of about +460°C). Under these conditions, the radiation error will reach appreciable magnitudes, since it depends on the fourth power of the gas temperatures inside the probe housing and of the housing wall. Figure 71 shows that it can amount to between 0.08% and 0.12% of absolute total temperature, corresponding to an absolute error between -0.35°C and -0.88°C. Among other effects, both of these errors are influenced by the fact that, at high altitudes, the boundary layer about the individual probe elements (sensor, radiation shield, inner wall of housing) become very thick, depending on the Reynolds number, and start penetrating each other so that measured data acquired at low altitudes cannot be simply extrapolated to higher altitudes (References 103 and 112). In the case of modern probes, this condition will not be encountered, especially at higher air speeds, as long as the total pressure behind the shock wave does not drop below a value of about 300 mm Hg (about 400 mb).* Also, we should keep in mind that the given equations and diagrams for supersonic flight apply only behind normal shock waves but not behind oblique or intersecting shock waves. However, with increasing Mach number and altitude the shock wave formed at the nose will become continuously sharper so that interaction between the nose shock and the shock wave formed at the probe inlet can take place. Conditions become even more complicated when the boundary layer at the fuselage, which becomes thicker with increasing airspeed as a result of the rising temperature, immerses the probe inlet. In addition, it must be kept in mind here that, at altitudes of 80,000 ft and at high total temperatures, we approach the zone where the air no longer behaves as a continuous fluid. In this zone significant changes in the aerodynamic and heat transfer processes take place. In that case the theories on rarefied gas dynamics must be used. When the ratio of mean free molecular path length to boundary layer thickness, known as the Knudsen number K, reaches a value of about 0.01, the continuous flow will be replaced by a so-called slip flow. The layer of air which contacts the surface of an aerodynamic body directly will no longer stick to it but will slide along the surface, and a corresponding temperature jump will take place (Reference 106). The practical implications of this phenomenon are not by any means completely understood from the viewpoint of instrumentation technology.

Moreover, it was already indicated in Section 5.2 that the value $\gamma = c_p/c_v = 1.40$ which is normally used to compute the kinetic temperature rise is no longer valid at very high temperatures. The change in the γ values is due to the fact that with increasing static temperature more and more degrees of freedom of diatomic molecules are excited (cf. References 107, etc.). Thus, the value of γ drops from 1.67 at very low temperatures to about 1.40 at room temperature. At about 350°K, vibrational degrees of freedom are excited in addition to the rotational and translational degrees of freedom. These achieve their full value at about 1.800°K, where γ drops down to 1.29. In addition, when the sensor is located immediately behind the shock wave, it must be remembered that γ is not only a function of temperature and pressure but also of the rate of expansion or compression, since the air molecules require a certain period of time to achieve temperature equilibrium. Under these conditions, the effective values of γ decrease at a much slower rate with respect to the total temperature than the γ values for the static temperatures (cf. Figure 62). This change of the γ value from its standard value of 1.40, also called caloric imperfection, is not a source of error in measuring the total temperature but, at total temperatures above about +150°C, that is, at air speeds

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* At high supersonic speeds exceeding $M = 3$, corrections of the TAT indications independent of the total pressure must be found. It must also be kept in mind that at high temperatures all insulators convert slowly to conductors and that at about $M = 6.7$ the TAT approaches the melting temp. of platinum-rhodium thermocouples (cf. Reference 112).

in excess of about Mach 2.1, the corresponding γ_{eff} value (instead of the standard value of 1.40) must be used when the static temperature is computed from the total temperature (e.g. in air data computers for SST).

Caloric imperfections can play a role even in the subsonic air speed range. Let us be reminded that, for instance, in flights near the ground at transonic air speeds and a static temperature of +30°C accompanied by a relative humidity of more than 80%, errors of more than 1% must be expected in the values supplied by an air data computer (true air speed, static air temperature, and the like). These increase very rapidly with increasing temperature and humidity (Reference 105). Another area where the practical implications are not understood are the consequences that result when water vapor condenses in expansion zones about the aircraft or in the engine inlet*, a phenomenon that is not unusual. Similar conditions prevail when the aircraft enters a cloud formation. Here a different temperature and air density prevails than in the adjacent clear air; however, it is certain that the values read from the altimeter, air speed indicator, and temperature indicator are subject to very large errors. This applies especially to the temperature values when the cloud contains undercooled water droplets that have subfreezing temperatures. All of these effects, in addition to the effects of an intensive rainfall, are not satisfactorily understood at all (References 39, 49, 97, 98, and 105 to 110). These effects deserve special attention not only for poor weather approaches by civilian aircraft, but even more so for military aircraft that must operate at high air speeds and low altitude in order to escape radar acquisition.

In the case of an OAT measurement, it is relatively simple to determine the absolute value of the velocity error as a function of recovery error and Mach number. In the case of the CIT and EGT measurements, the velocity error is more difficult to determine since the flow rate (mass velocity) in the engine is dependent, among other things, on the flight Mach number and engine rpm or engine output. On the other hand, little experimental confirmation exists for the mathematical relationships between local Mach number in the engine and mass velocity as a function of pressure and temperature (cf. Figure 62c). Detailed investigations are still needed on the γ_{eff} values to be used in EGT measurements (static value about 1.316 near 800°C, normally used "effective value" = 1.33). Since this is an area where relatively primitive probe types are used, and the service life of the engine is a logarithmic function of EGT, the remaining open questions should be clarified experimentally, considering the continuous increases in engine performance.

As a rule, the remaining types of measurements are relatively straight forward and do not require detailed discussion here. One exception could be the measurement of the temperature of a surface located in an inflow. Here the problems outlined above are coupled with the uncertainties involved in measurements obtained in a boundary layer, and special attention will have to be devoted to the radiation error.

In compiling this volume the authors encountered a highly variable nomenclature in the different publications on the subjects discussed here. Therefore, an attempt was made to achieve a certain uniformity of presentation, at the same time retaining certain terms and symbols that have become well established in practical use. Especially great divergence in the terms and symbols used exists between the purely scientific literature (for instance on wind tunnel investigations) and the flight testing results published by the practitioners. But even among the latter, it is rarely possible to evaluate and compare the results directly, since important information is lacking in most instances (such as, whether data were obtained in steady-state or accelerated flight, whether or not corrected for instrument error or self-heating error, resistance value of the element, instrument type and harness type used, statement of probe loading, and the like). It would be of great advantage if the symbols and terms used in science and practical aeronautics could be standardized.

* This applies just as well for condensation shocks behind shock waves in supersonic flight (Reference 107).

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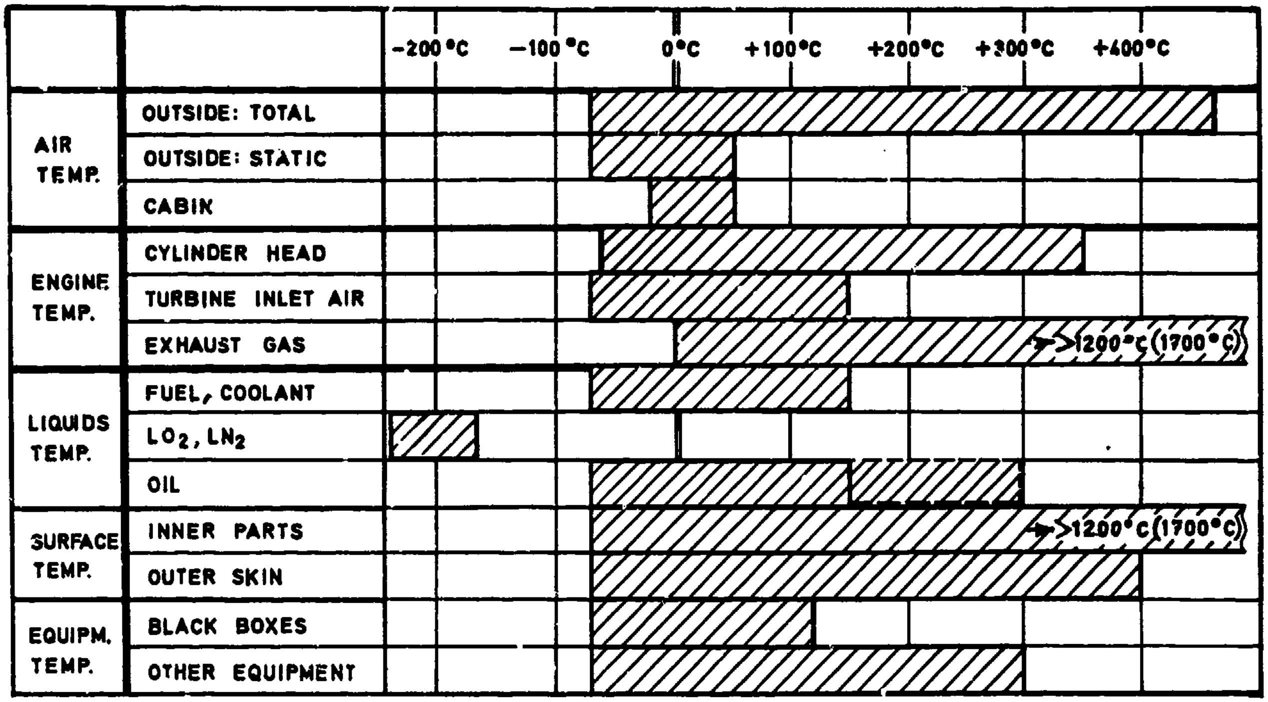


Fig.1 Ranges of temperature measurements in aircraft

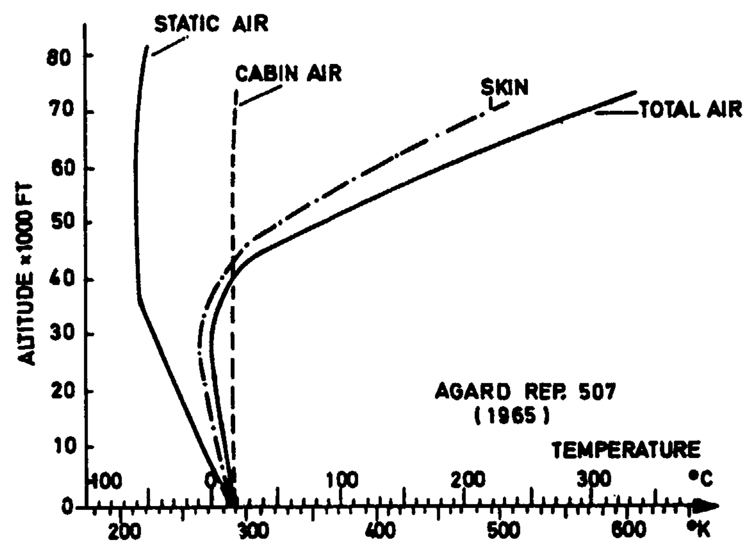


Fig.2 Environmental temperatures of supersonic aircraft

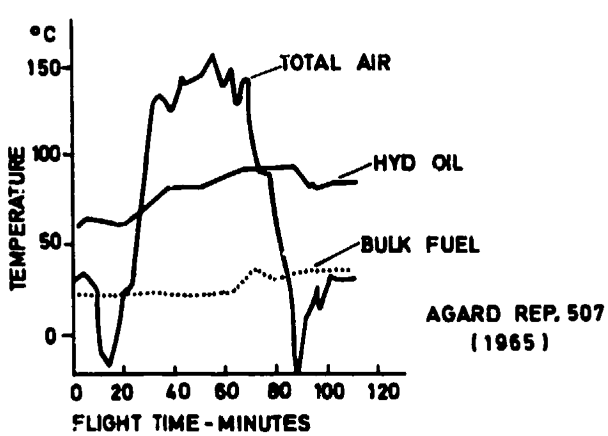


Fig.3 Temperatures vs time for supersonic flight

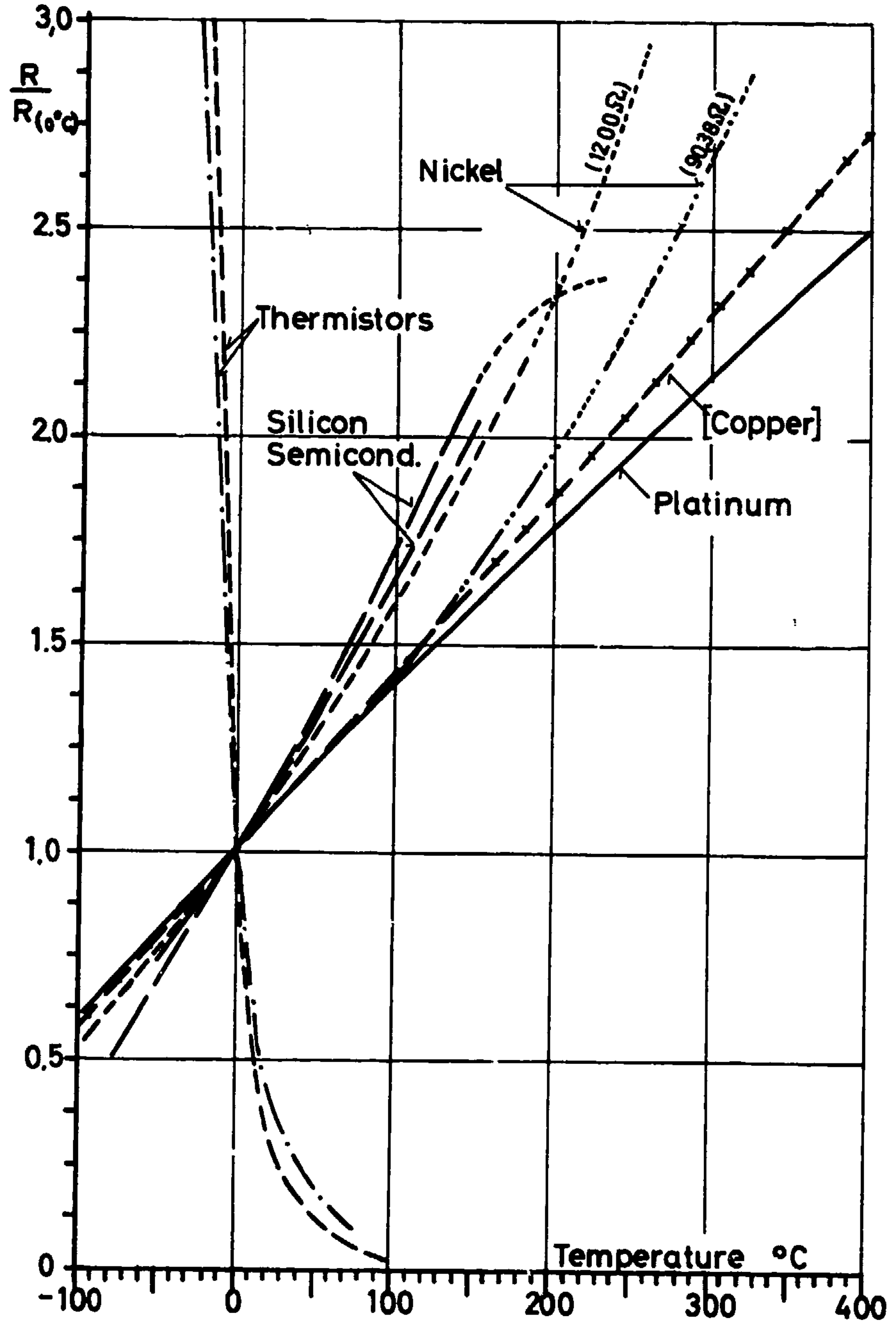


Fig.4 Resistance-temperature relationship of some temperature elements

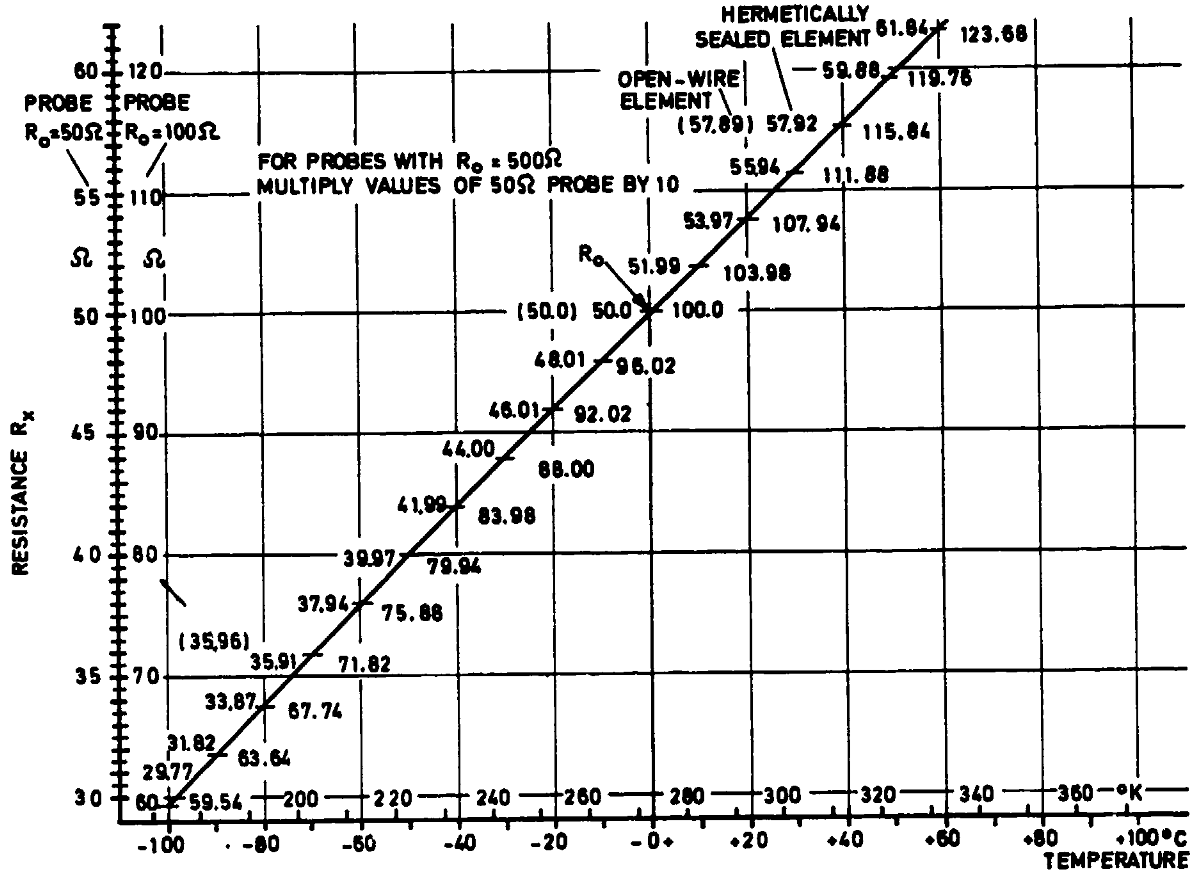


Fig.5(a) Resistance-temperature relationship of platinum elements MIL–P–28726 range –100 to +60°C

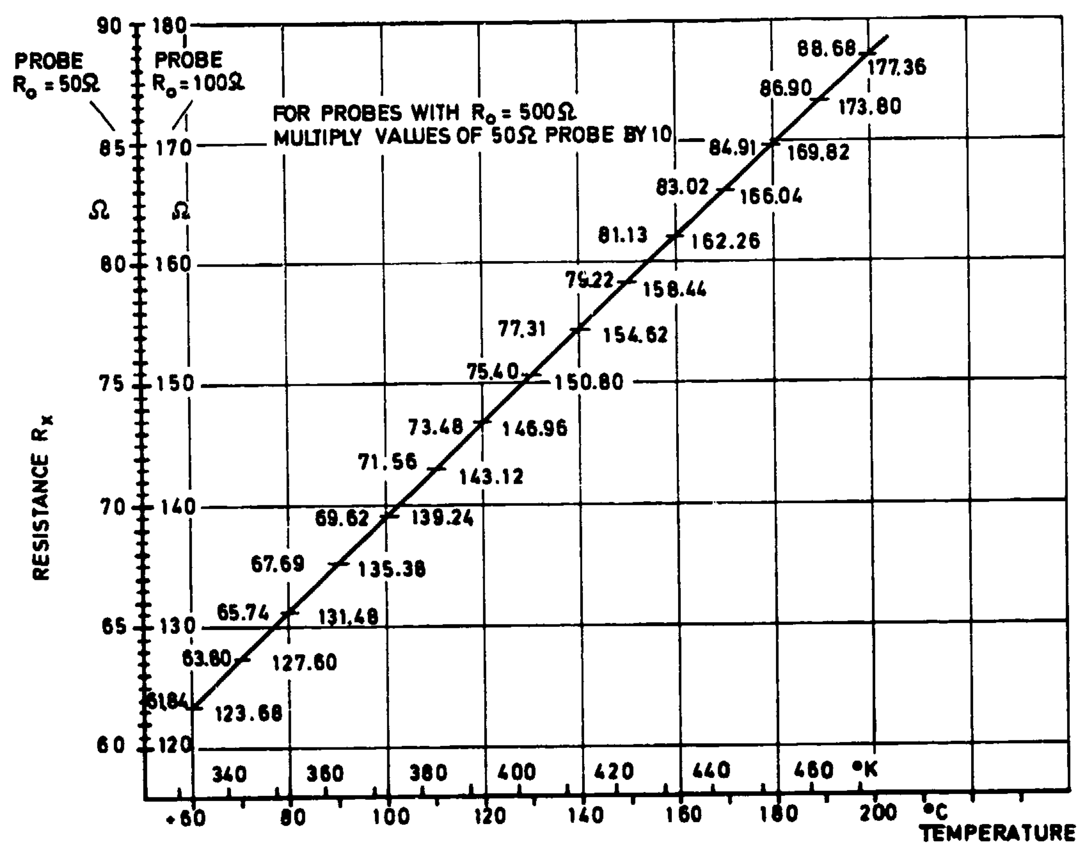


Fig.5(b) Cont., range +60 to +200°C (platinum, MIL–P–25726).

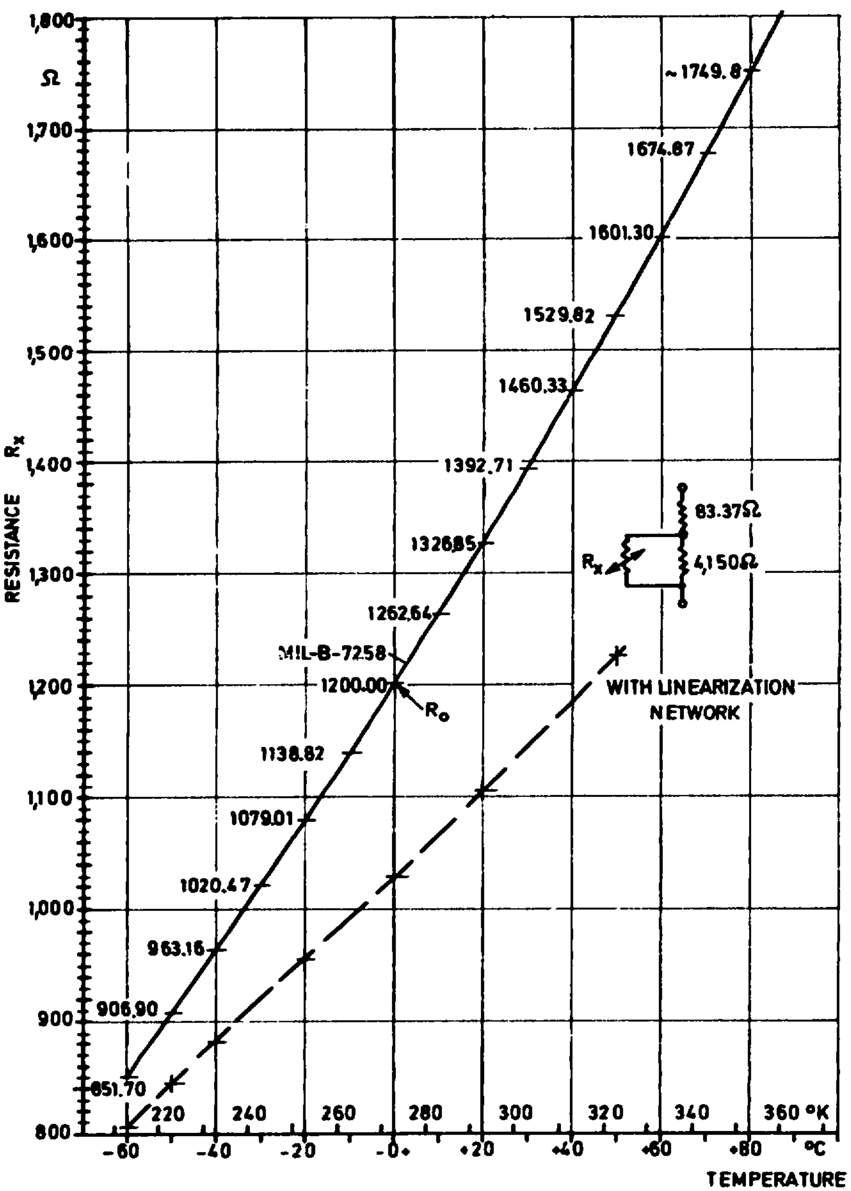


Fig.6(a) Resistance-temperature relationship of nickel element MIL–B–7258

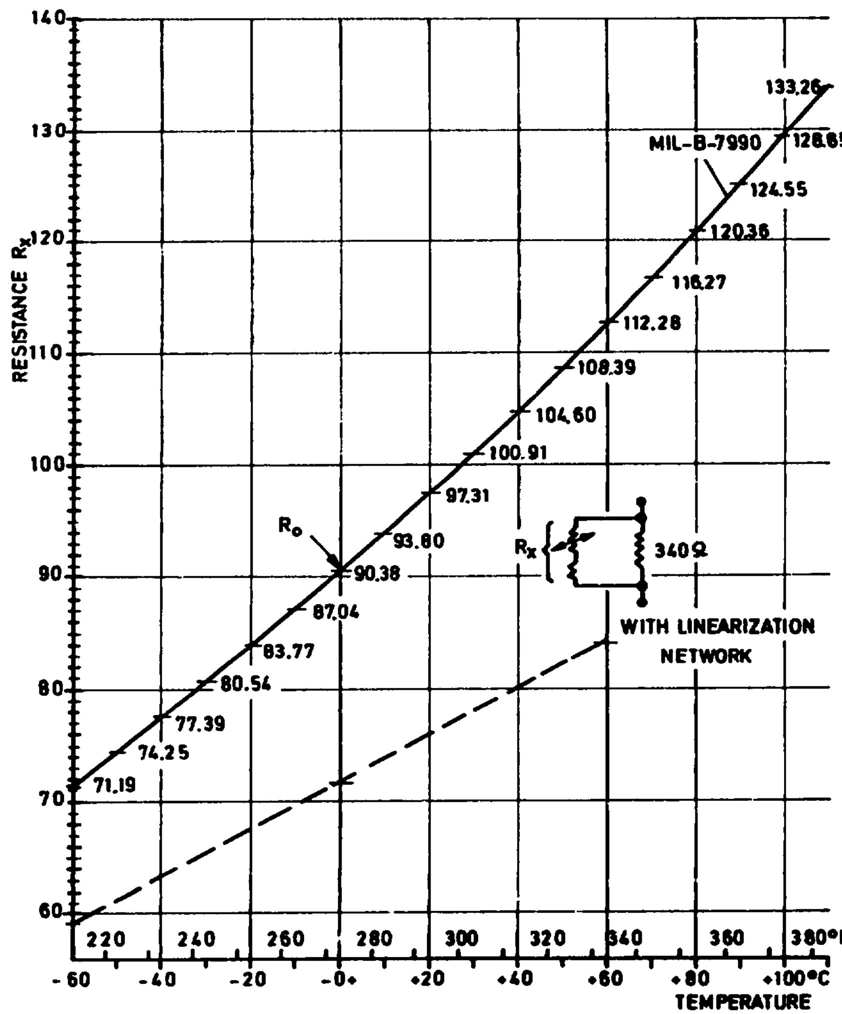


Fig.6(b) Resistance-temperature relationship of nickel element MIL–B–7990

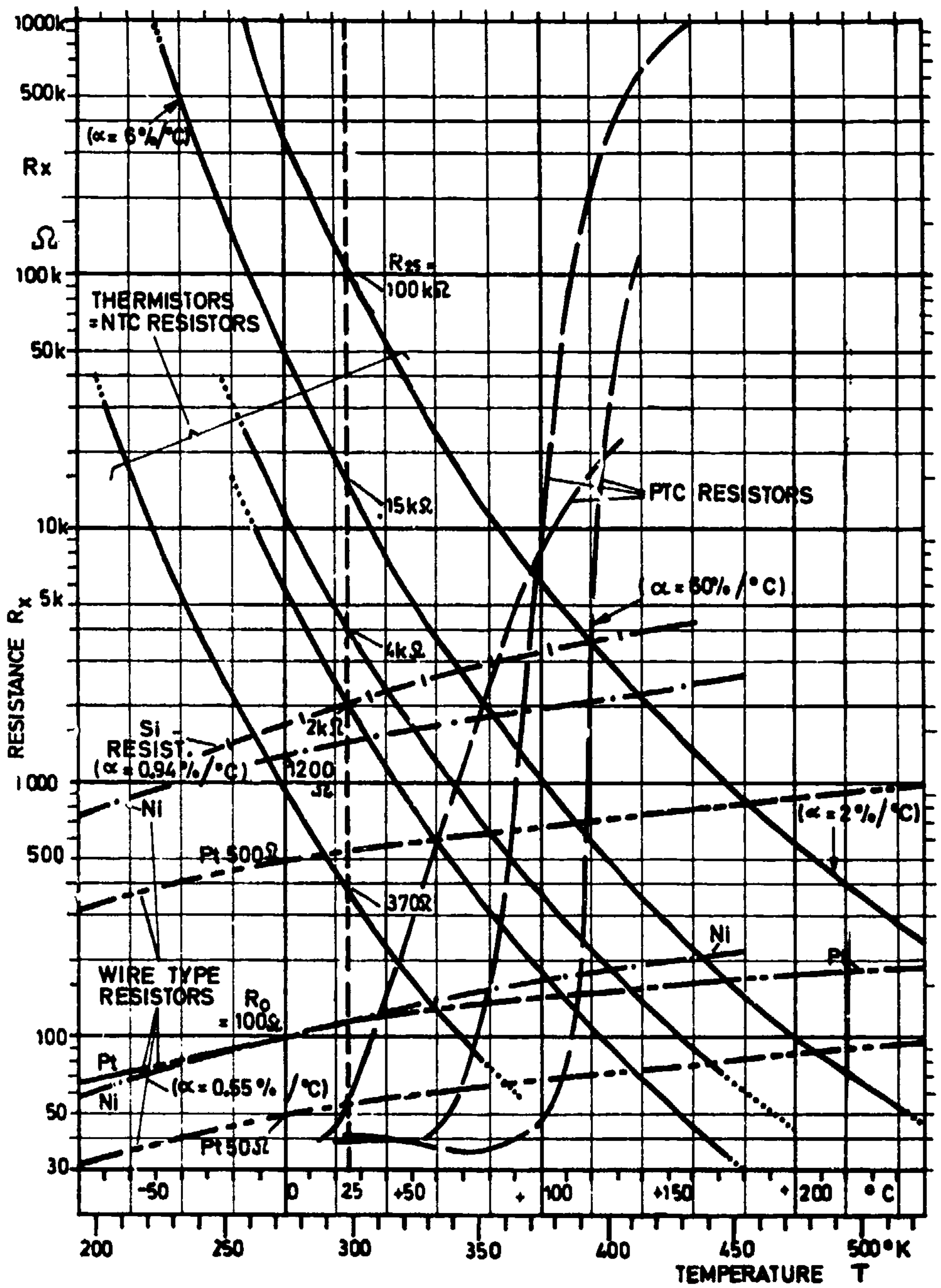


Fig.7 Characteristics of resistance elements, temperature vs resistance

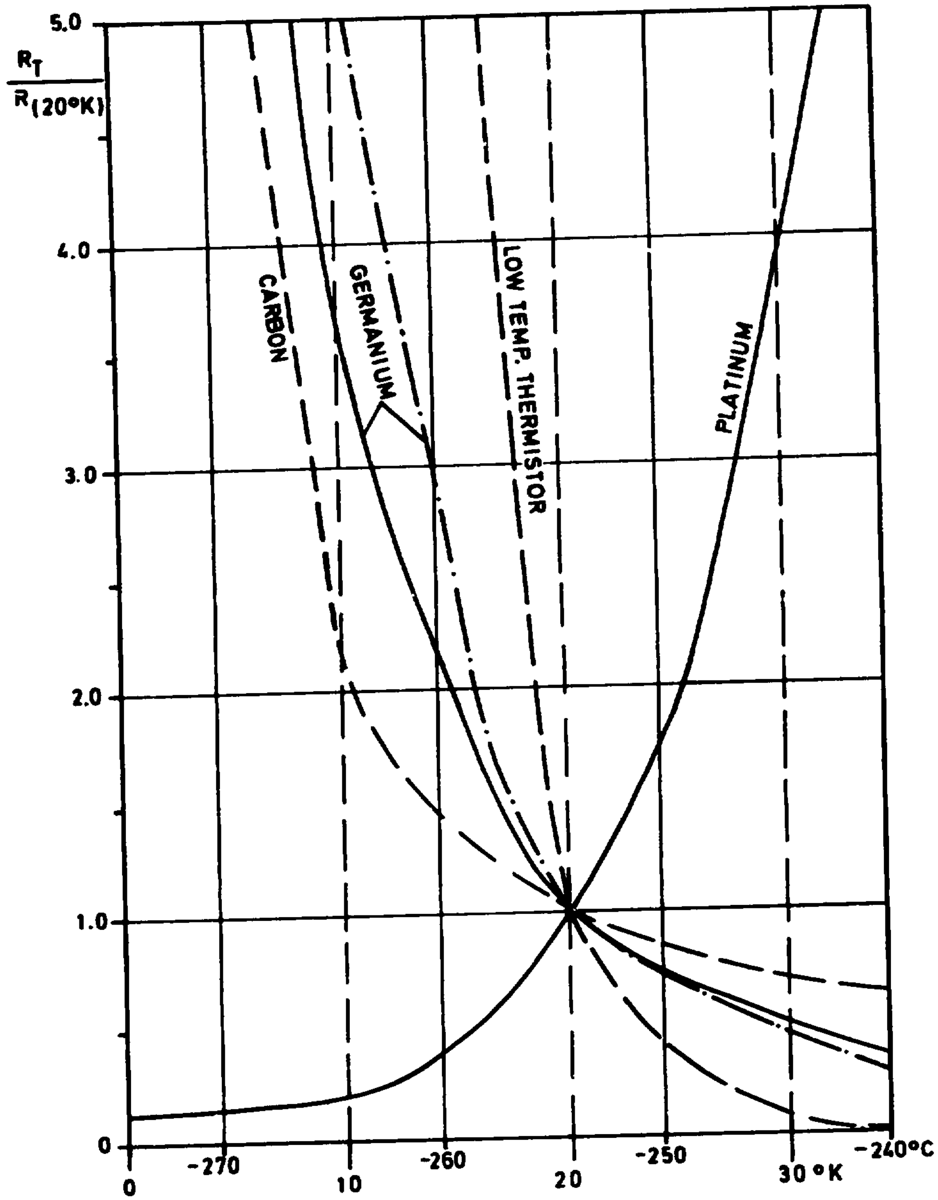


Fig.8(a) Resistance-temperature relationship of low temperature sensors.

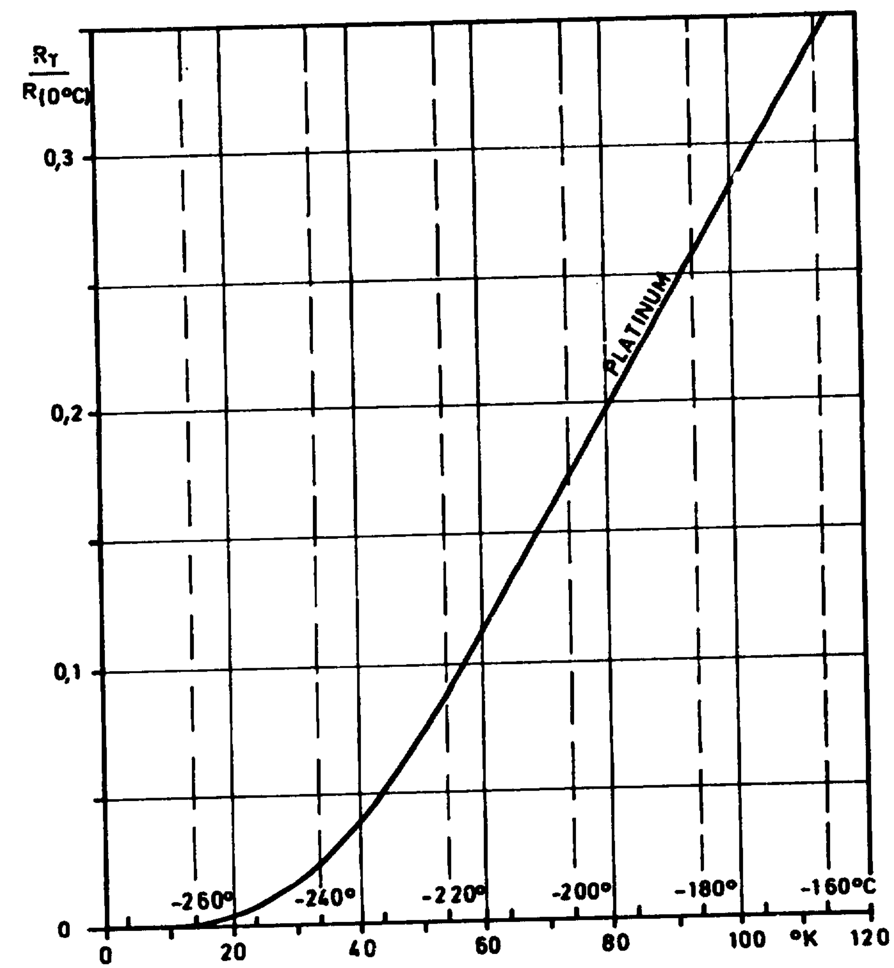
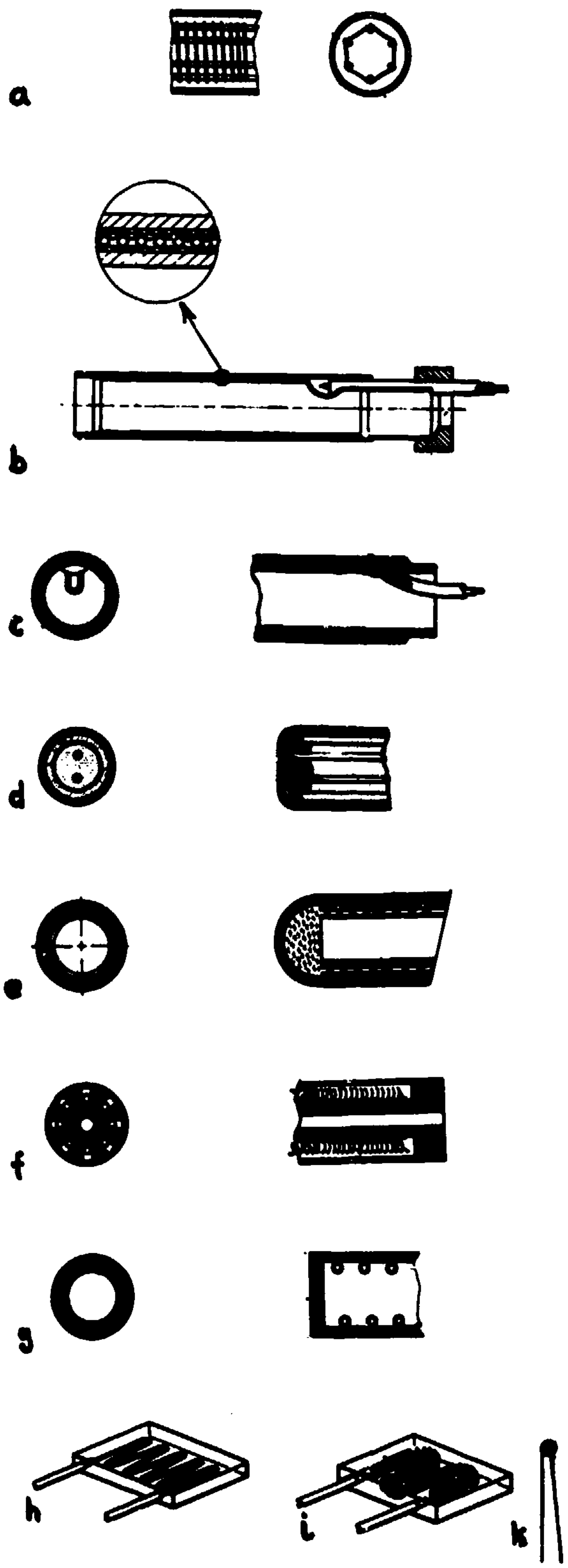


Fig.8(b) Resistance-temperature relationship of low temperature sensors



(a) open-wire element

(b) hermetically sealed tubular element

(c) ceramic coated tubular element

(d) miniature ceramic coated element

(e) well-type element

(f) rugged well-type element

(g) fast response well-type element

(h) flat element (strain-sensitive)

(i) flat element (strain-free)

(k) thermistor

Fig.9 Construction of resistance probes

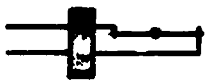


Fig. 10 Thermistor
(special type)

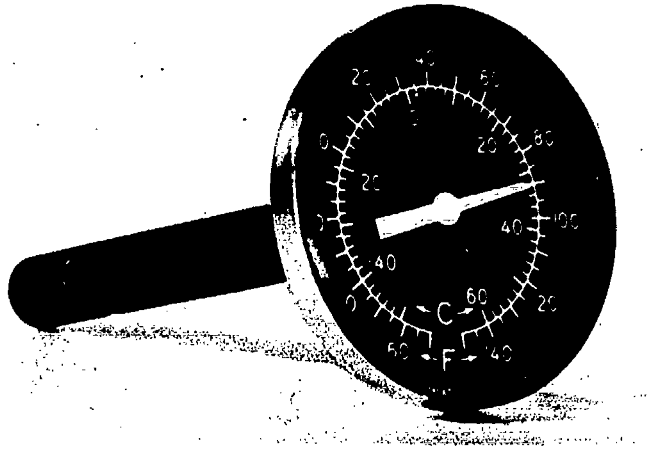
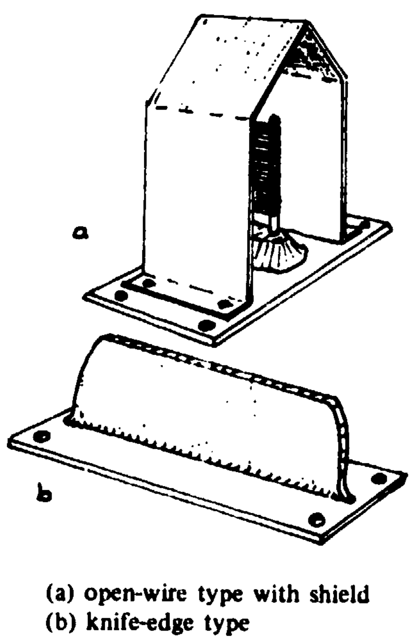


Fig. 11 Bimetallic thermometer with radiation shield

(a,b) for use in cross-flow
(c) L-shaped probe for parallel-flow
(d) ditto, with protecting shield

Fig. 12 Simple well-type probes, stem-sensitive



(a) open-wire type with shield
(b) knife-edge type

Fig. 13 Old types of probes

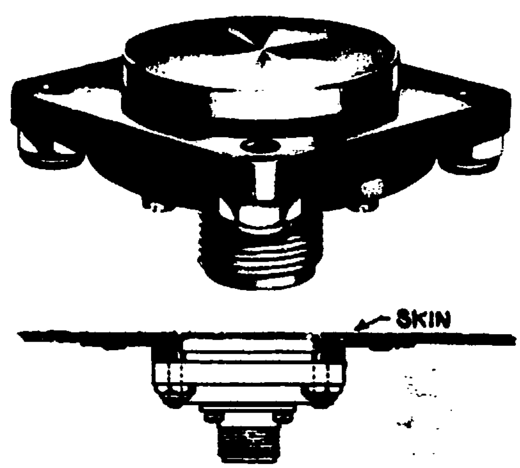


Fig. 14 Flush-bulb

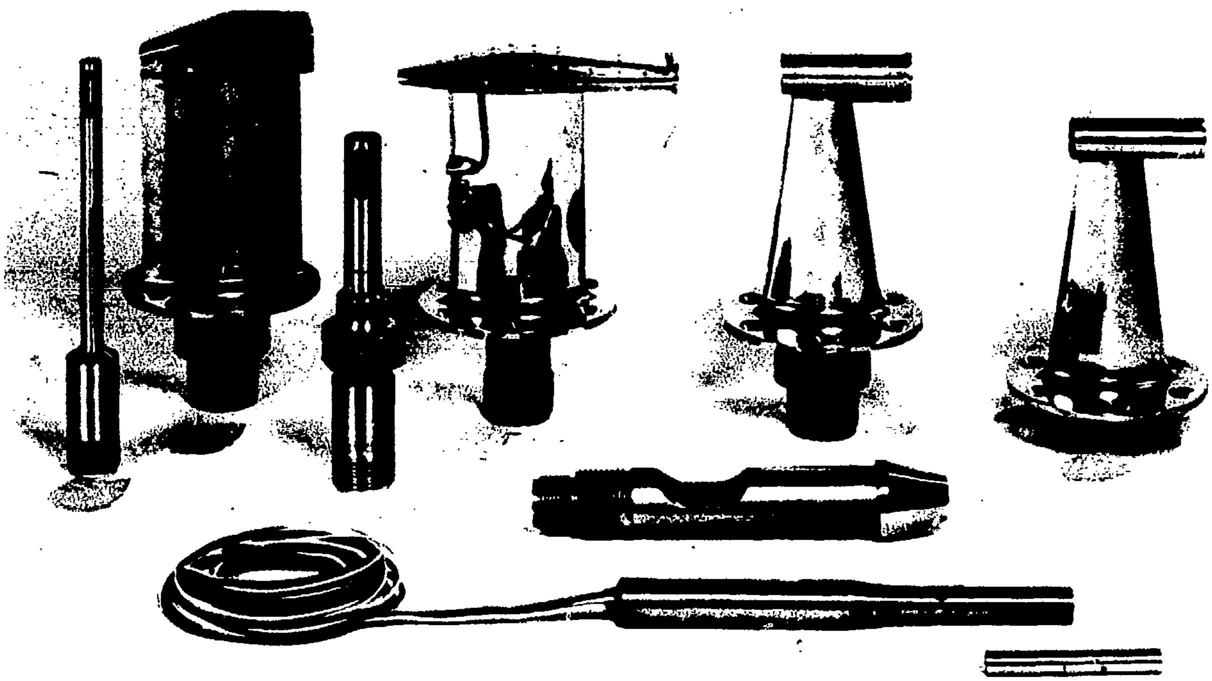


Fig.15 Total temperature probes

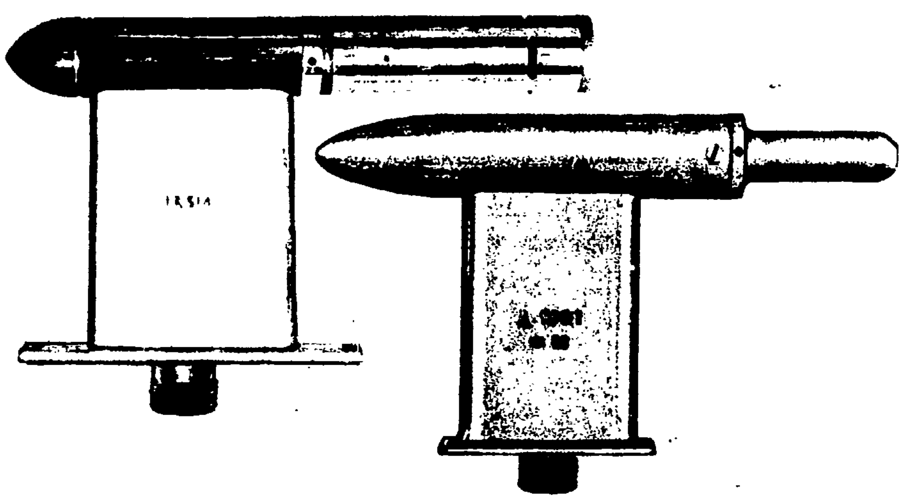
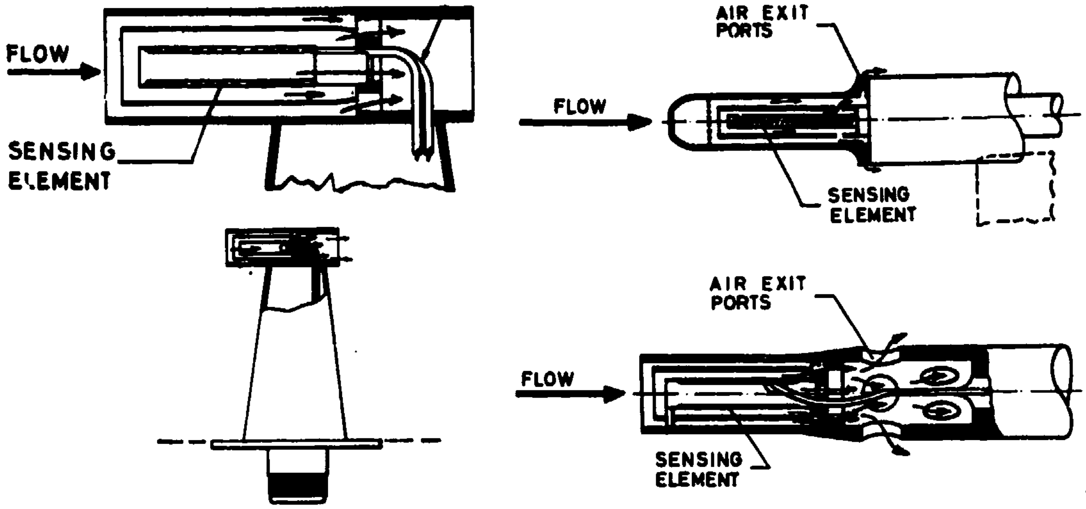


Fig.16 Total temperature probes



left: type without air flow deflection
right: types with air flow deflection aft of element

Fig.17 Internal construction of TAT probes

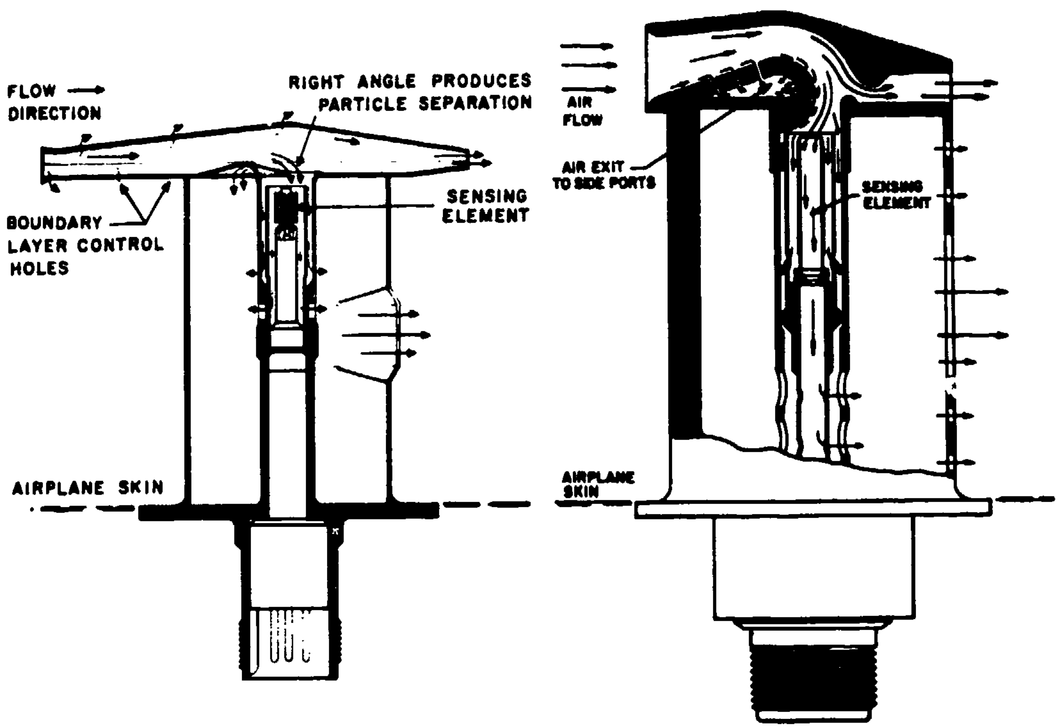
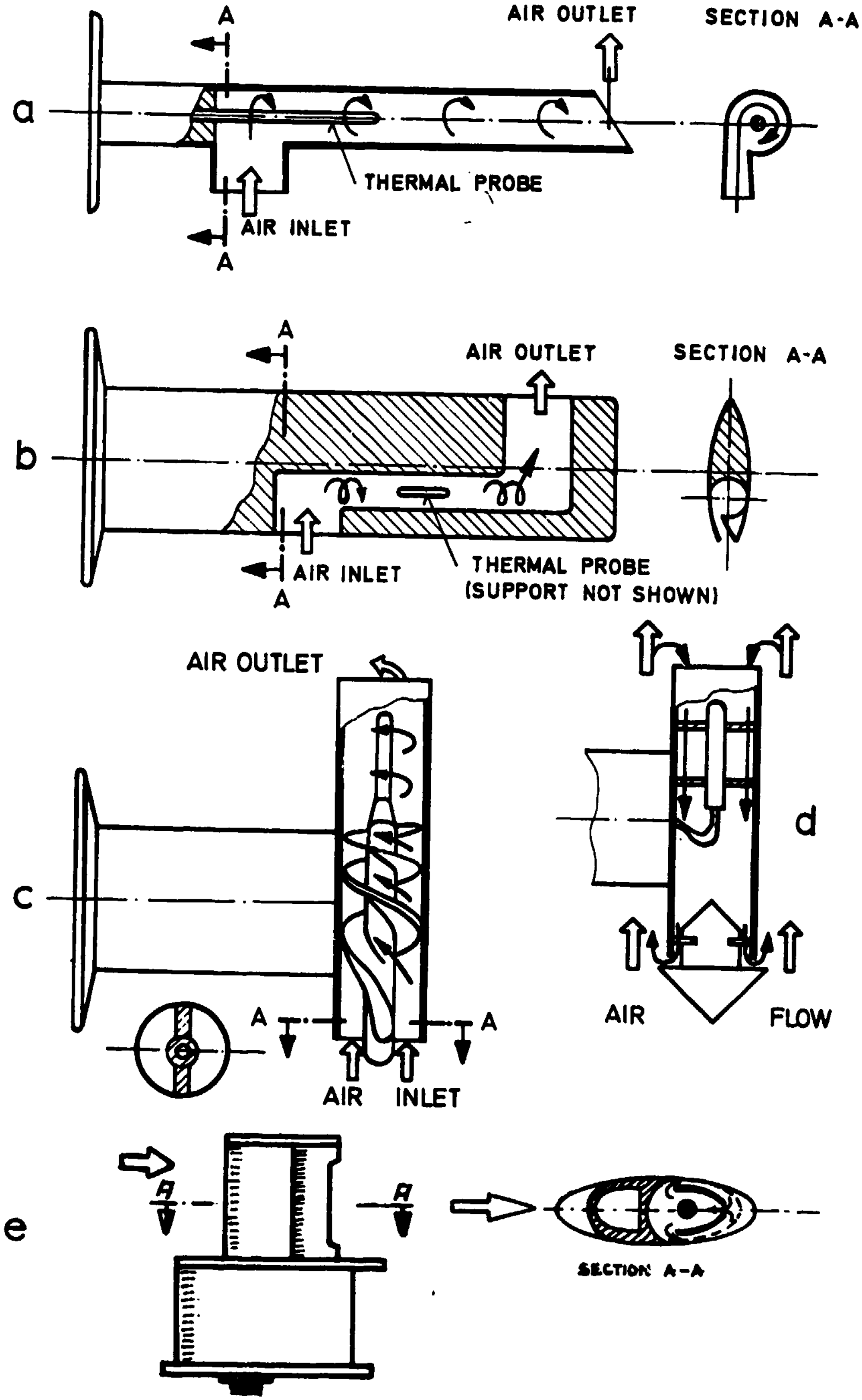


Fig.18 Internal construction of TAT probes with air flow deflection in front of the elements



(a) tangential flow vortex temp. probe
(b) tangential flow vortex temp. probe
(c) axial flow vortex temp. probe
(d) reverse flow probe
(e) reverse flow probe for helicopters

Fig. 19 Special probes

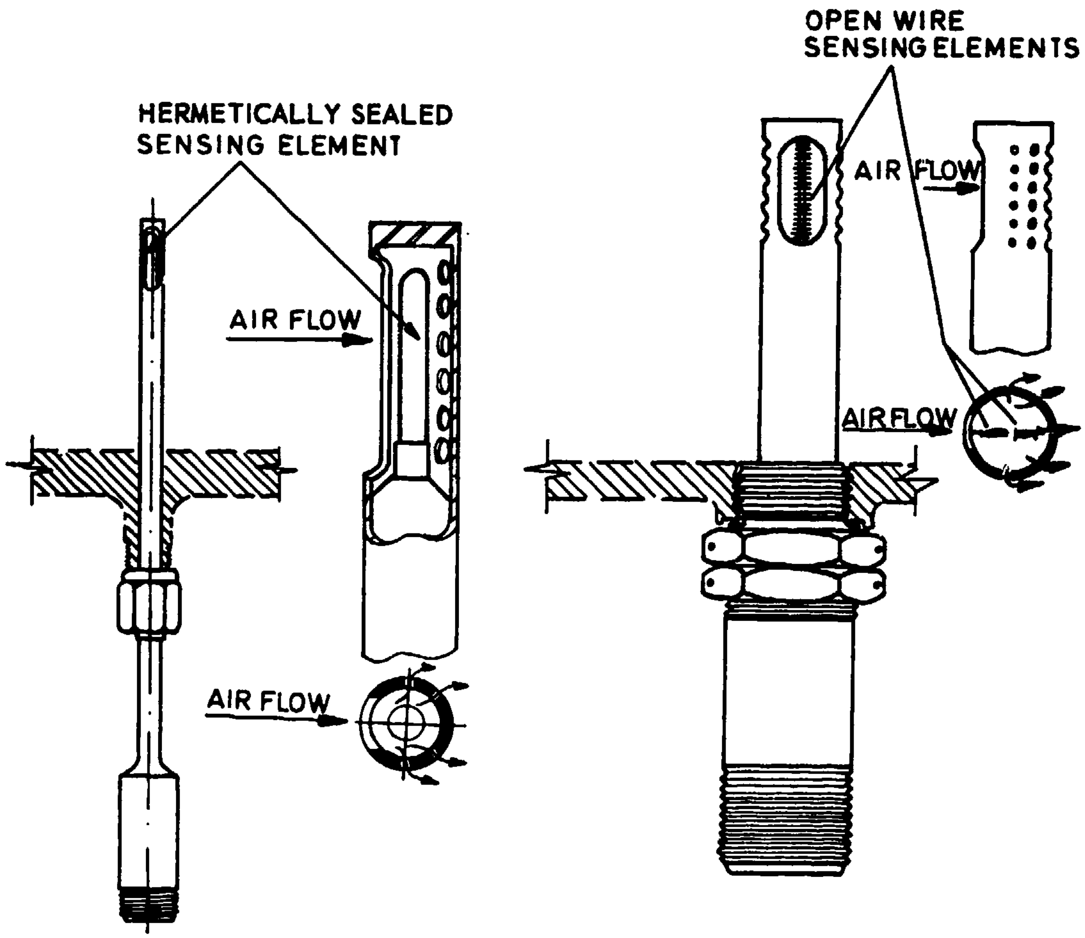


Fig.20 Internal construction of simple TAT probes for compressor inlet temperature measurements

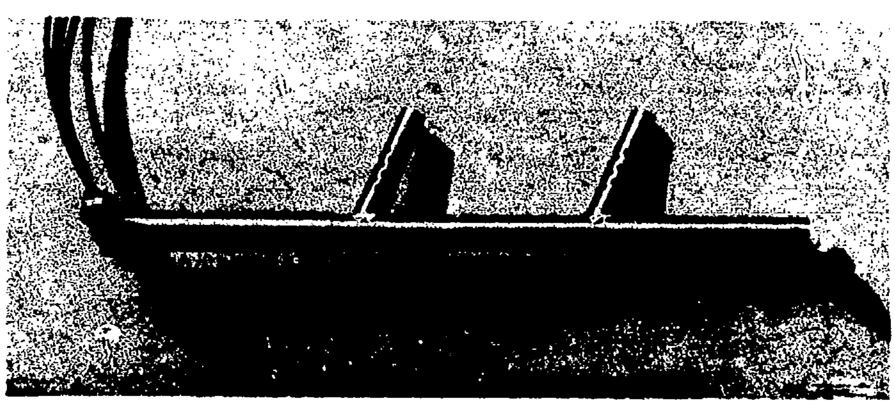


Fig.21 Special mounting arm for Rolls Royce engine with 2 TAT probes for compressor inlet temperature measurements

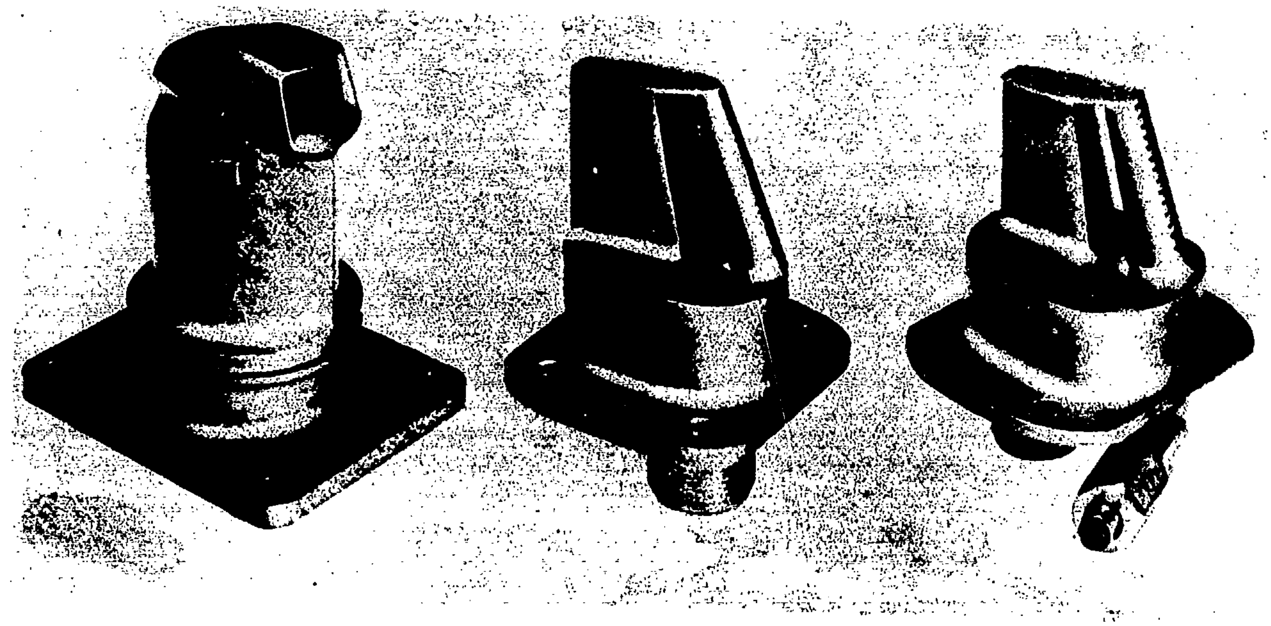
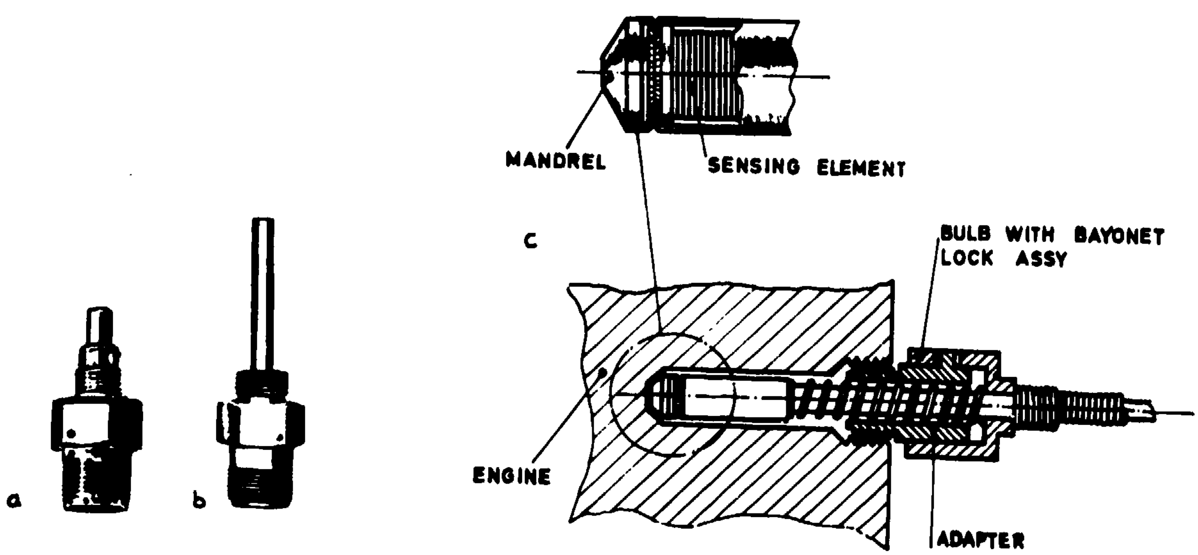


Fig.22 Compressor inlet temperature probes for wall mounting



(a,b) fixed well-type probes
(c) spring-loaded type with bayonet lock assembly

Fig.23 Tip-sensitive temperature probes for measurements in solids

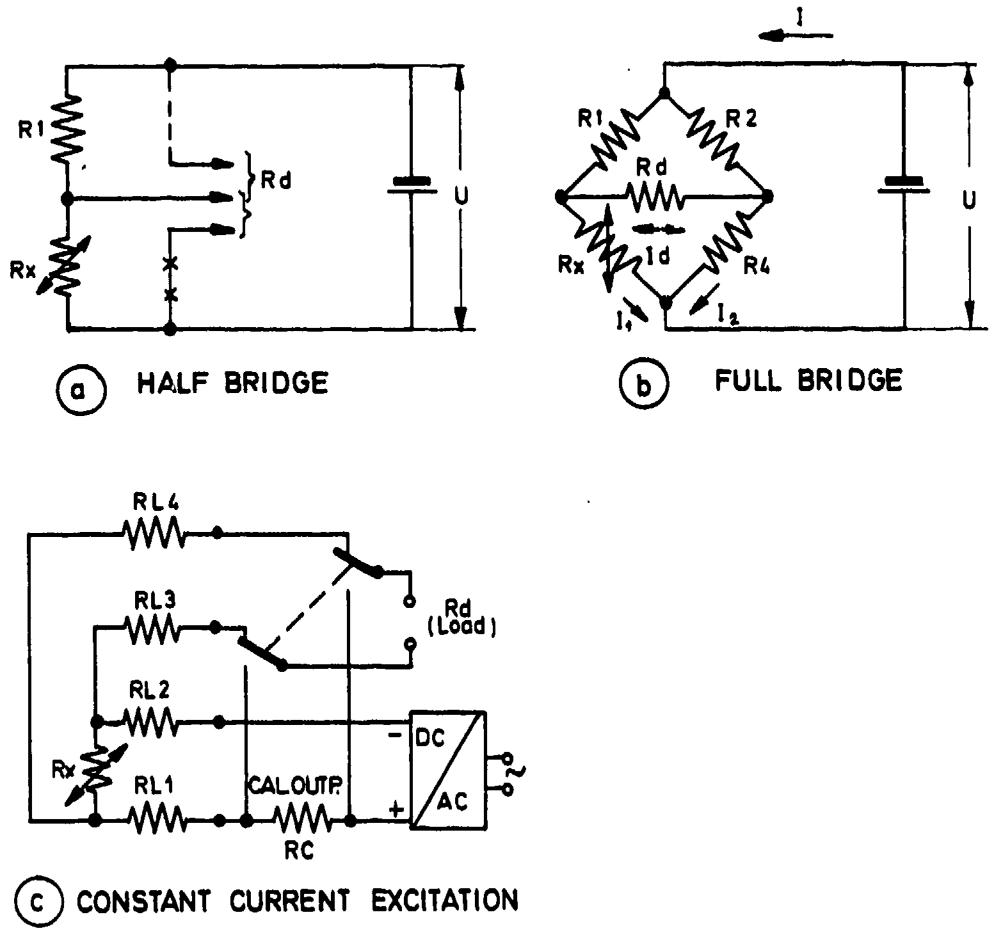


Fig.24 Typical bridge circuits

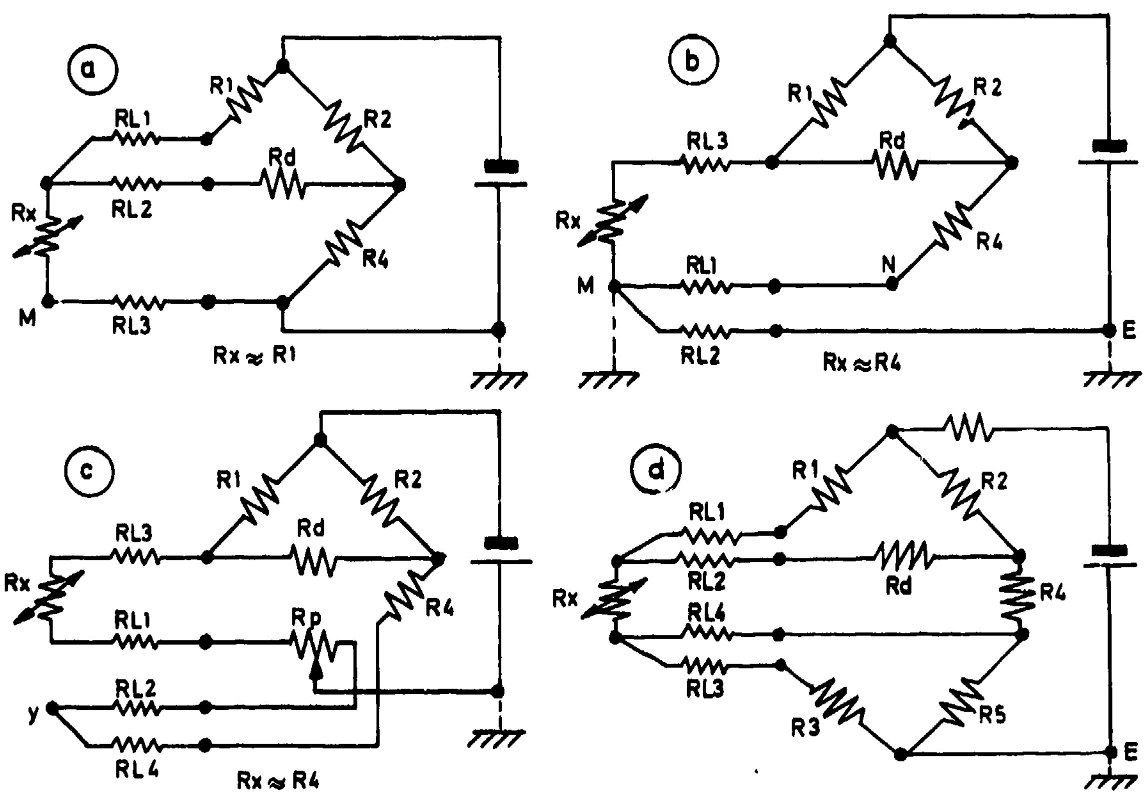
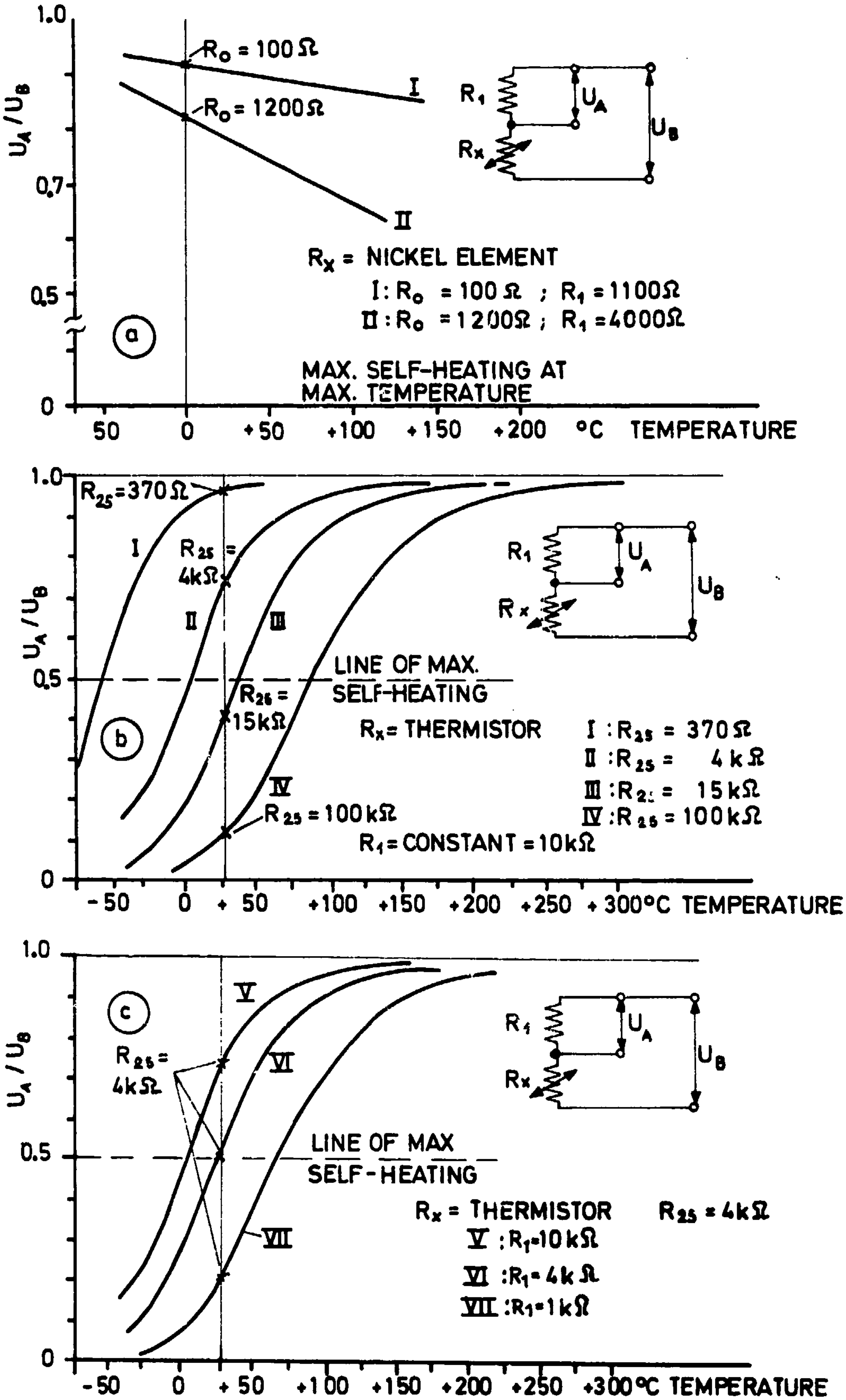
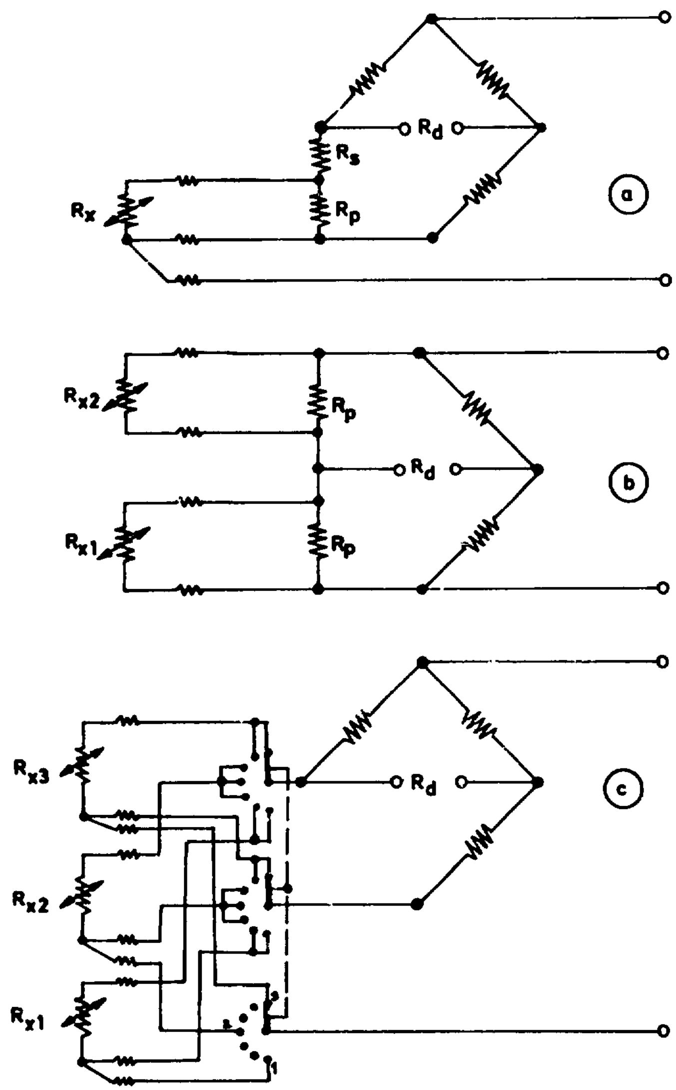


Fig.25 Compensation of lead-resistances



(a) with different Ni elements
(b) with different thermistors
(c) with different values of R_1

Fig.26 Output of half-bridges vs temperature



(a) linearization of nickel element
(b) measurement of differential temperature or temperature changing rate
(c) 3 probes with common indicator

Fig.27 Special circuits for:

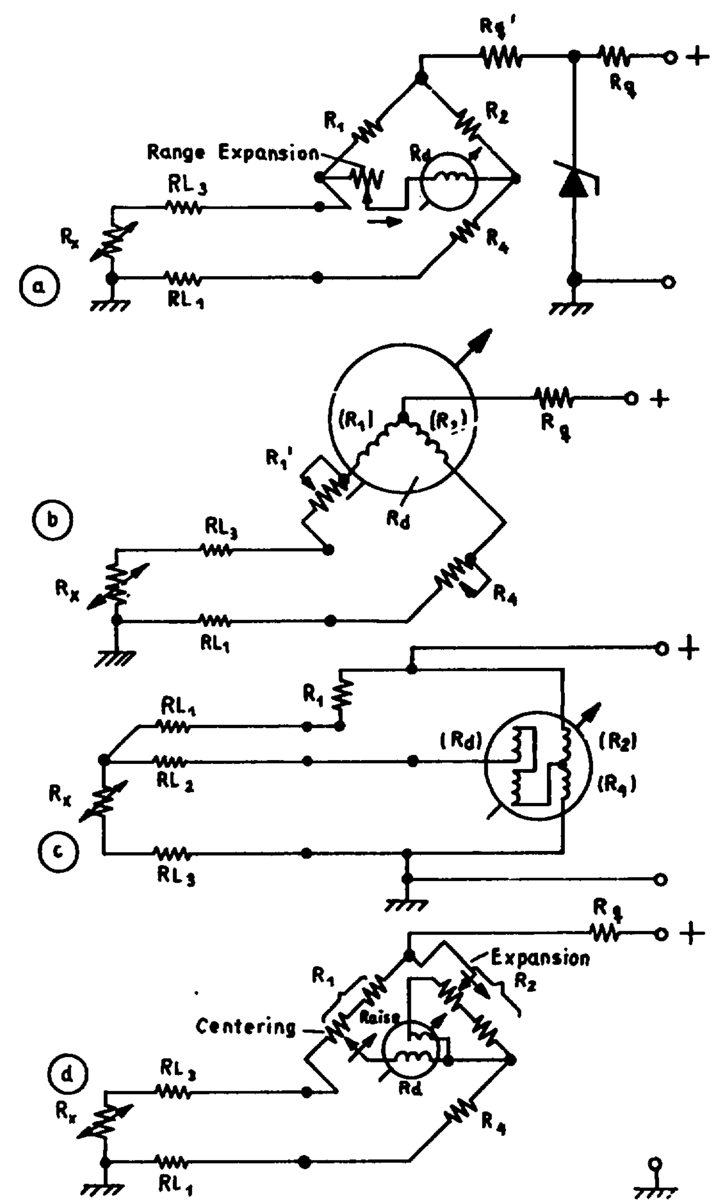
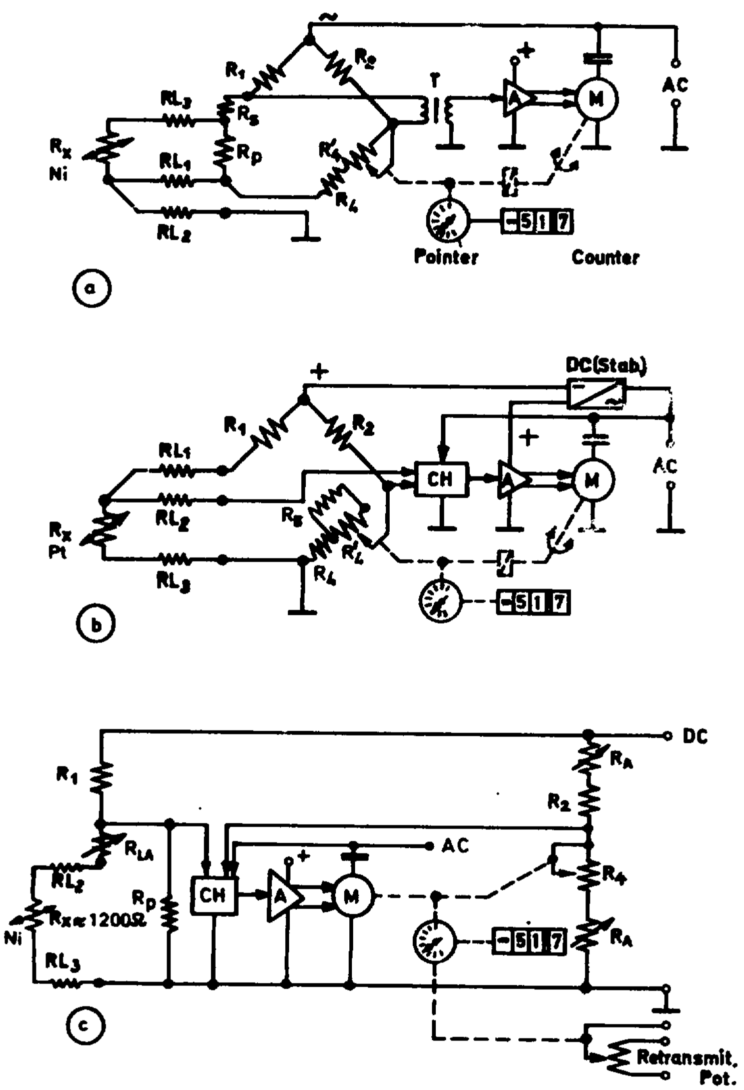


Fig.28 Circuits of indicators with galvanometer (a) and ratiometers (b, c, d)



(a) AC-bridge with Ni-probe
(b) DC-bridge with Pt-probe
(c) DC-bridge with Ni-probe

Fig.29 Servoed indicators

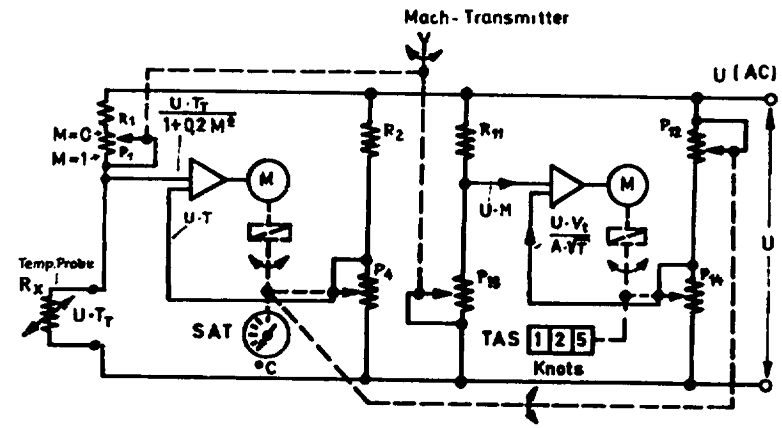


Fig.30(a) Resistance bridge circuit of SAT and TAS indicators

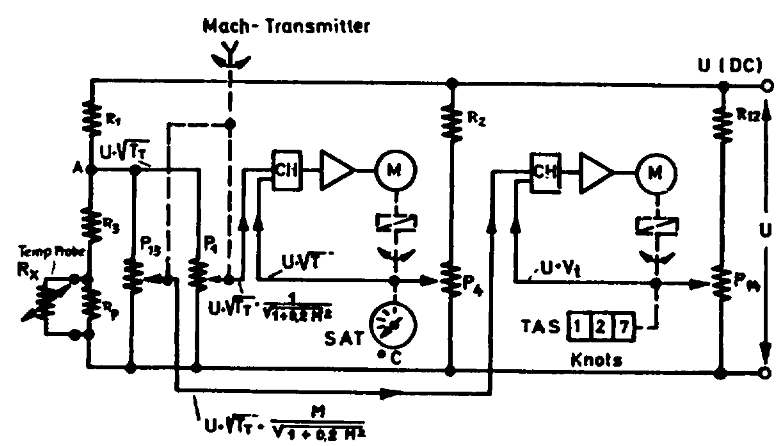
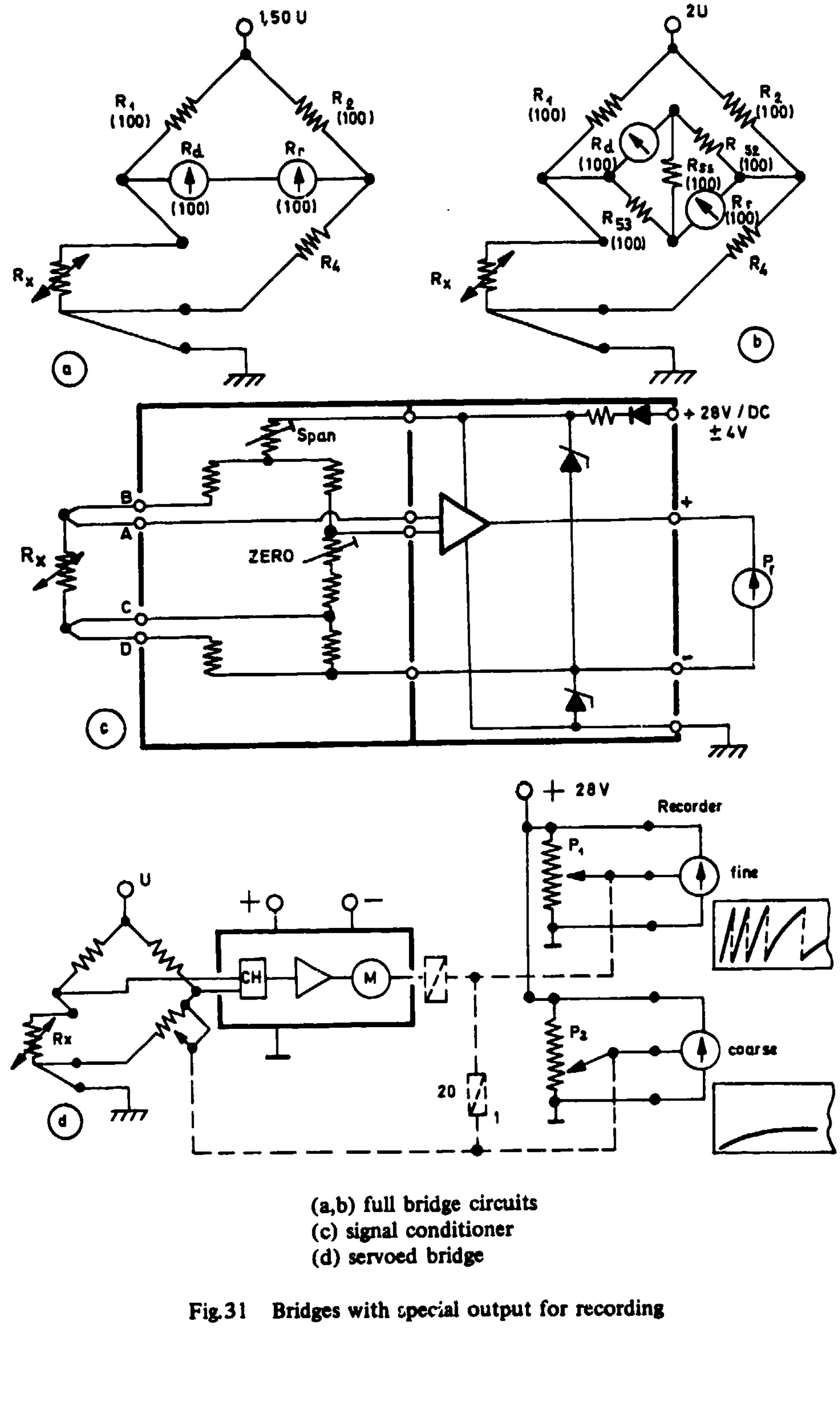
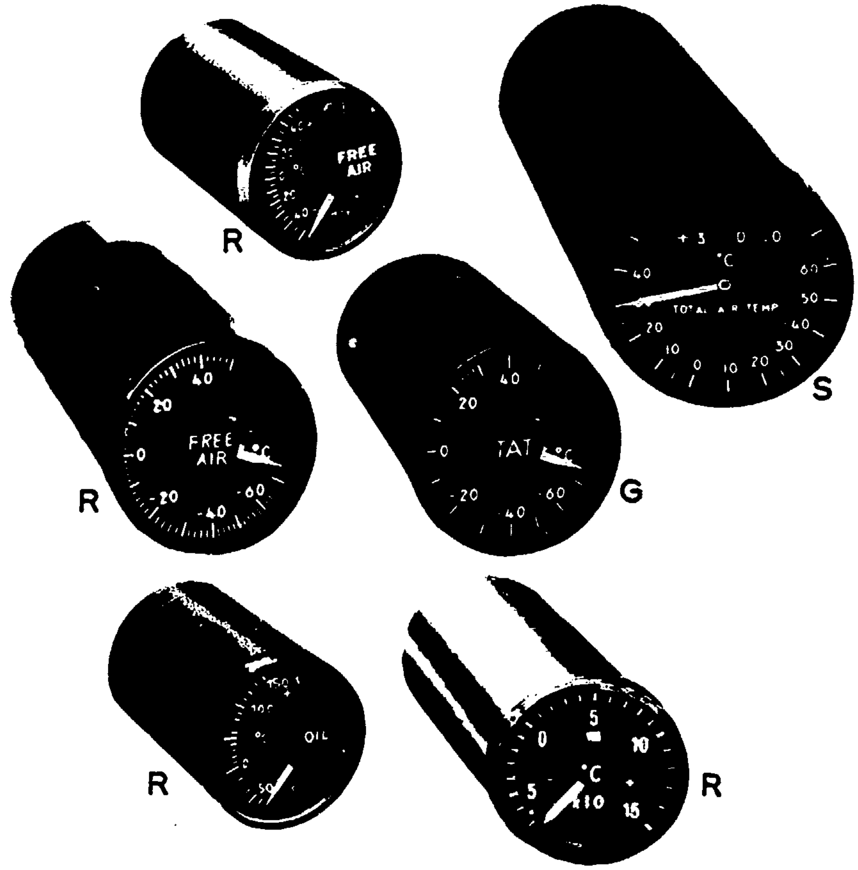


Fig.30(b) Voltage bridge circuit of SAT and TAS indicators



(a,b) full bridge circuits
(c) signal conditioner
(d) servoed bridge

Fig.31 Bridges with special output for recording



G = galvonometer types
R = ratiometer types
S = servoed types

Fig.32 Resistance thermometer indicators with different scale presentations

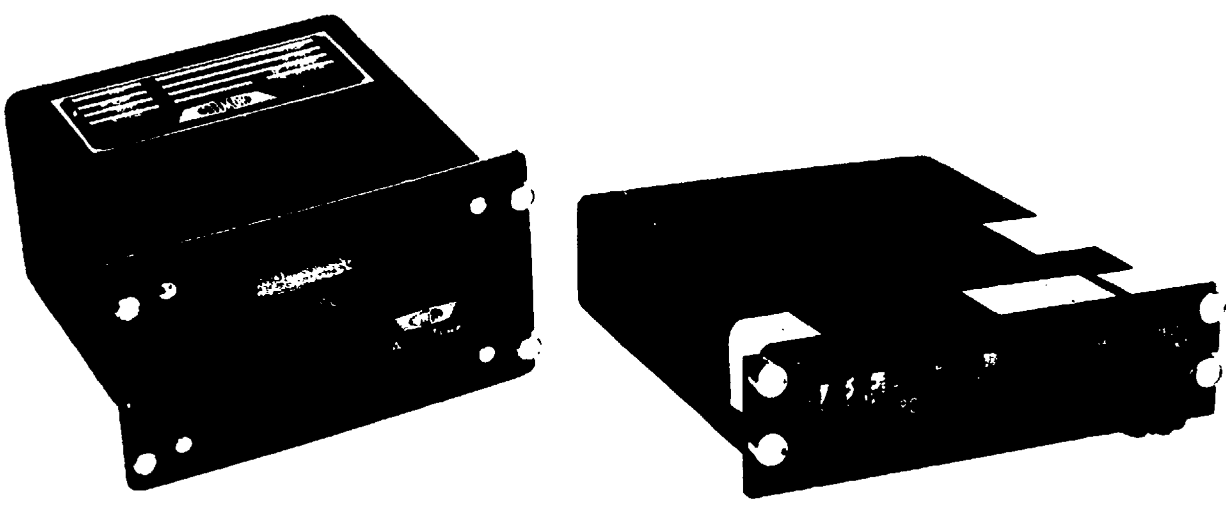


Fig.33 Servoed indicators with digital scale presentation and 2 potentiometer outputs (coarse and fine) for recording

Fig.34 Different types of servoed indicators, single and combined versions

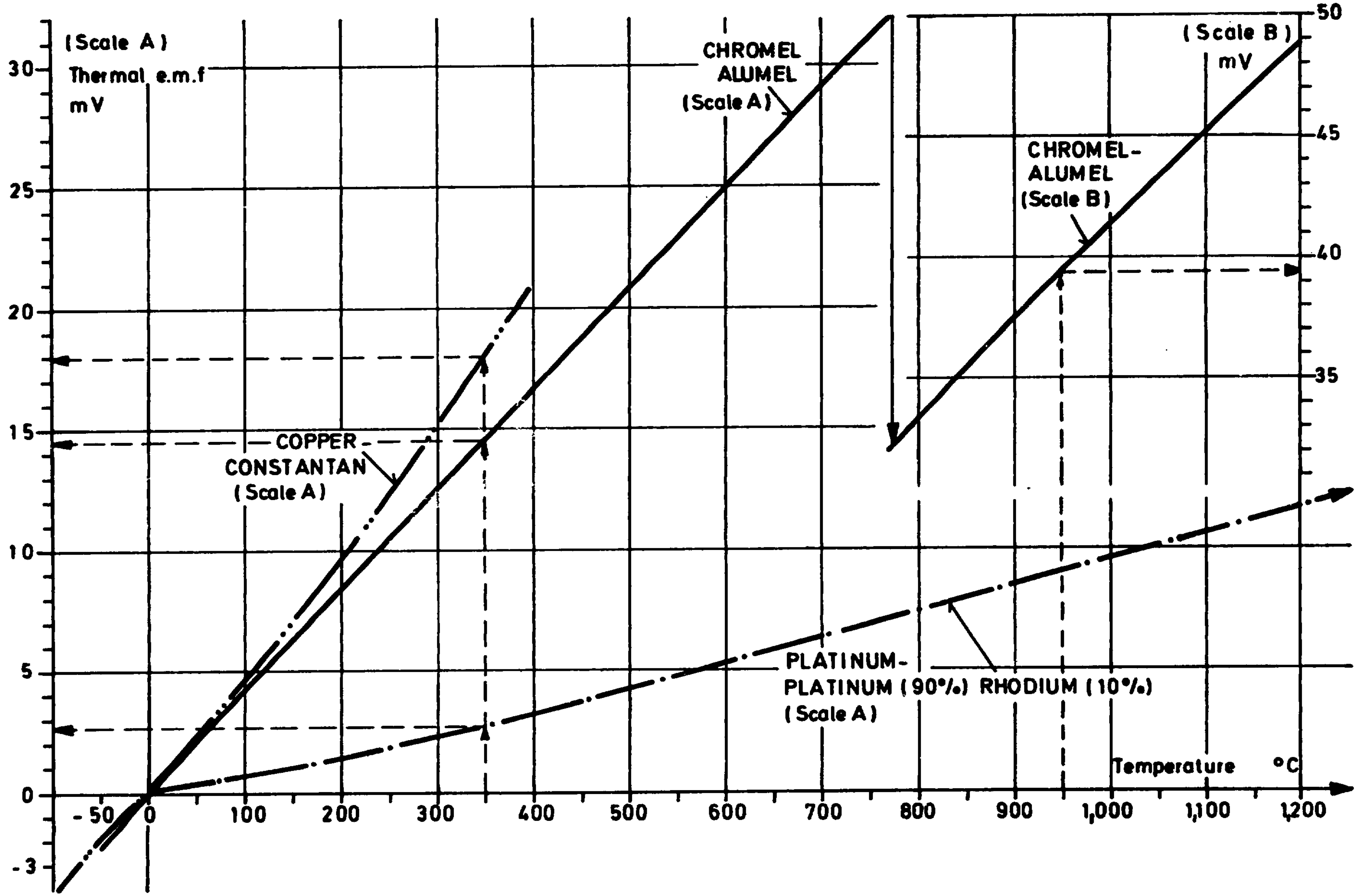
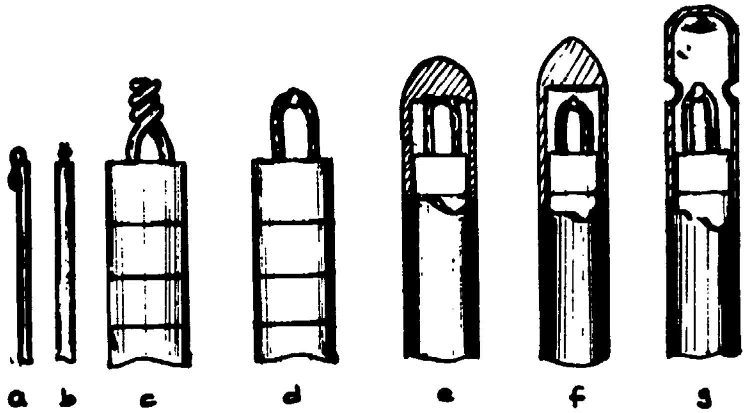


Fig.35 Thermocouples, thermal e.m.f. vs temperature



(a) thermocoax, one-lead
(b) thermocoax, twin-lead
(c) double bore ceramic bead (twisted contact)
(d) double bore ceramic bead (welded contact)
(e) well-type, hermetic (grounded junction)
(f) well-type, hermetic (ungrounded junction)
(g) exposed junction with shield

Fig.36 Construction of thermocouples

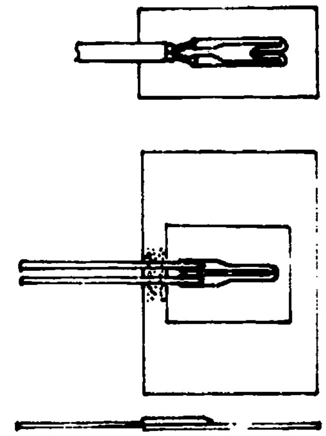
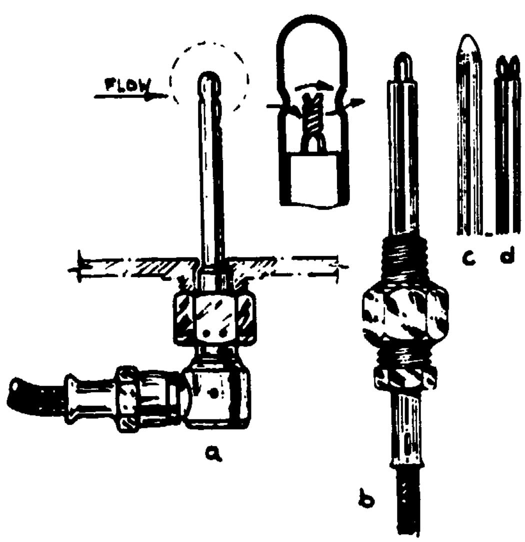


Fig.37 Foil thermocouples



(a) probe with fixed fitting
(b) probe with adjustable fitting (shown with different elements c and d)

Fig.38 Thermocouple probes

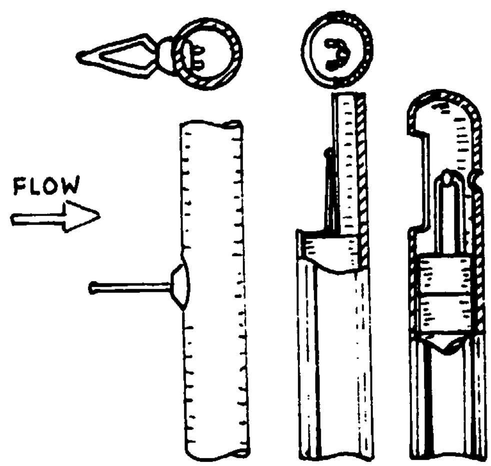


Fig.39 Simple TAT probes with thermocouples

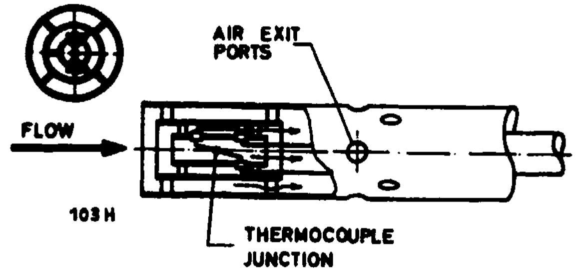
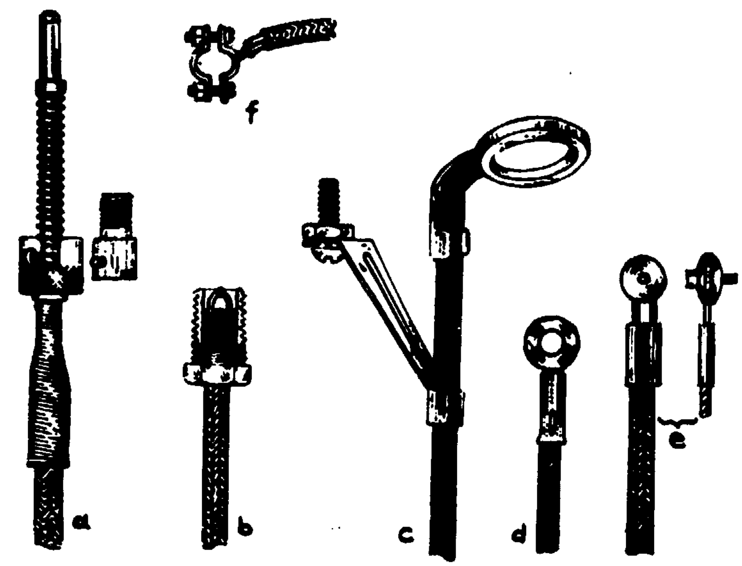


Fig.40 Special TAT probe with thermocouple



(a) spring loaded bayonet lock type
(b) brake drum thermocouple probe (open tip)
(c) gasket-brazed junction
(d) little gasket-type junction
(e) blind rivet junction
(f) junction with pipe clamp adapter

Fig.41 Thermocouple probes for measurements on solids

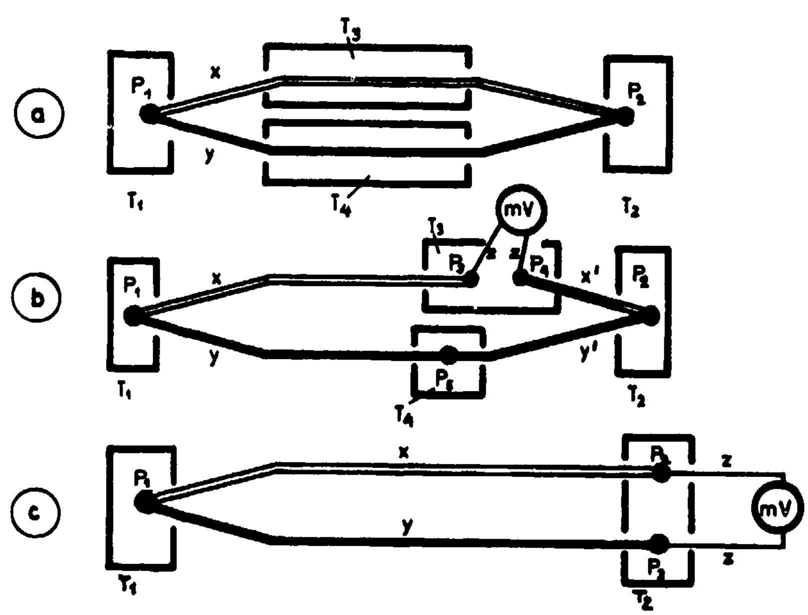


Fig.42 Circuits with thermocouples

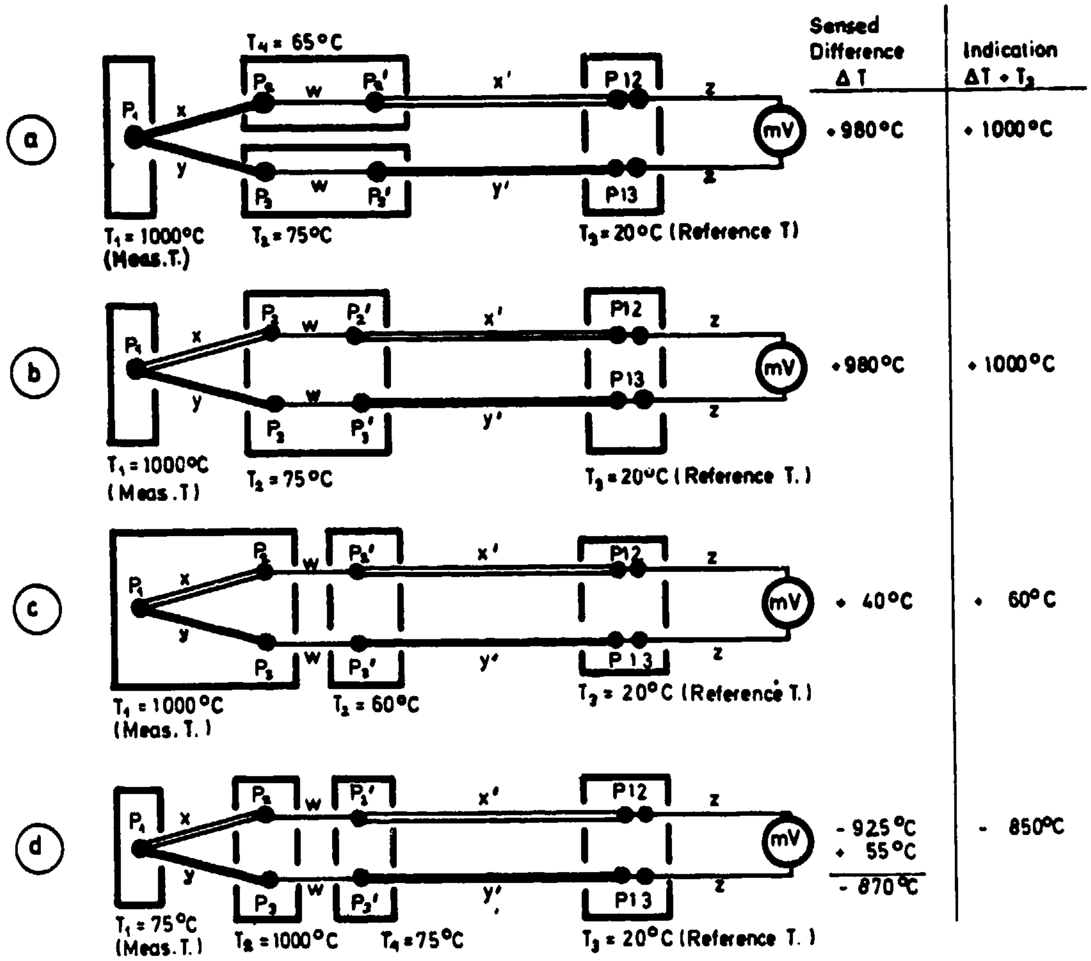


Fig.43 Temperature indications with different locations of the gradient

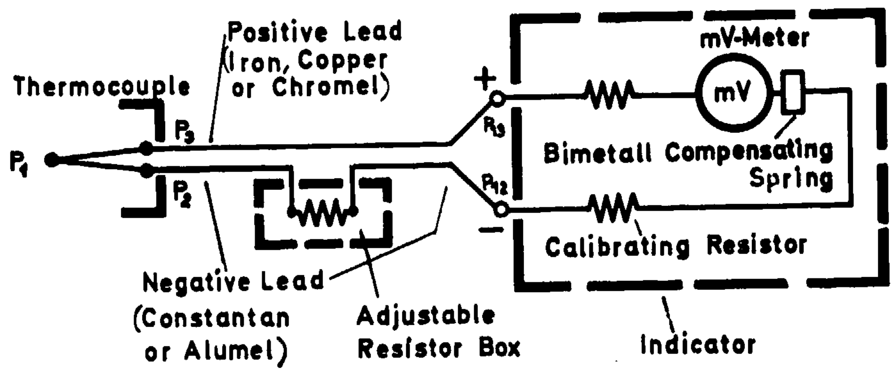


Fig.44 Wiring diagram -- simple thermocouple thermometer

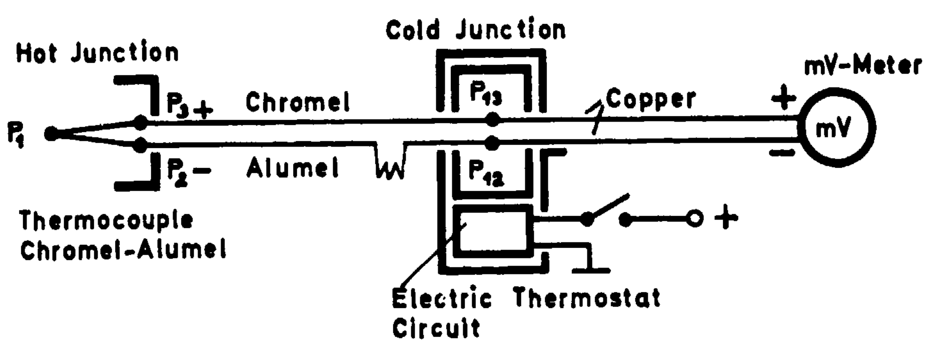


Fig.45 Wiring diagram – thermocouple thermometer with thermostat as reference point

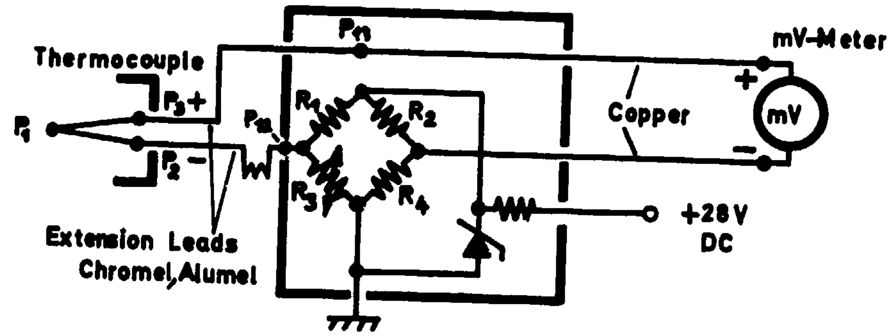
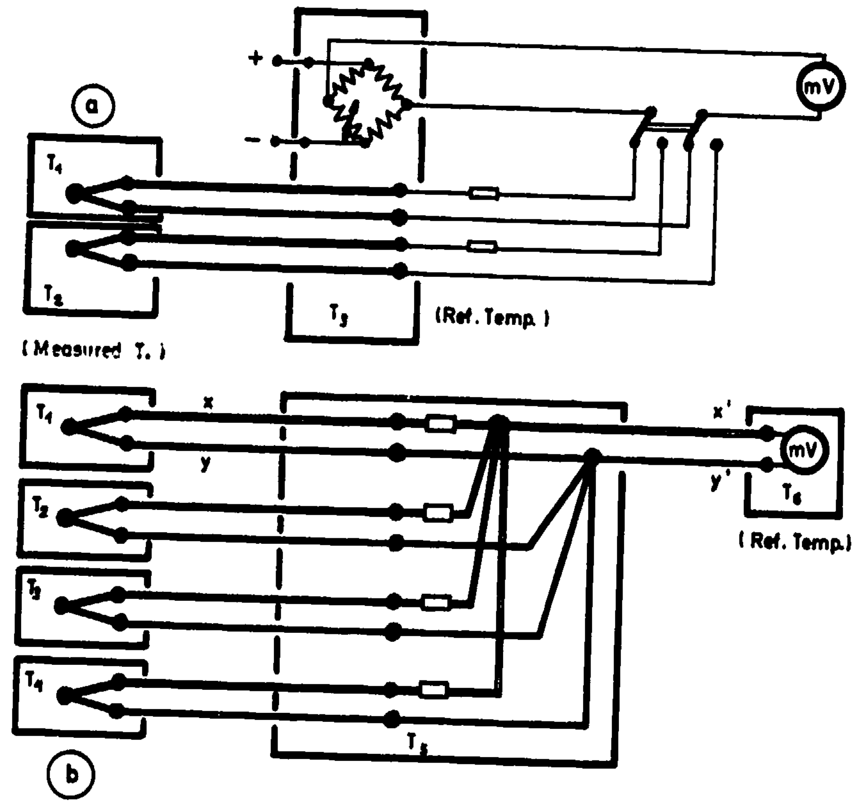


Fig.46 Wiring diagram – thermocouple thermometer with compensating bridge circuit



(a) measurement on different positions
(b) measurement of mean temperature

Fig.47 Special circuits

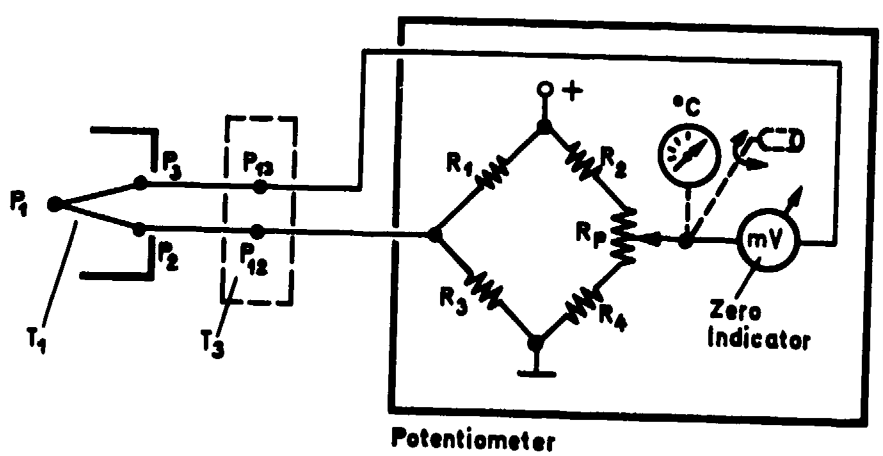


Fig.48 Thermocouple with potentiometer circuit

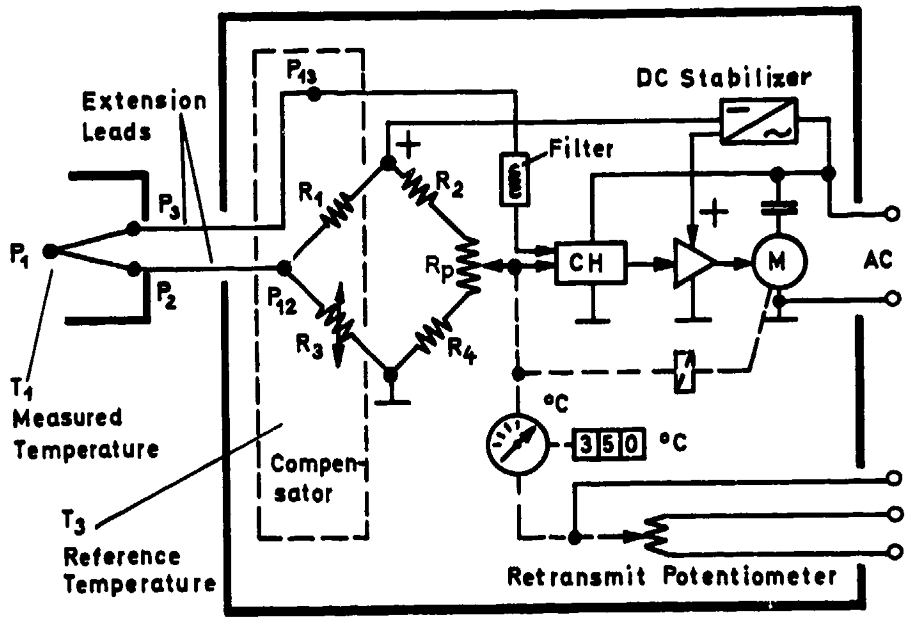
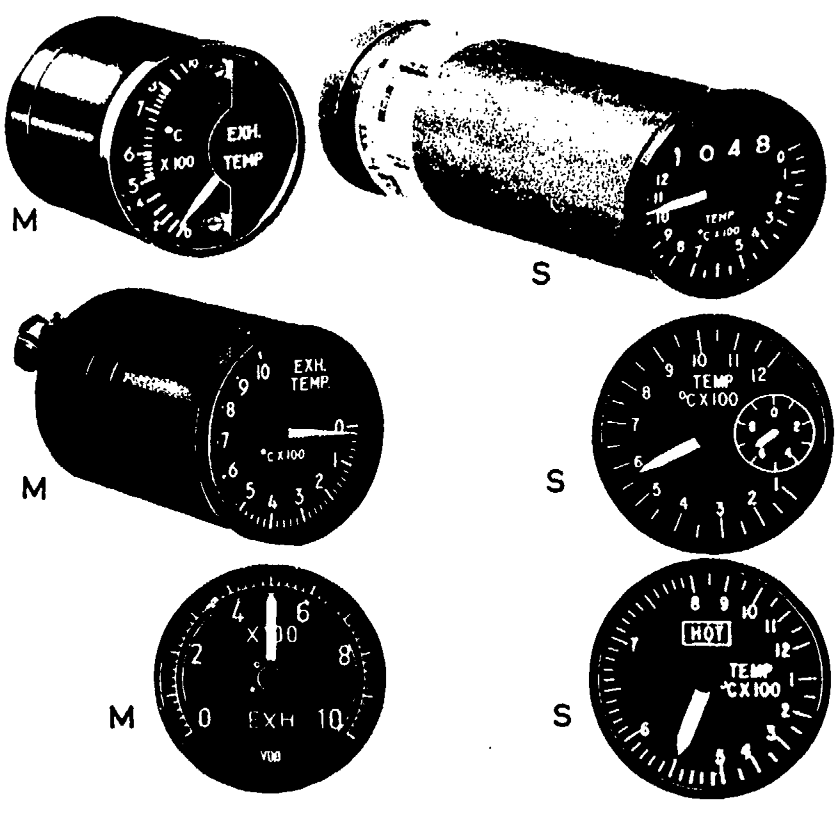


Fig.49 Thermocouple with servo temperature indicator



left: simple indicators = millivoltmeters (M)
right: servoed indicators (S)

Fig.50 Thermocouple indicators with different scale presentations

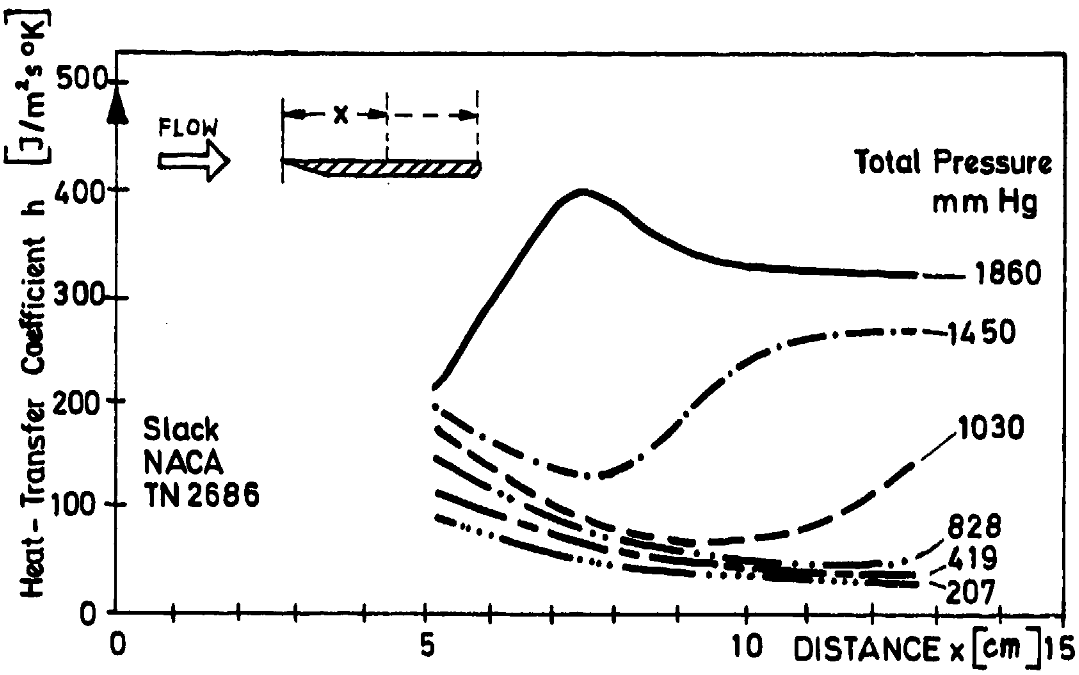


Fig.51 Heat transfer coefficient vs distance and pressure

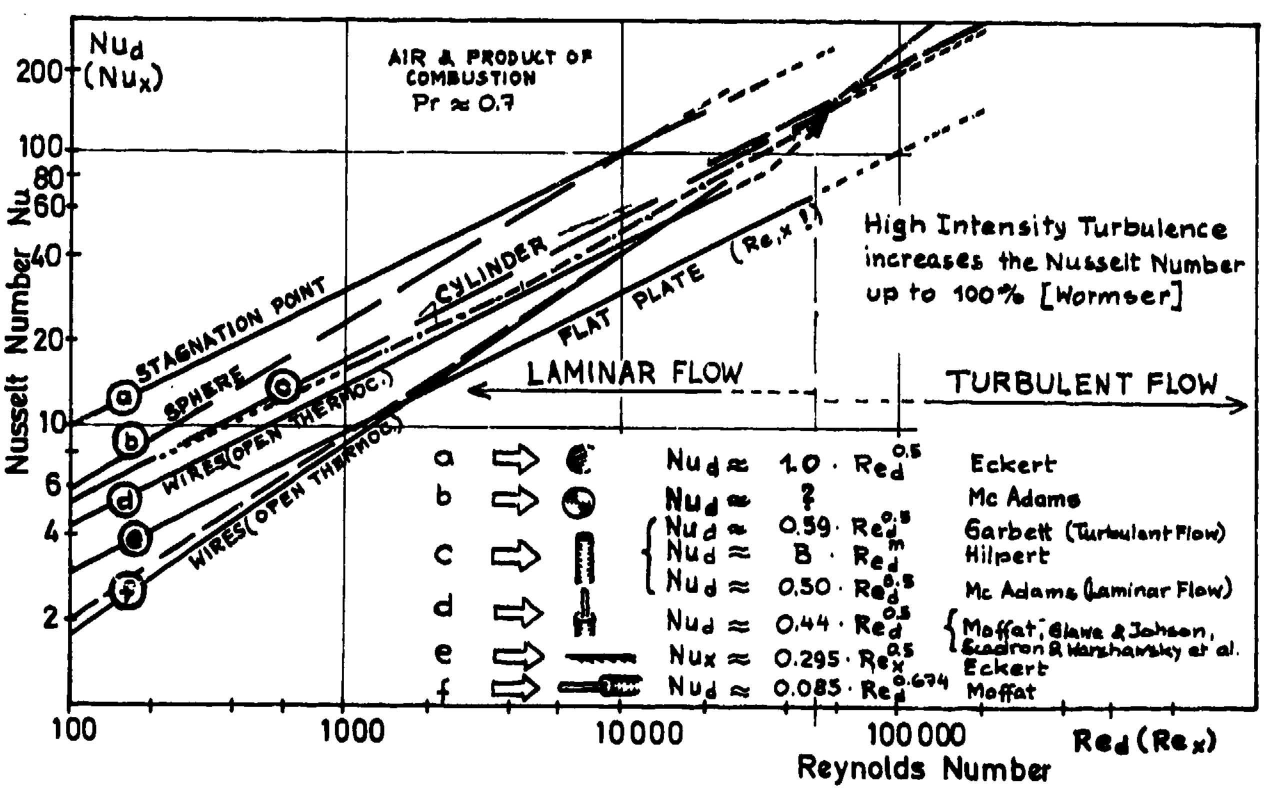


Fig.52 Nusselt number as a function of Reynolds number and shape of probe

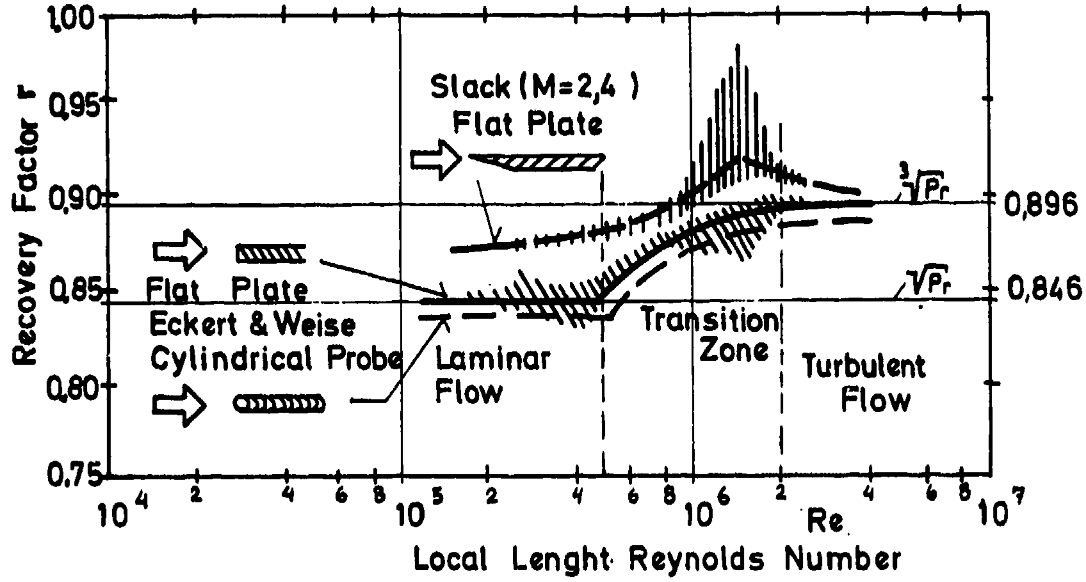


Fig.53 Recovery factor as a function of Reynolds number and shape of probe

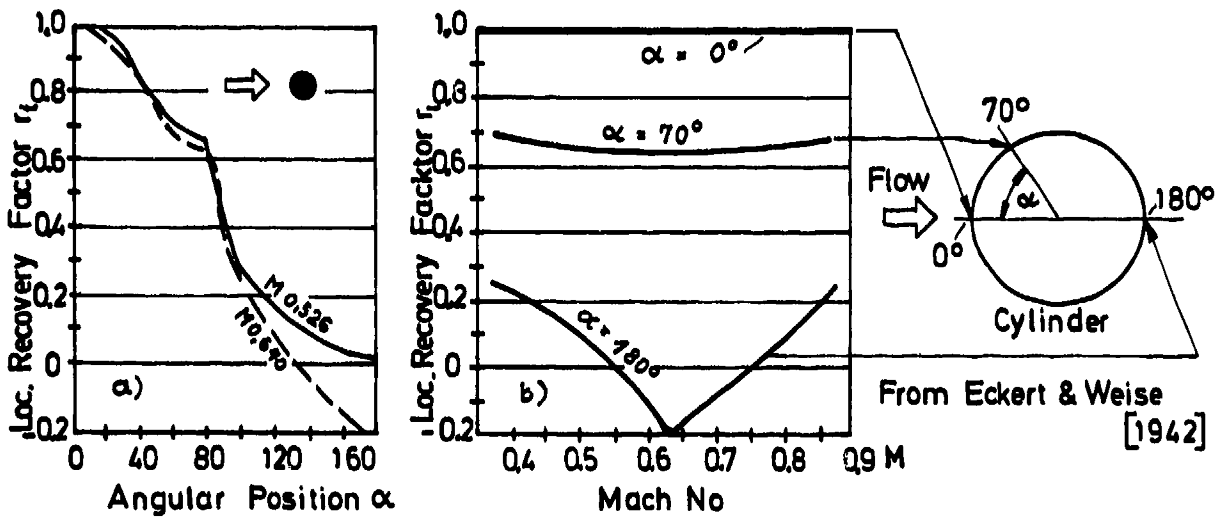


Fig.54 Distribution of local recovery factors around a cylinder in cross flow vs Mach number

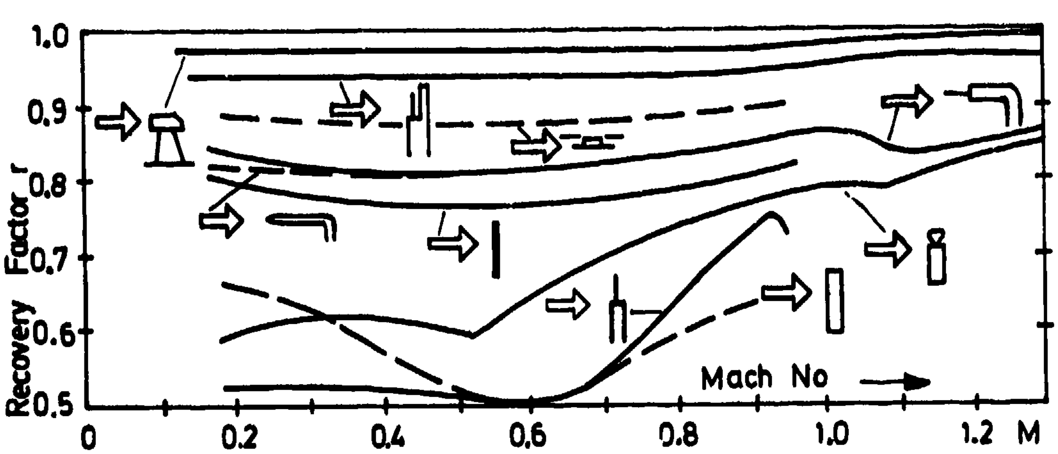


Fig.55 Recovery factors of different probes vs Mach number (for typical mounting positions)

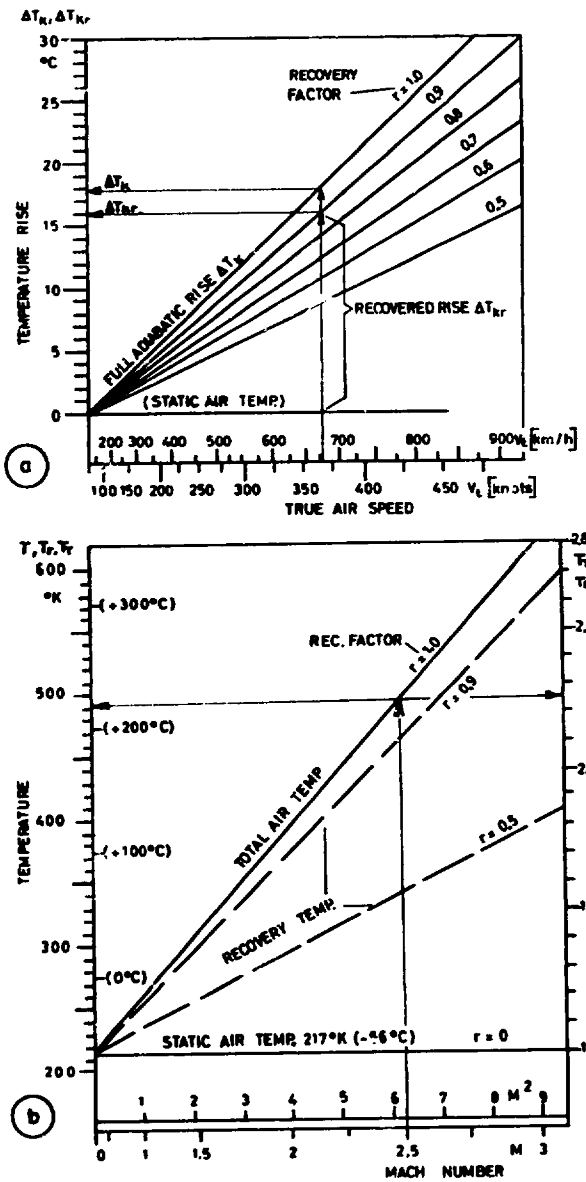


Fig.56 Temperature rise as a function of true airspeed, Mach number and recovery factor

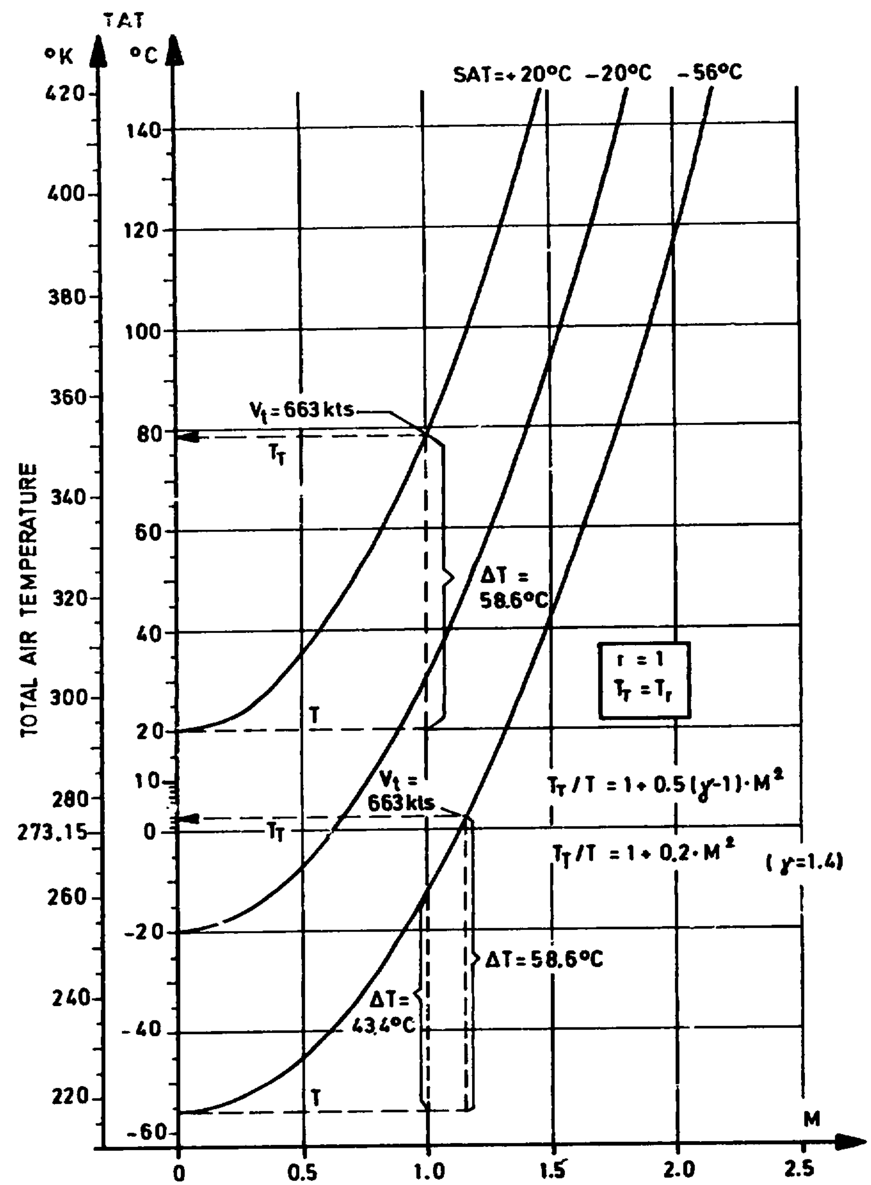


Fig.57 Temperature rise for various static air temperatures vs Mach number

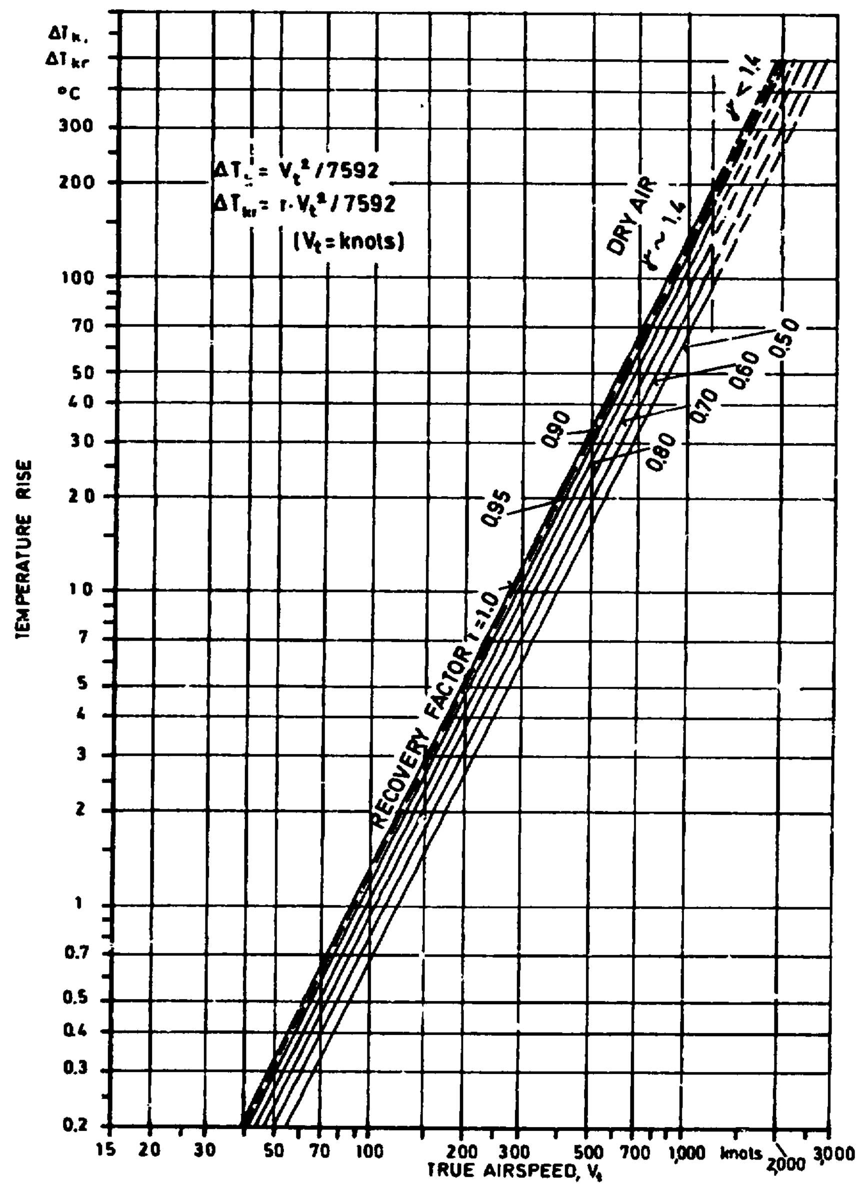


Fig.58 Kinetic temperature rise for various recovery factors vs true airspeed

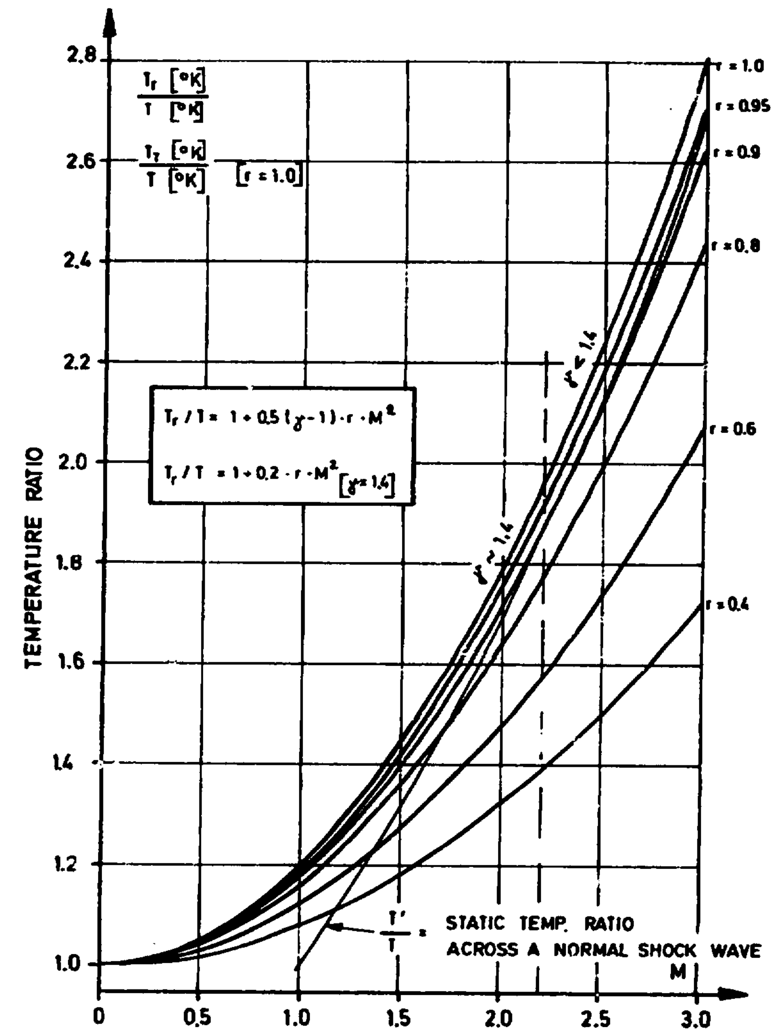


Fig.59 Kinetic temperature rise for various recovery factors vs Mach number

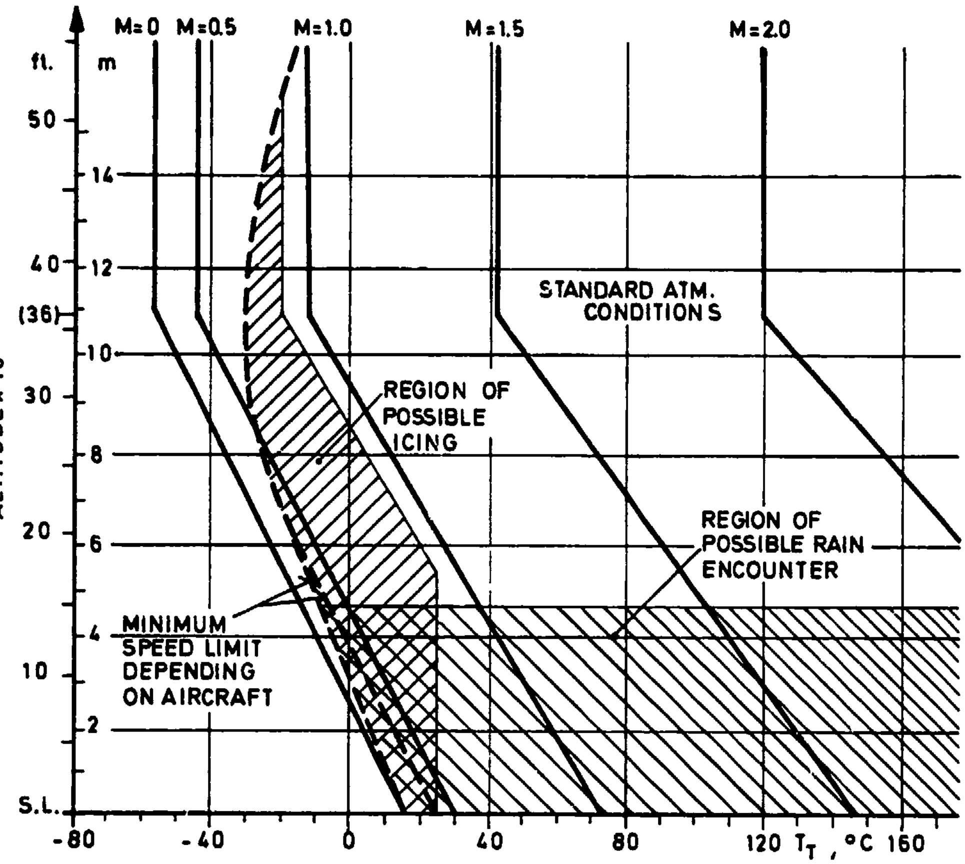


Fig.60 Total temperature as a function of altitude and Mach number

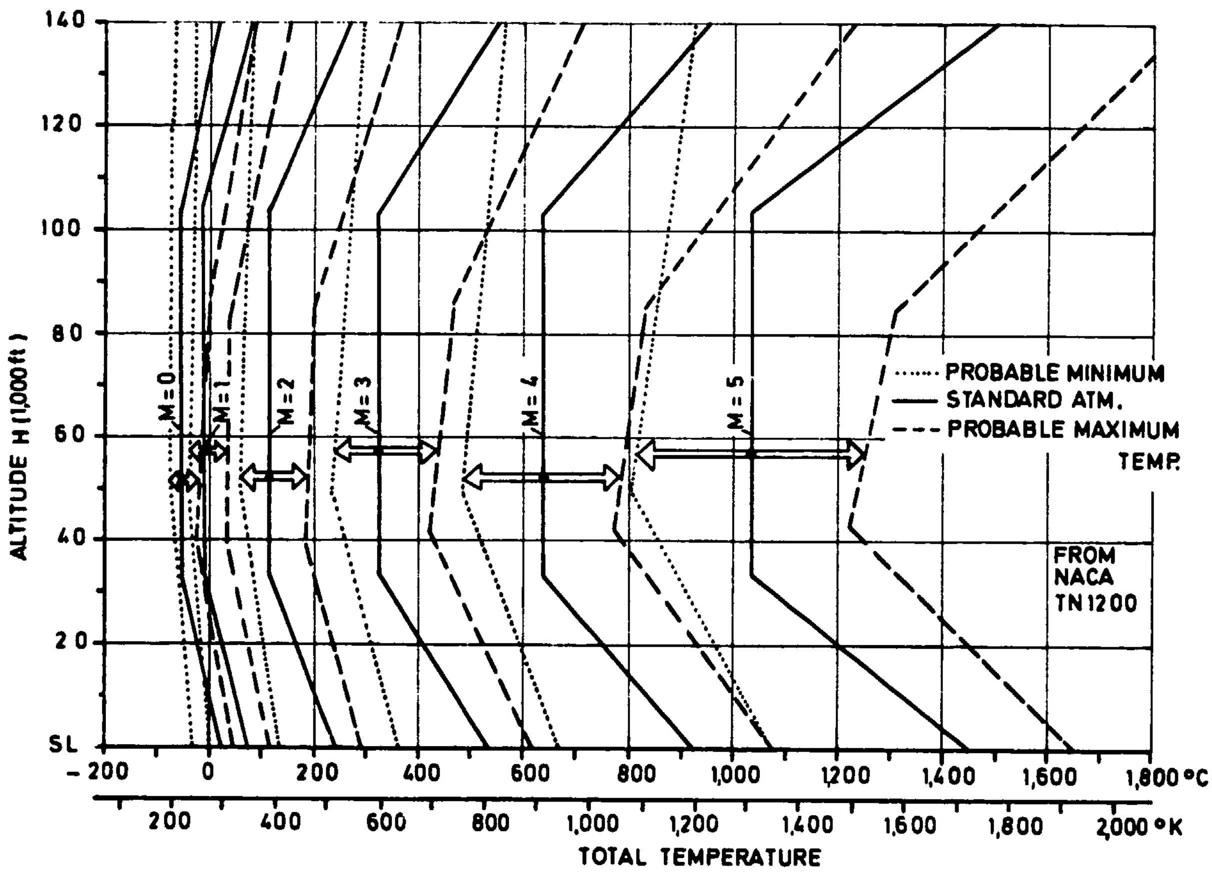


Fig.61 Total temperature as a function of altitude and Mach number

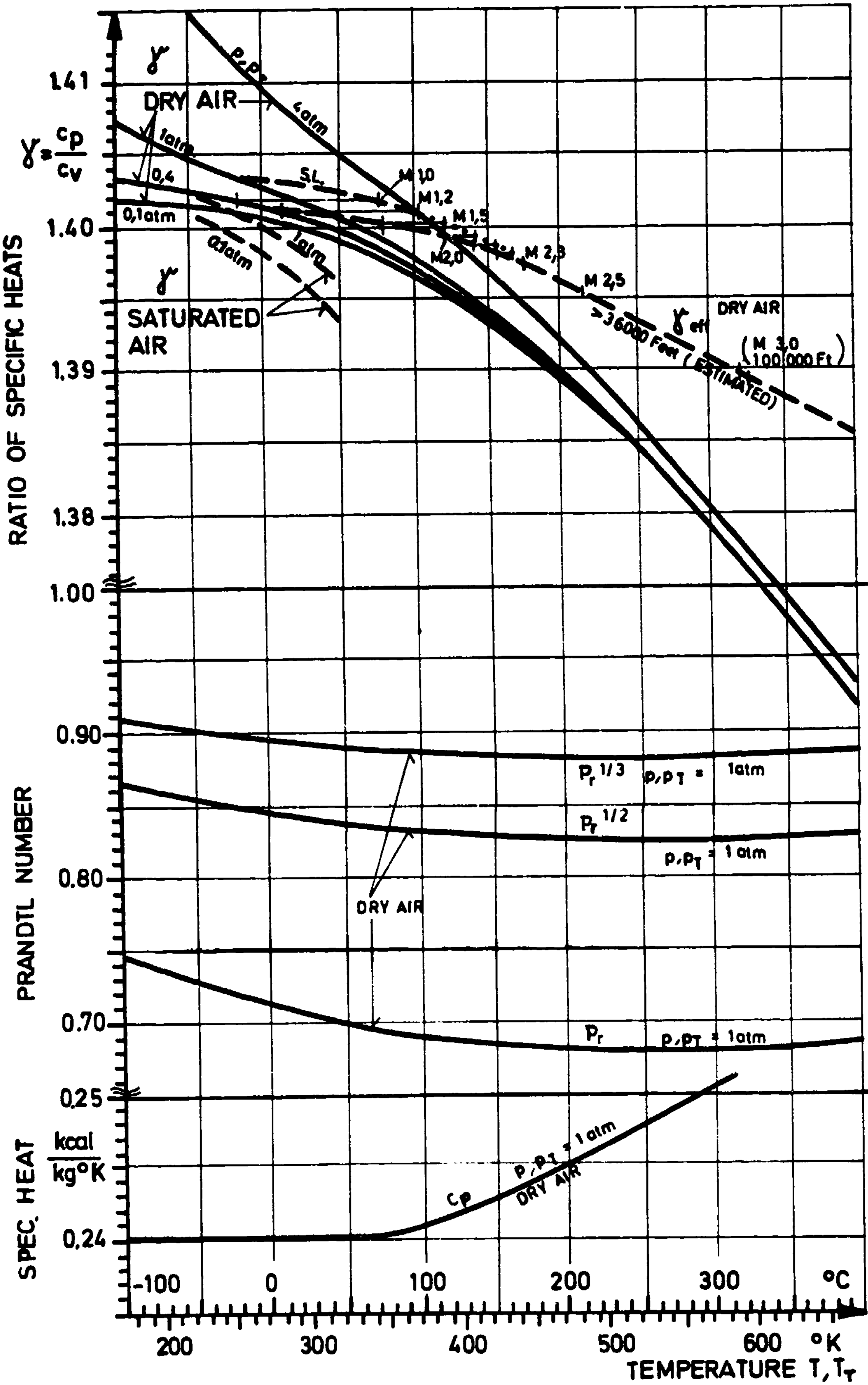


Fig.62(a) Variations of gas properties vs temperature

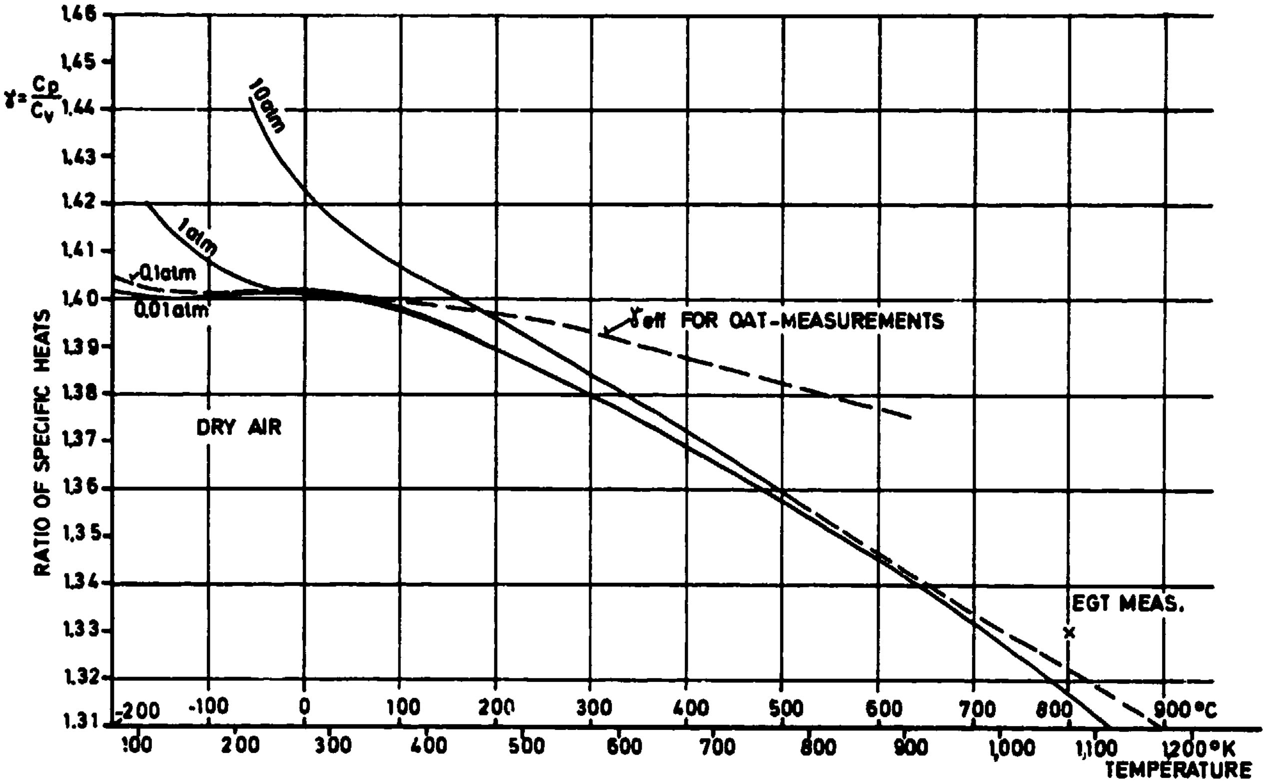


Fig.62(b) Variation of γ vs temperature

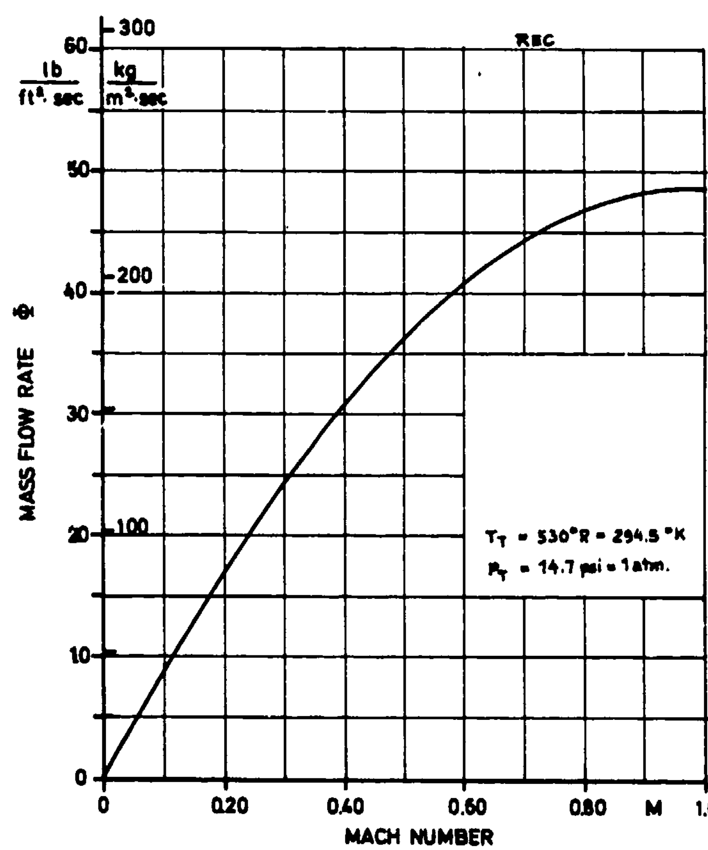


Fig.62(c) Variation of mass flow rate with Mach number for constant T_T and p_T

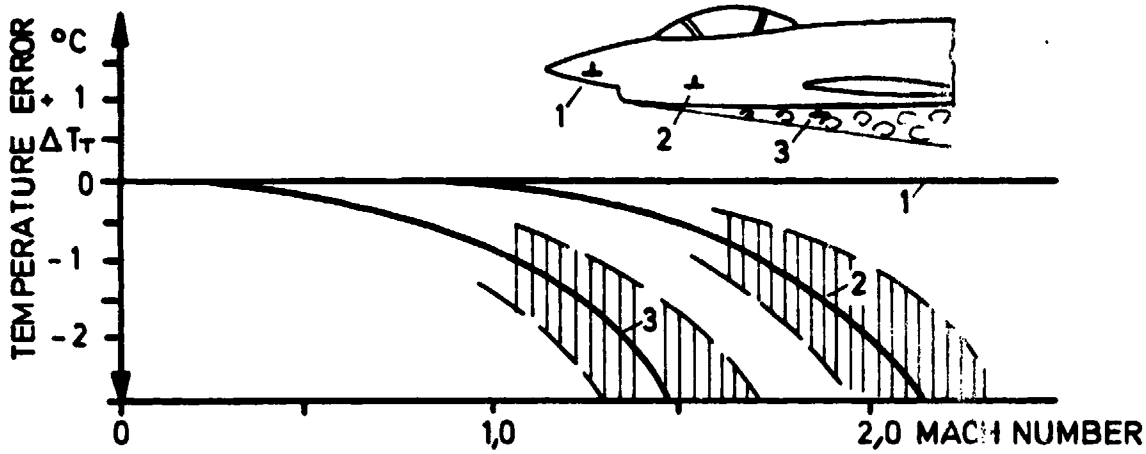


Fig.63 Typical probe location errors (estimated)

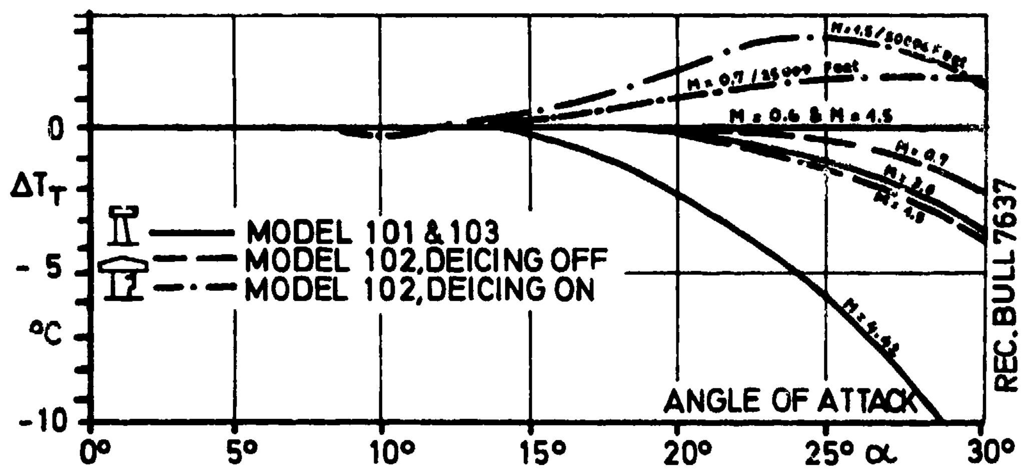


Fig.64 Attitude error of some TAT-probes

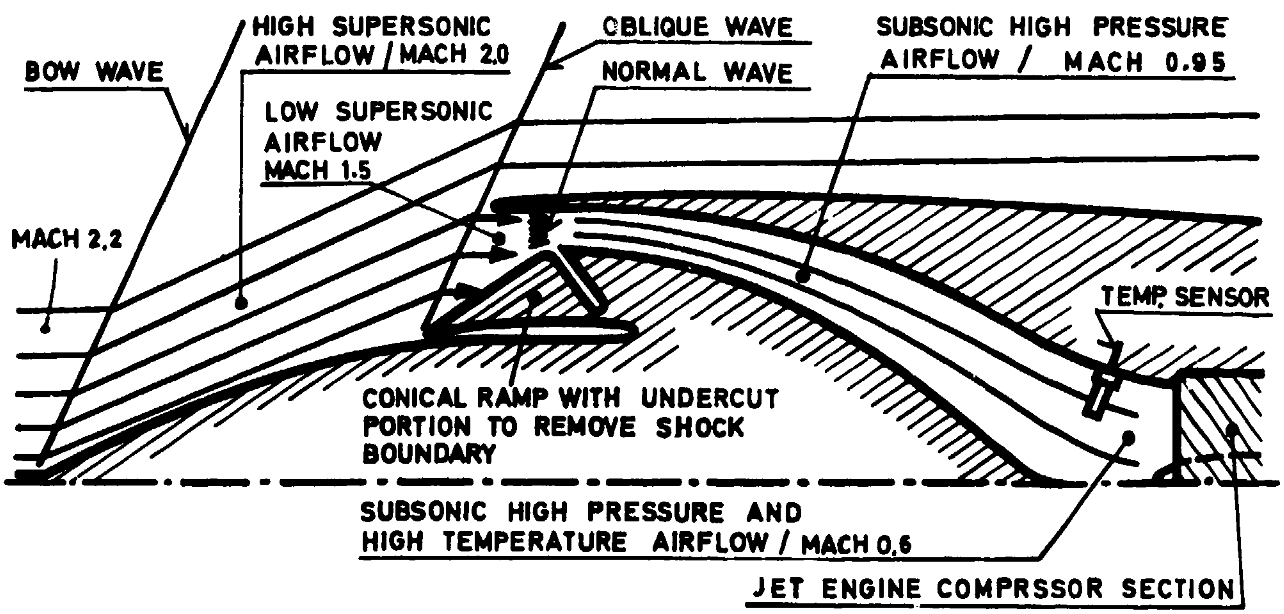


Fig.65 Compressor inlet temperature measurement (typical flow conditions at flight Mach number 2.0)

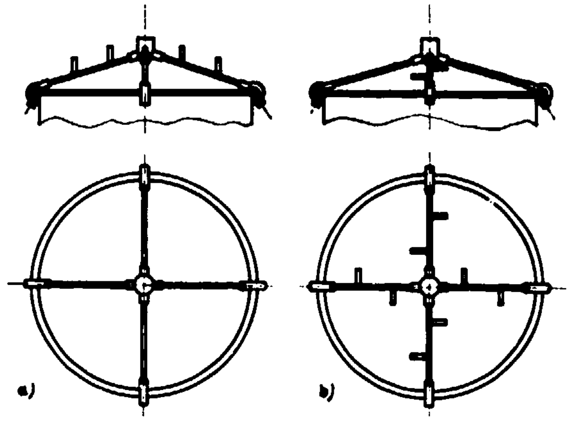


Fig.66 Alternative mounts of TAT-sensors for CIT measurement in a vertical lift engine

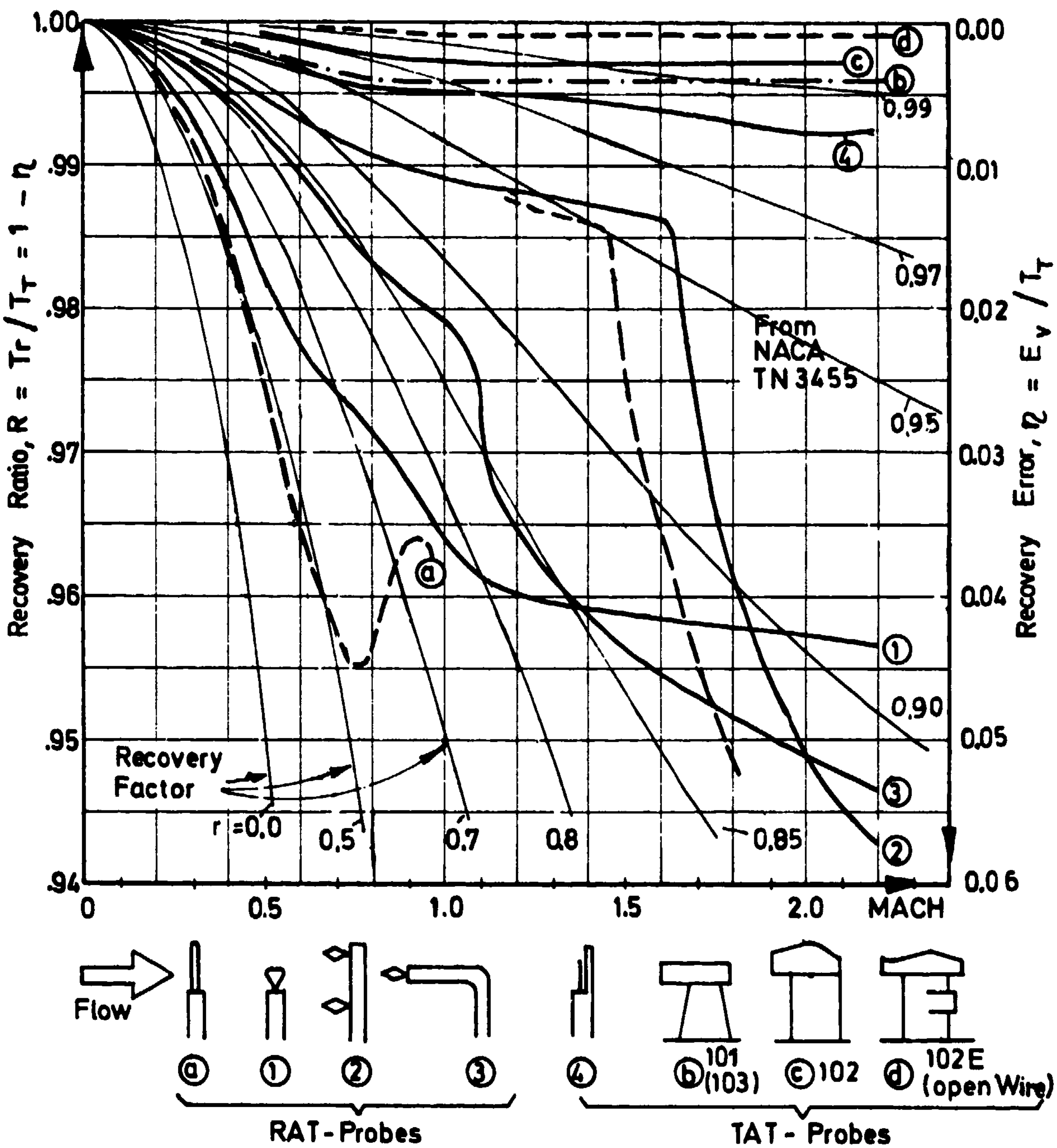


Fig.67 Recovery ratio, recovery factor and recovery error of different probes

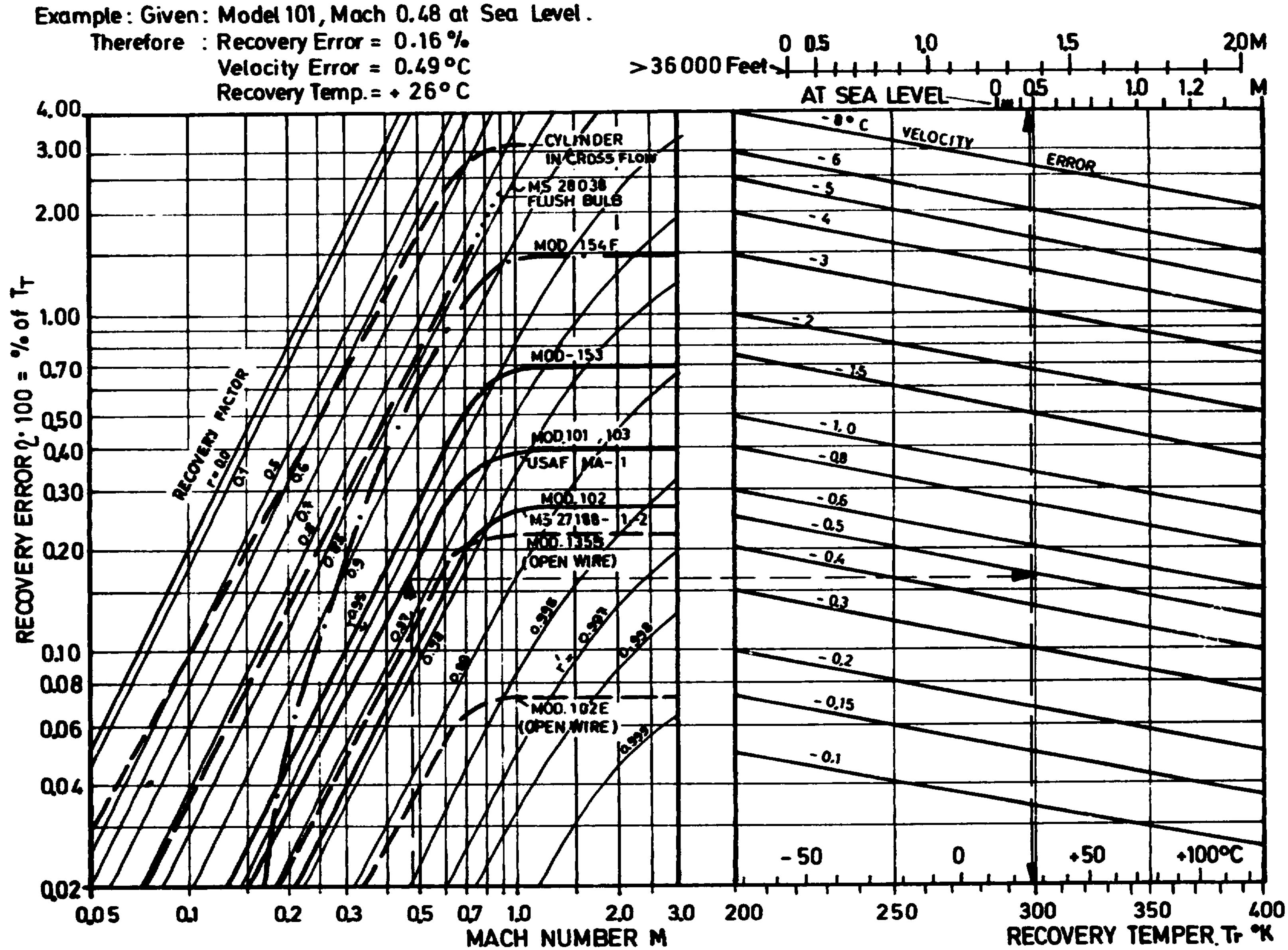


Fig.68 Recovery and velocity error as a function of Mach number or recovery temperature

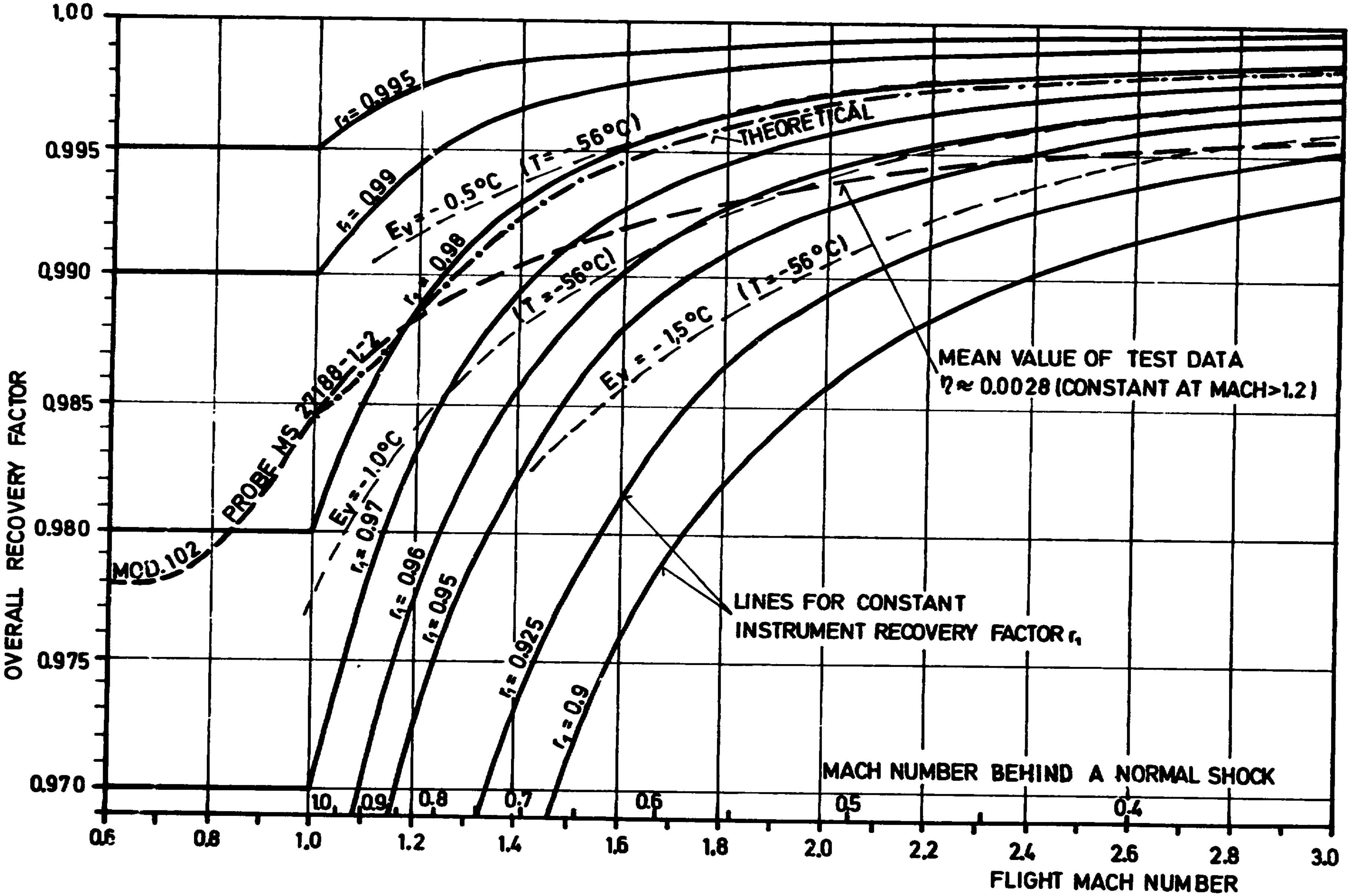


Fig.69 Variation of overall recovery factor with Mach number

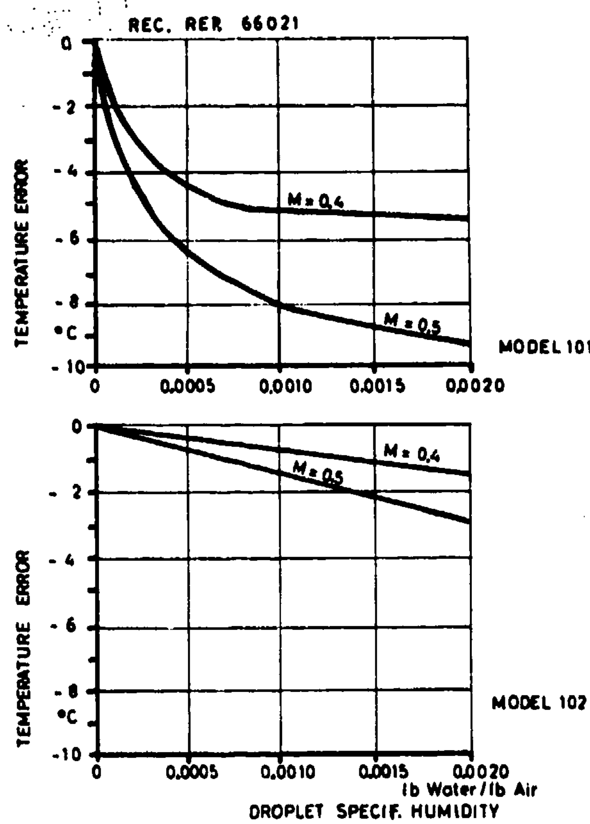


Fig.70 Typical errors due to water droplets

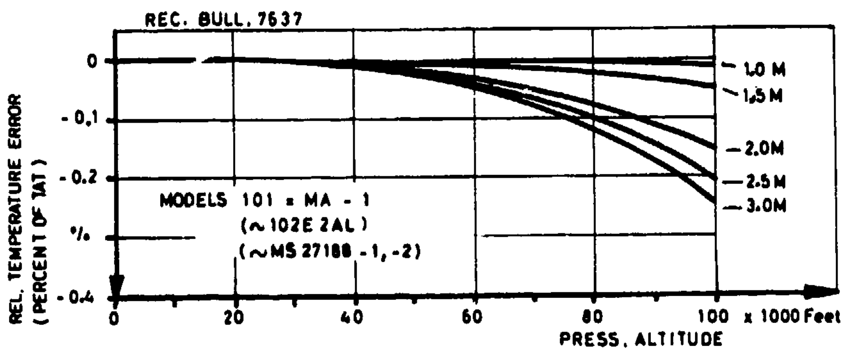


Fig.71 Typical radiation error of some TAT-probes

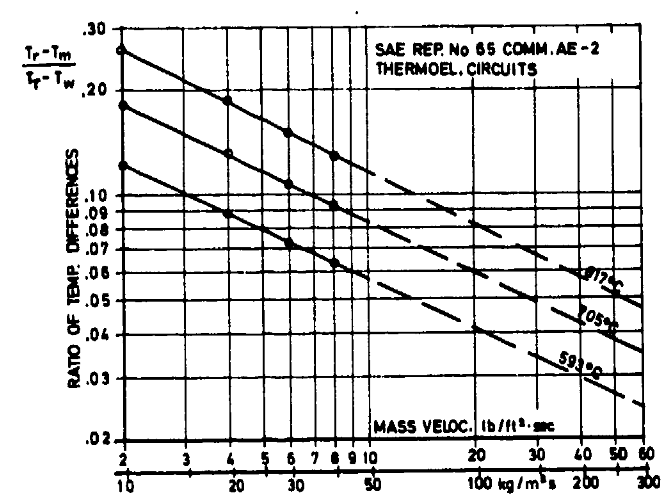
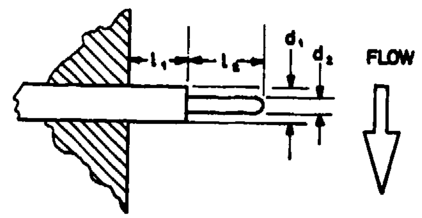


Fig.72 Radiation and conduction correction factors vs flow rate for a typical tailpipe temperature sensor with bare wire loop junction (16 Ga round wire)



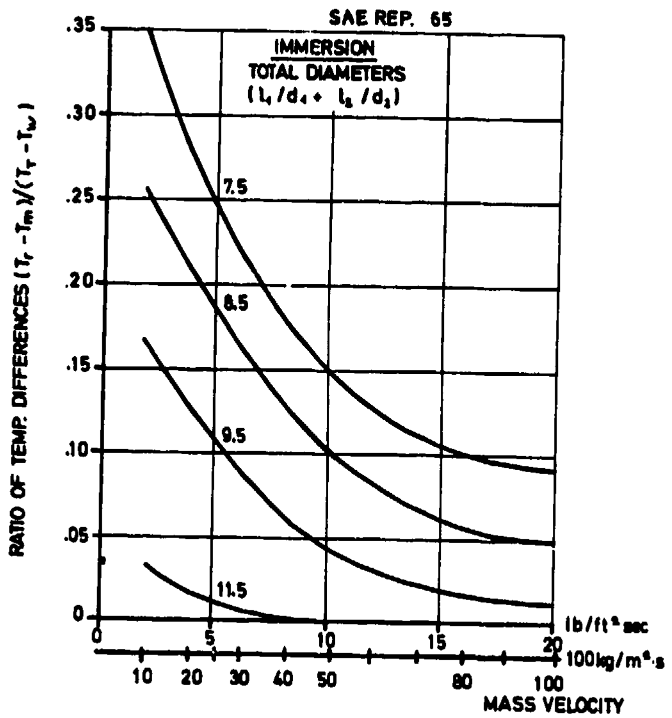


Fig.73 Conduction correction factor of EGT-probes vs. mass velocity

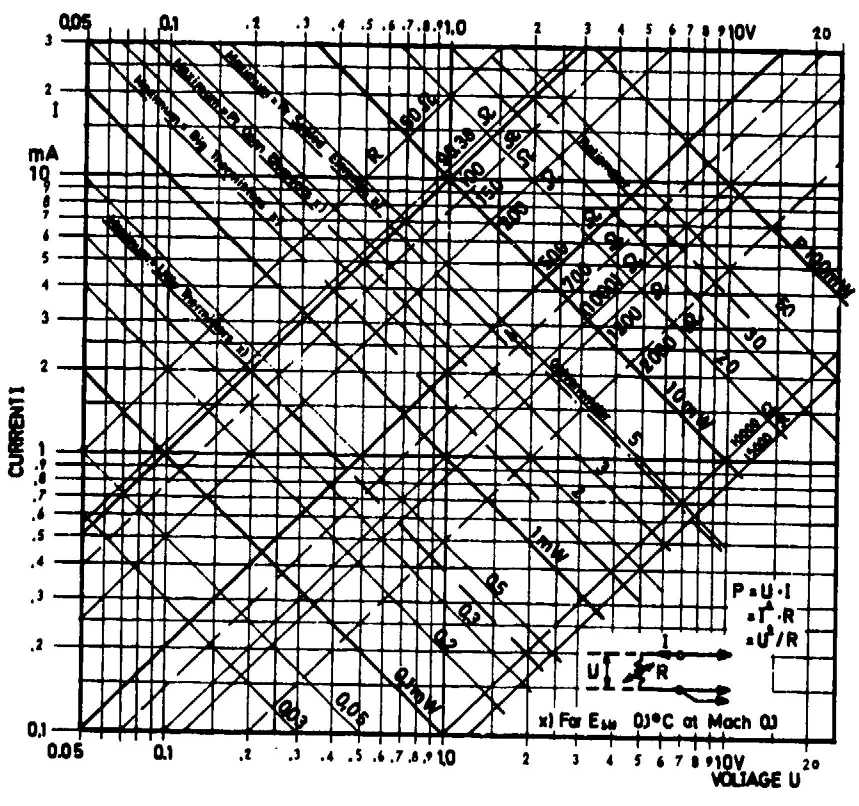


Fig.74 Self-heating power in temperature probes

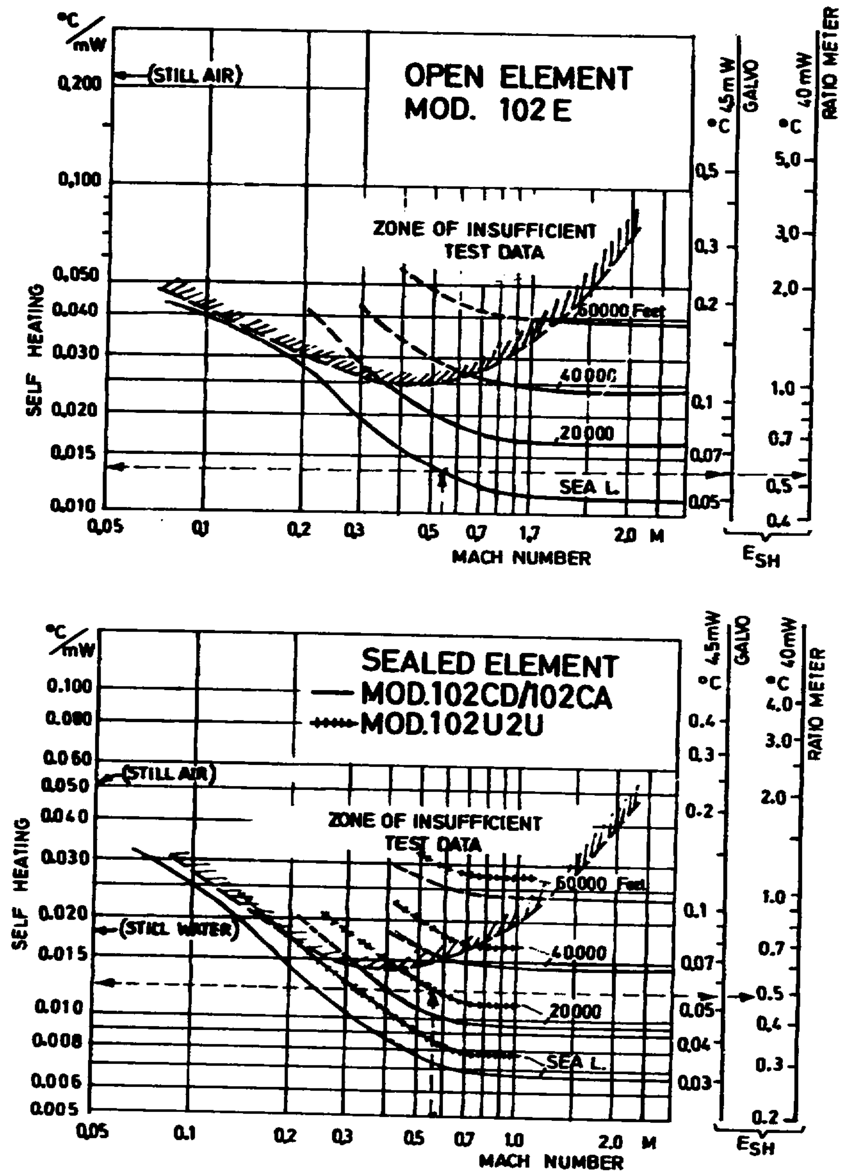


Fig.75(a,b) Self-heating of some elements vs. Mach number and altitude and resulting self-heating errors

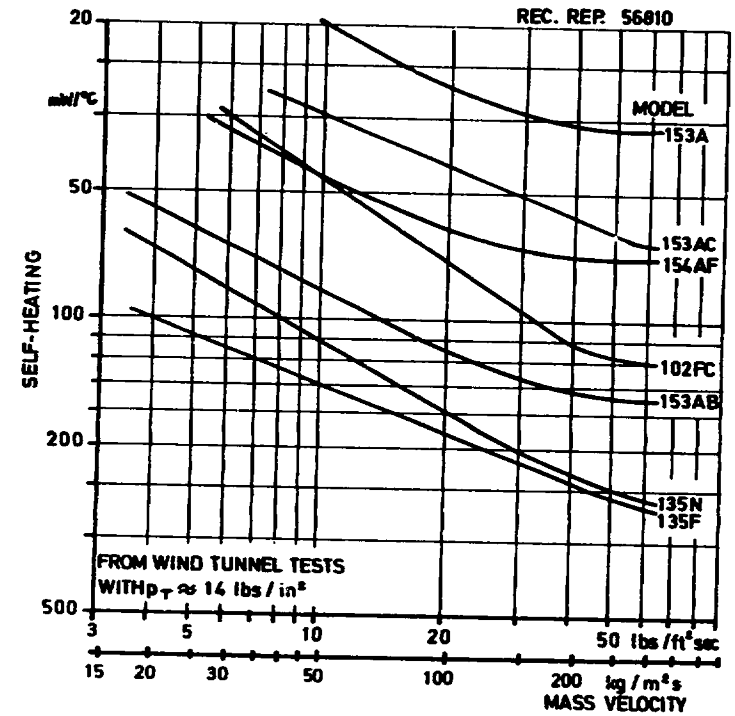


Fig.76 Self-heating of CIT-probes vs. mass velocity

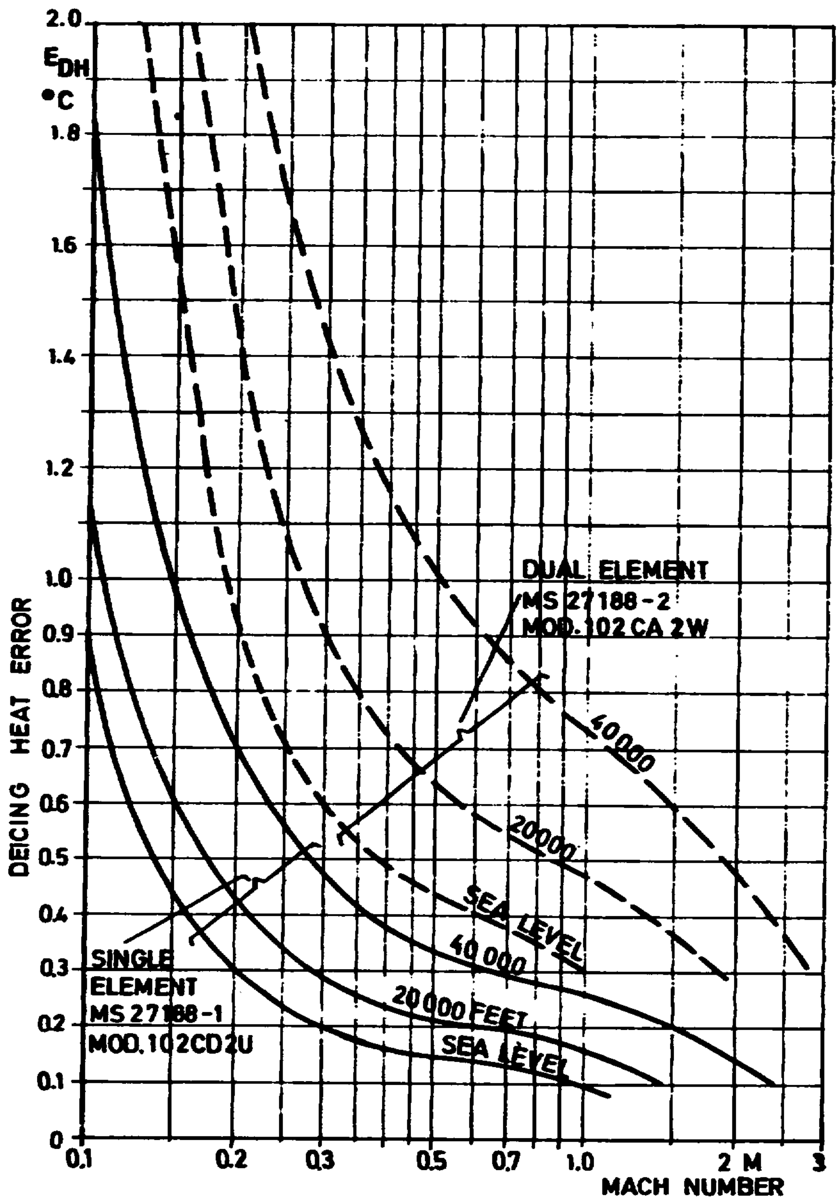


Fig.77 Deicing heat error (estimated)

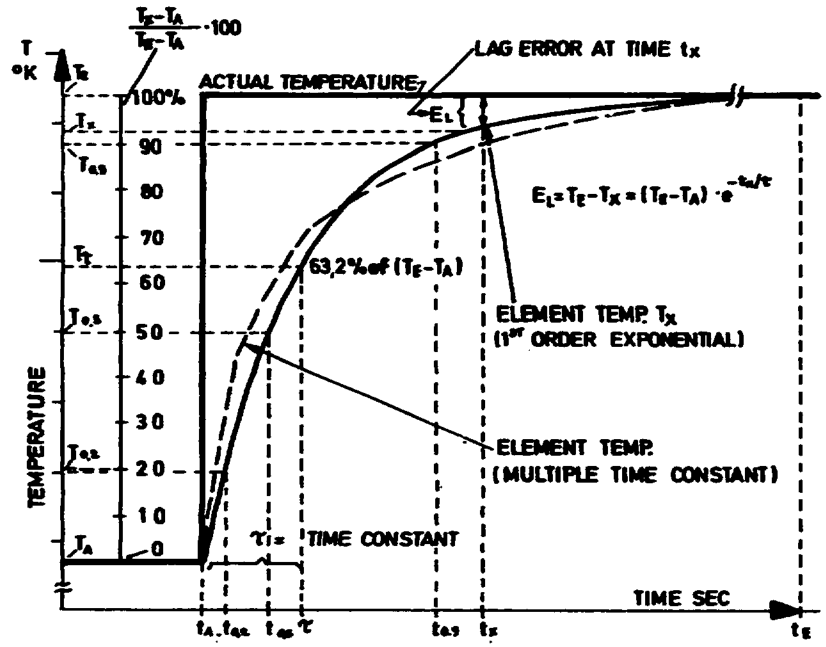


Fig.78 Actual and measured temperature at a step change

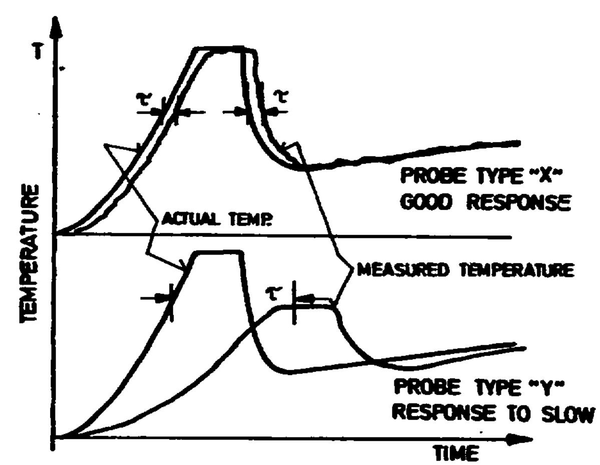


Fig.79 Different responses to a temperature change

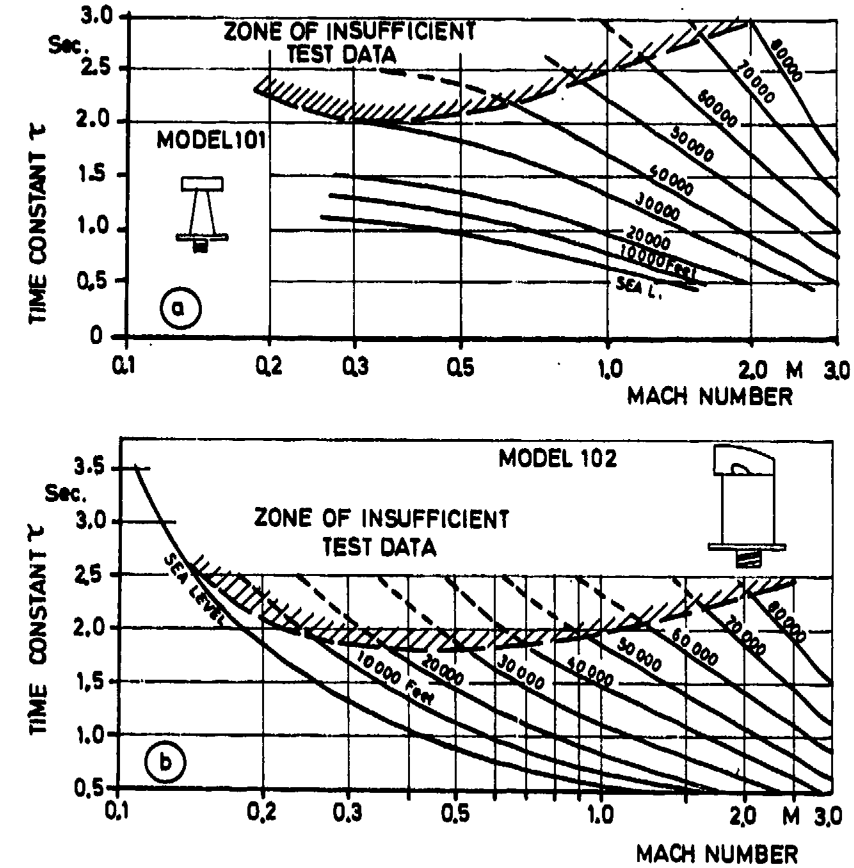


Fig.80(a,b) Time constant of some probes

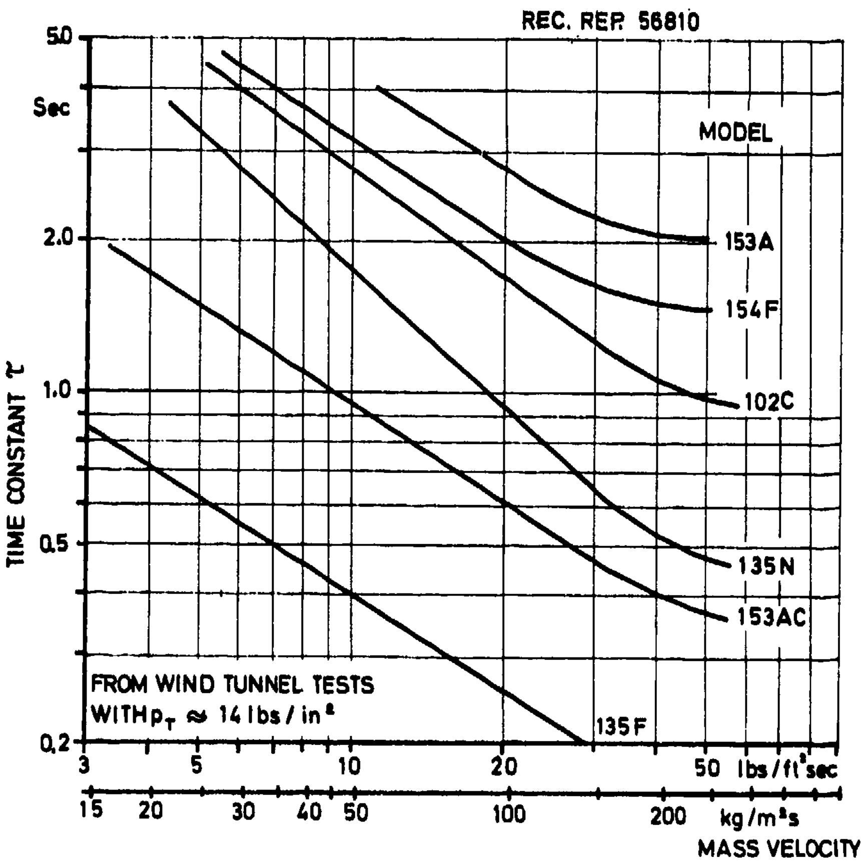


Fig.81 Typical time constants of some CIT-probes vs. mass velocity

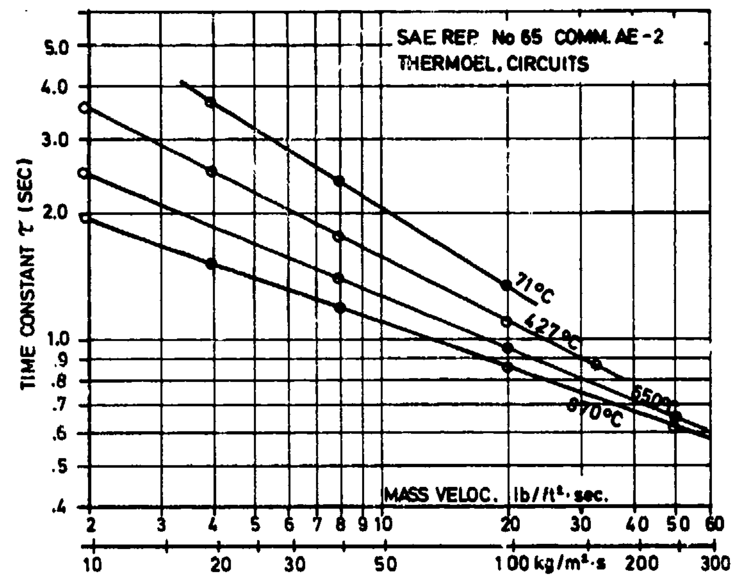
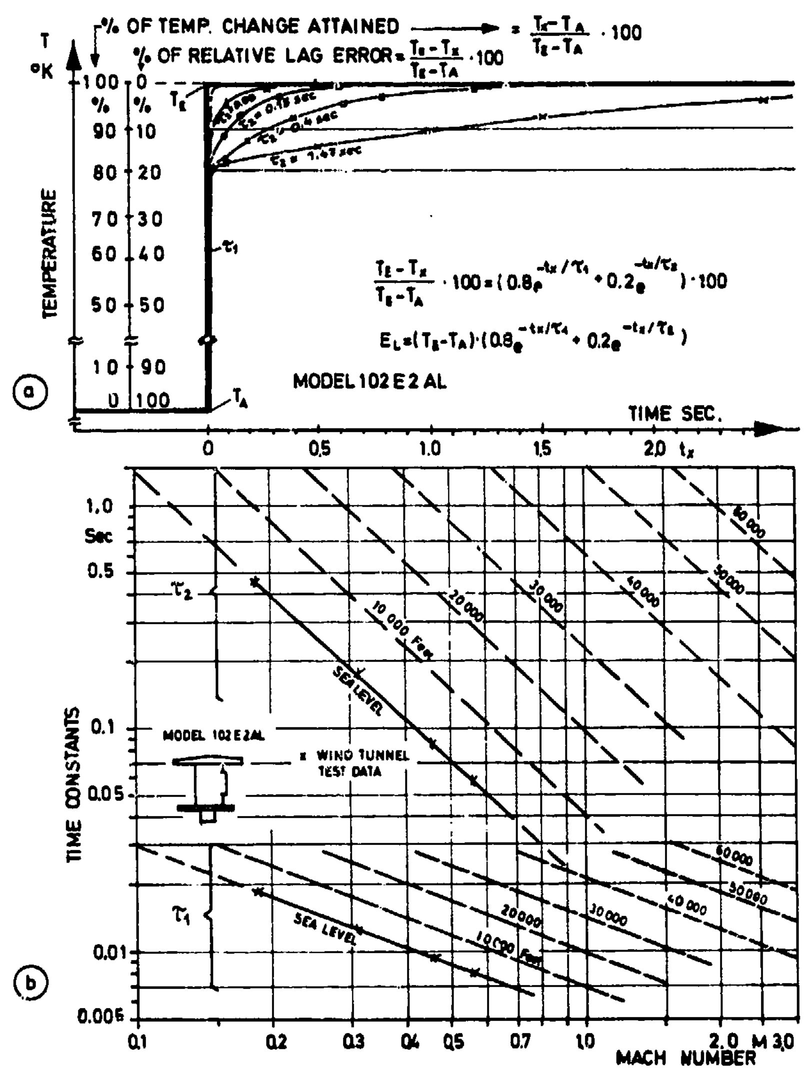


Fig.82 Time constant vs. flow rate for a typical tailpipe temperature sensor with bare wire loop junction (16 Ga. round wire)



(a) response to a step change
(b) time constants vs. Mach number and altitude

Fig.83(a,b) Probe with open wire element

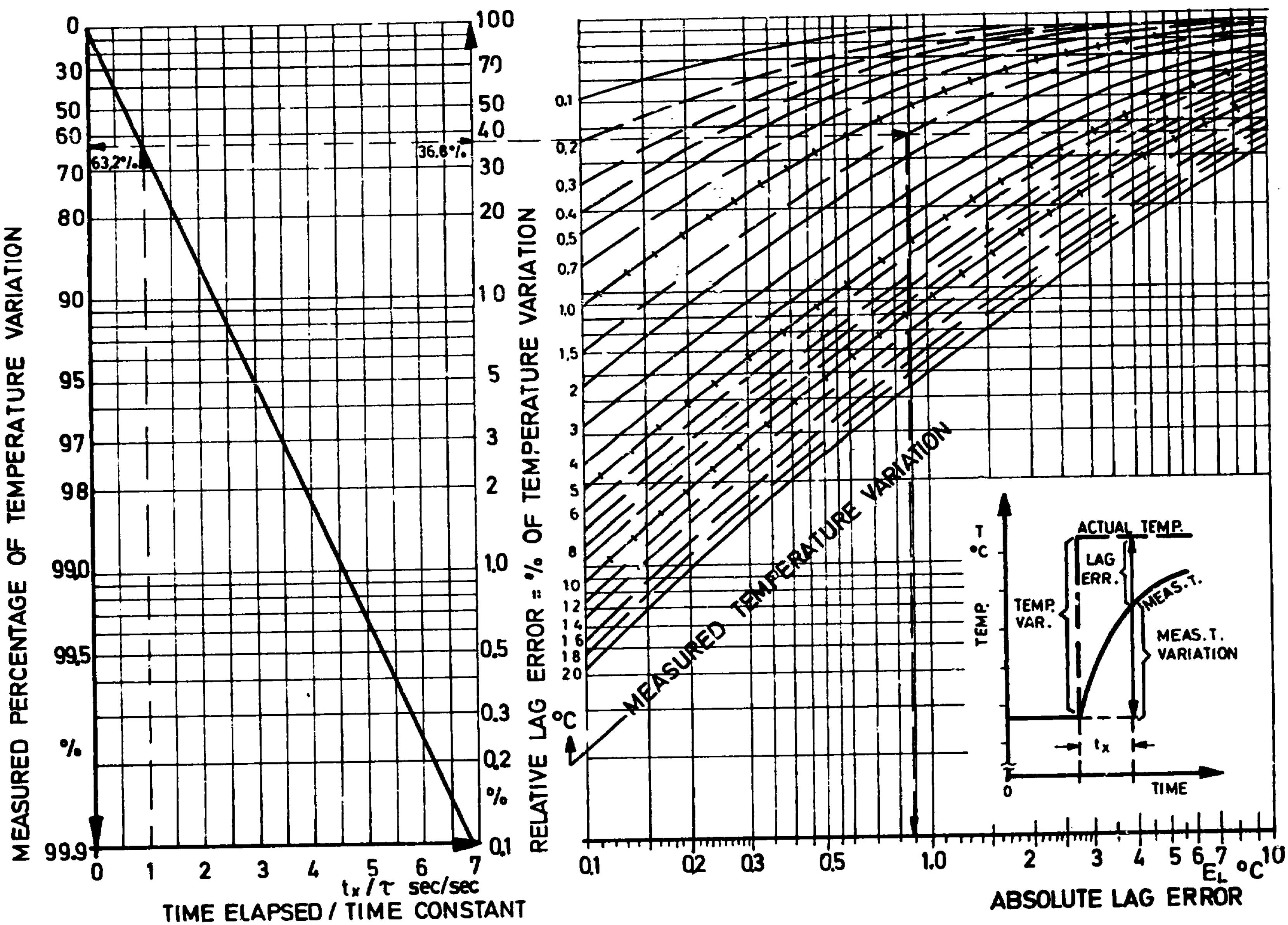


Fig.84 Temperature lag error as a function of time elapsed and measured variation in temperature

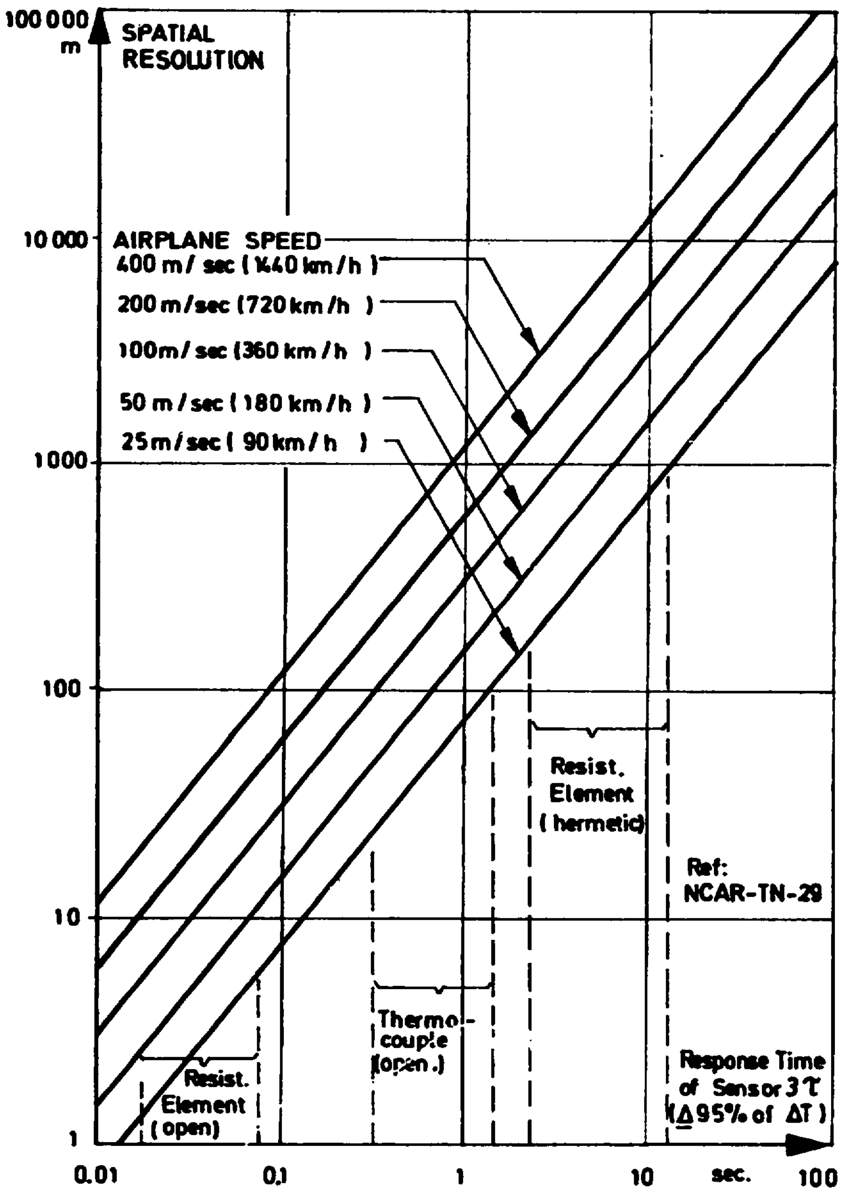


Fig.85 Spatial resolution of measurement as a function of sensor response

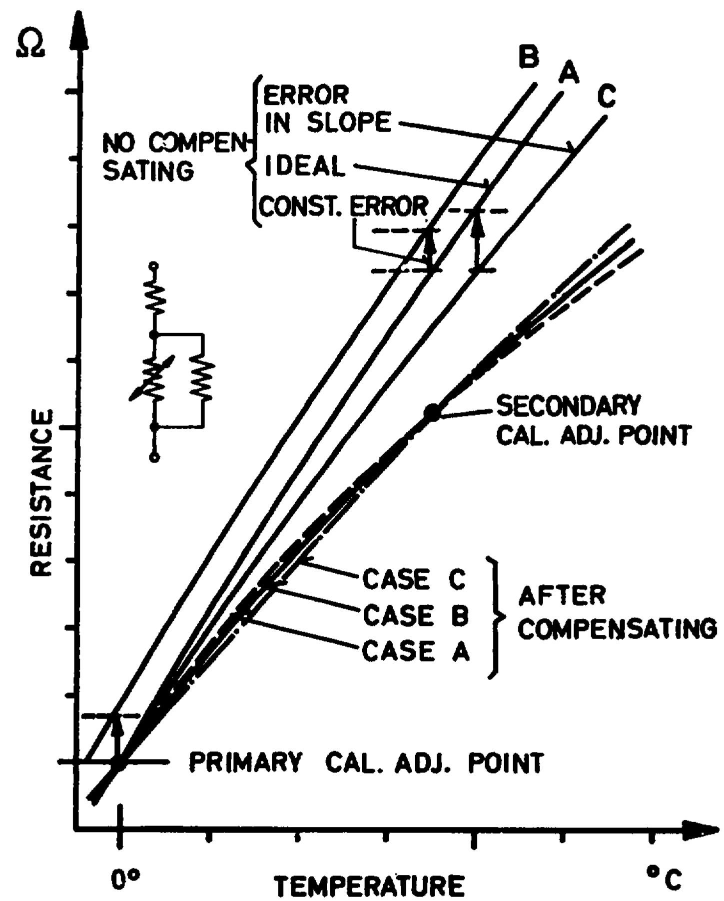


Fig.86 Effect of precision calibration interchangeability (PCI) trimming

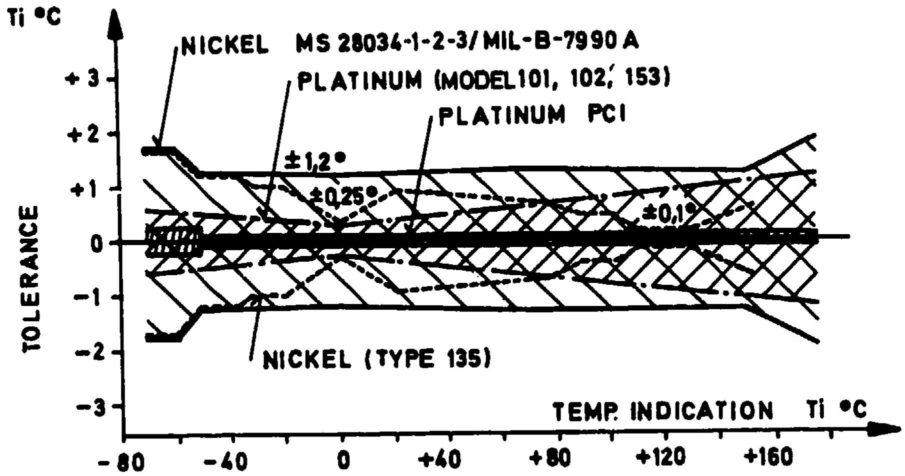


Fig.87(a) Typical tolerances (calibration error) of resistance probes.
Note: PCI = precision calibration interchangeability

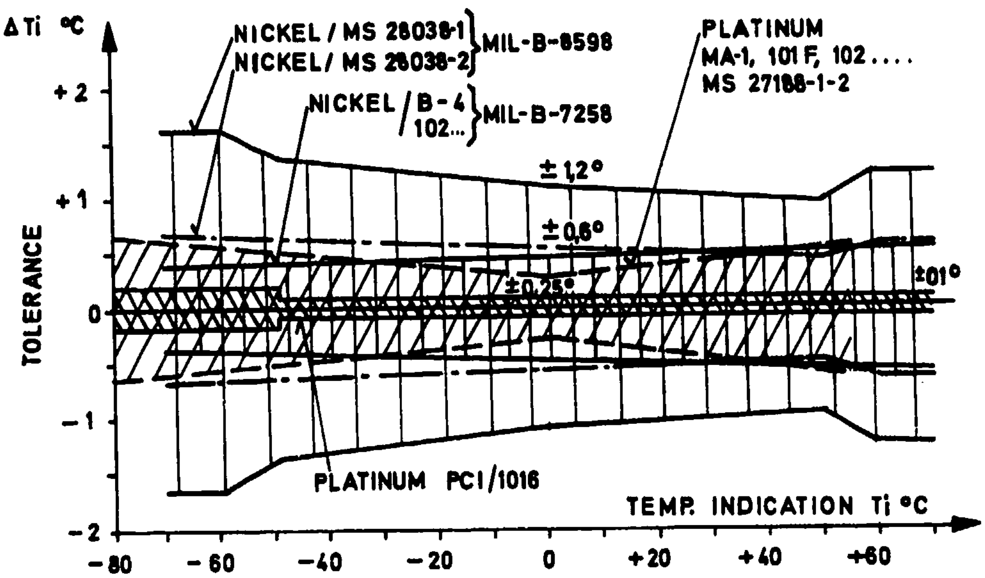


Fig.87(b) Typical tolerances (calibration error) of resistance probes

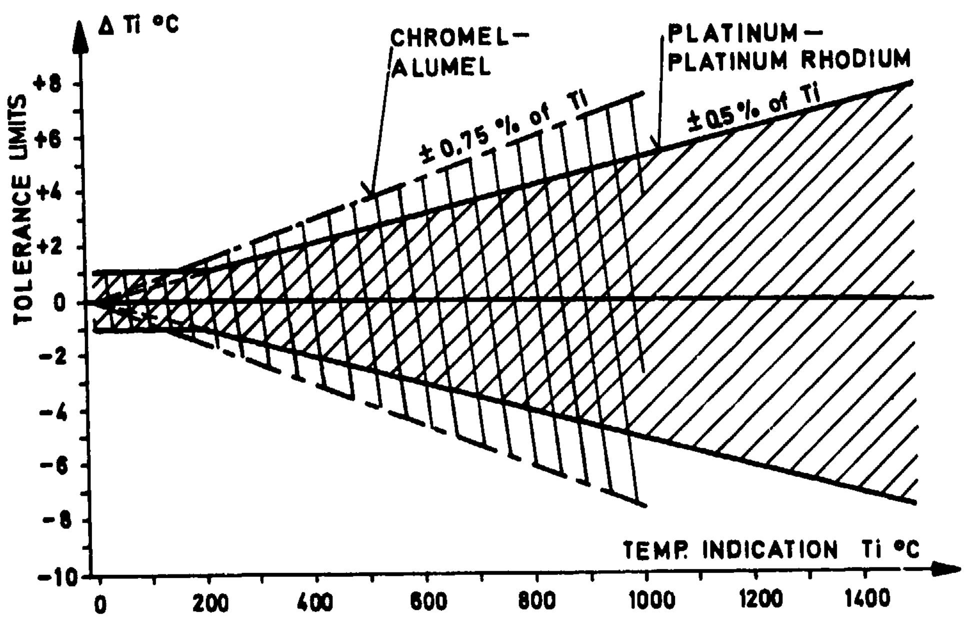


Fig.88 Typical tolerances (calibration error) of thermocouples

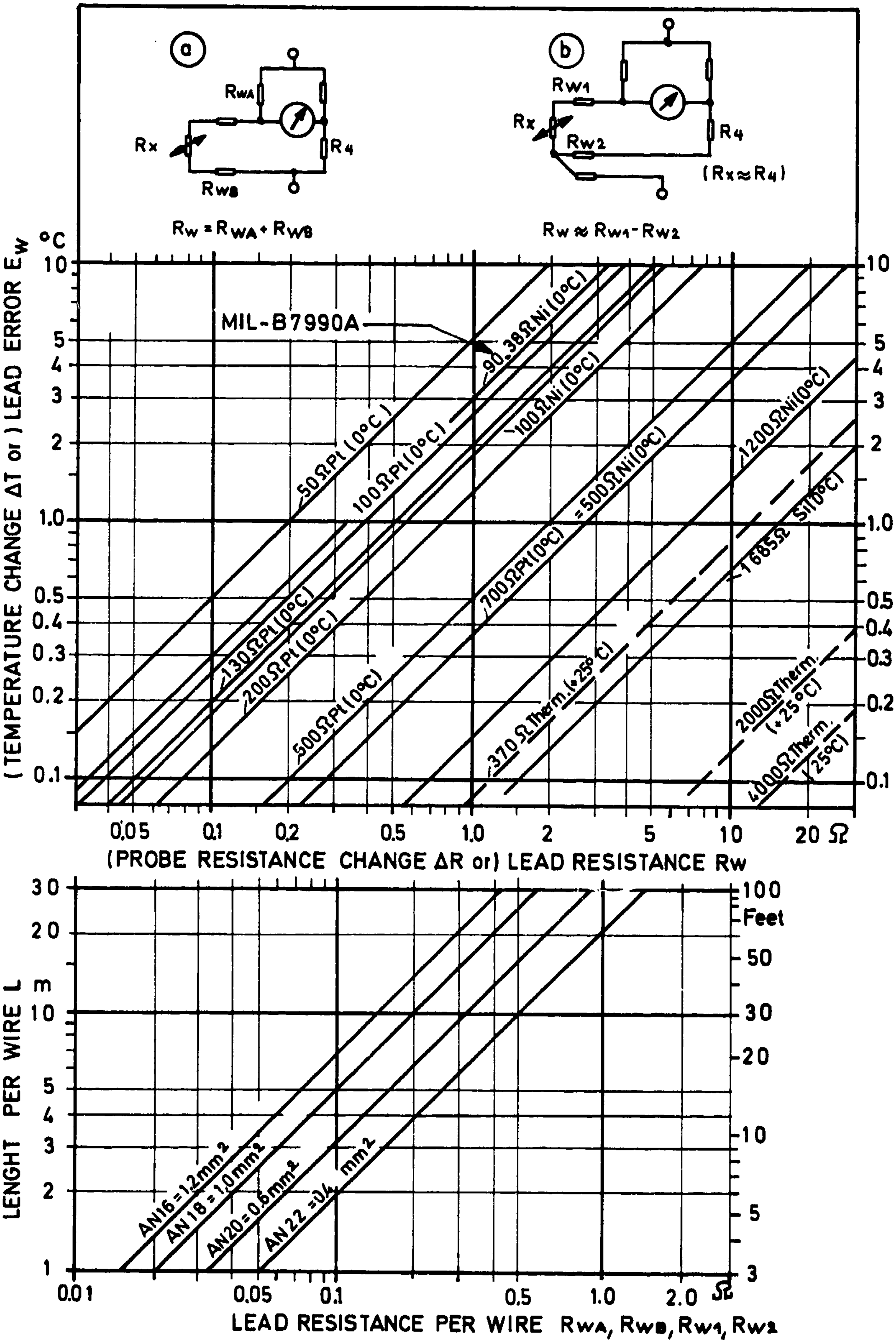


Fig.89 Electric lead error of temperature probes

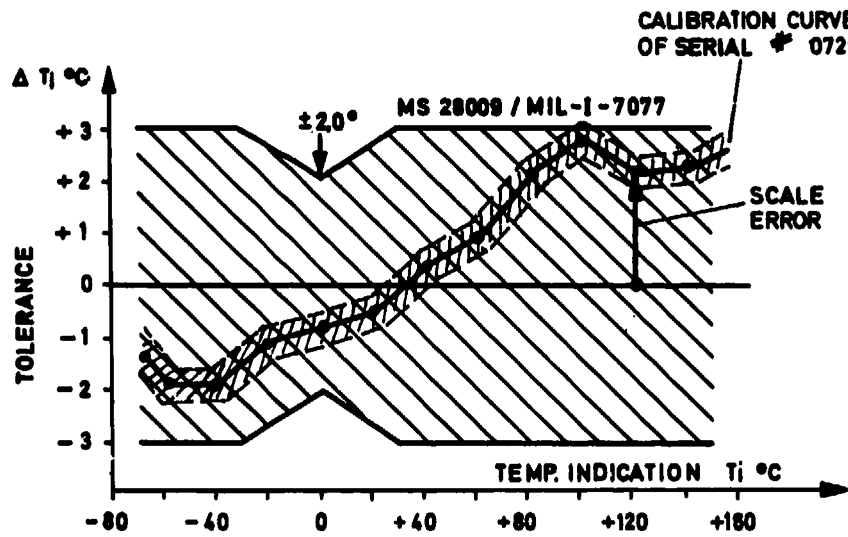


Fig.90(a) Typical tolerances of resistance thermometer indicator at room temperature (ratiometer without probe)

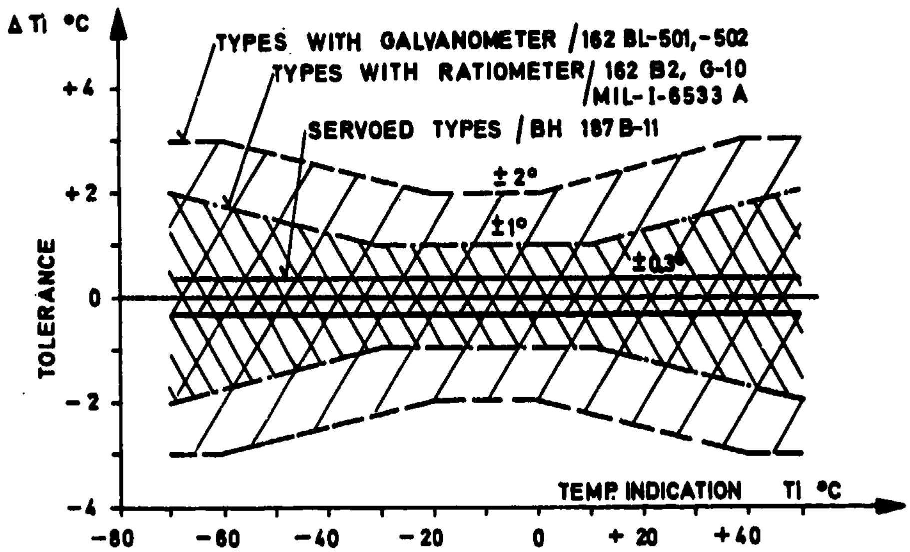


Fig.90(b) Typical tolerances (scale error) of some indicators (without probe)

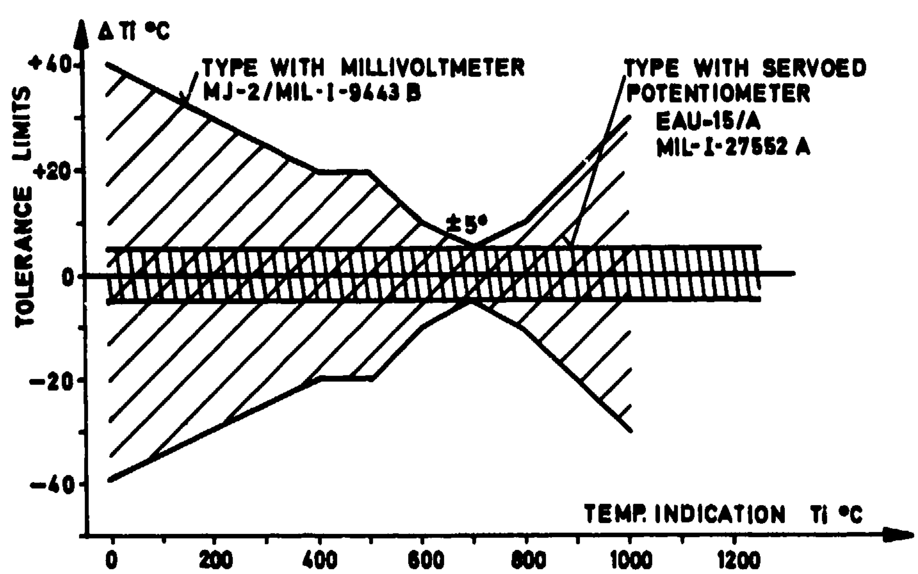


Fig.91 Typical tolerances (instrument error) of indicators (without probe)

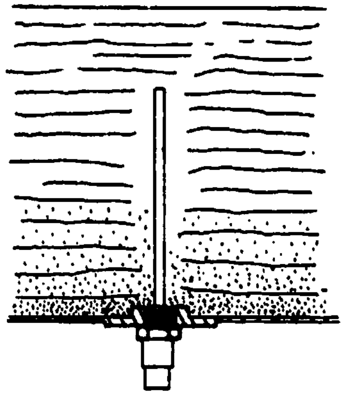


Fig.92 Probe installation in tank

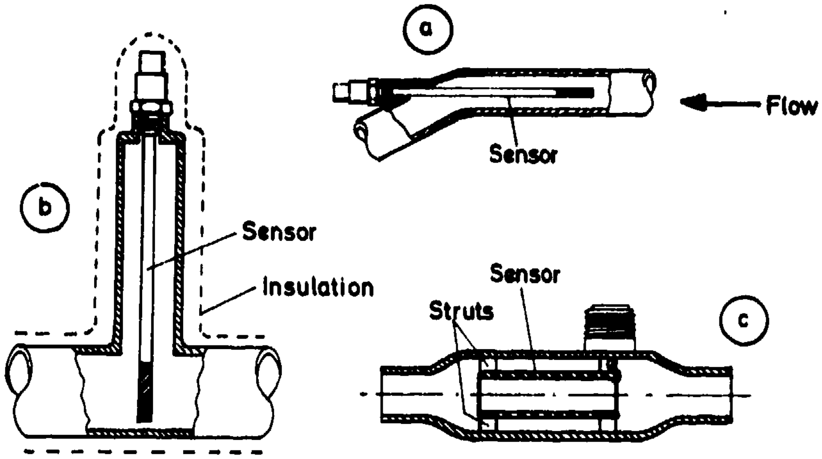
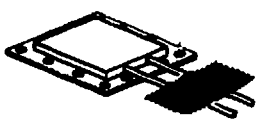


Fig.93 Typical probe installations in tubing

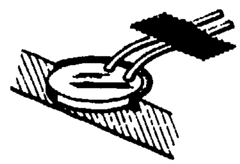


a

Sensor spotwelded to surface.

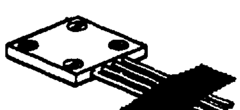
d

Sensor cemented to the surface to measure sub-surface temperature.



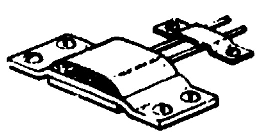
g

Sensor mounted in spot faced cavity. Lead exit from the top at the sensor.



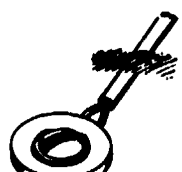
b

Sensor clamped with screws.



e

Sensor clamped with bracket.



h

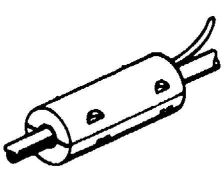
Mount with Single Screw

c

Sensor cemented to measure temperature at the surface.

f

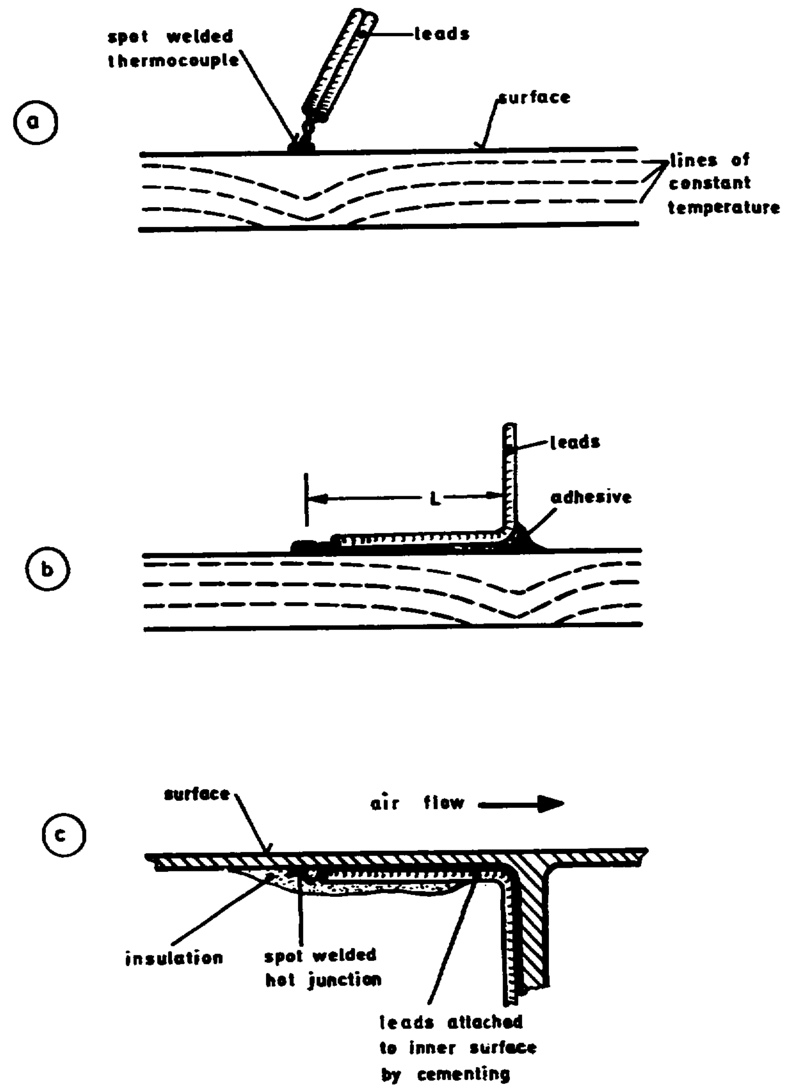
High temperature mounting. Screen over sensor with high temperature cement bonding through the screen.



i

Sensor clamped around pipe.

Fig.94 Installation of surface temperature sensors



(a) surface measurement with excessive conduction error
(b) good surface measurement
(c) subsurface measurement

Fig.95 Surface and subsurface temperature measurement

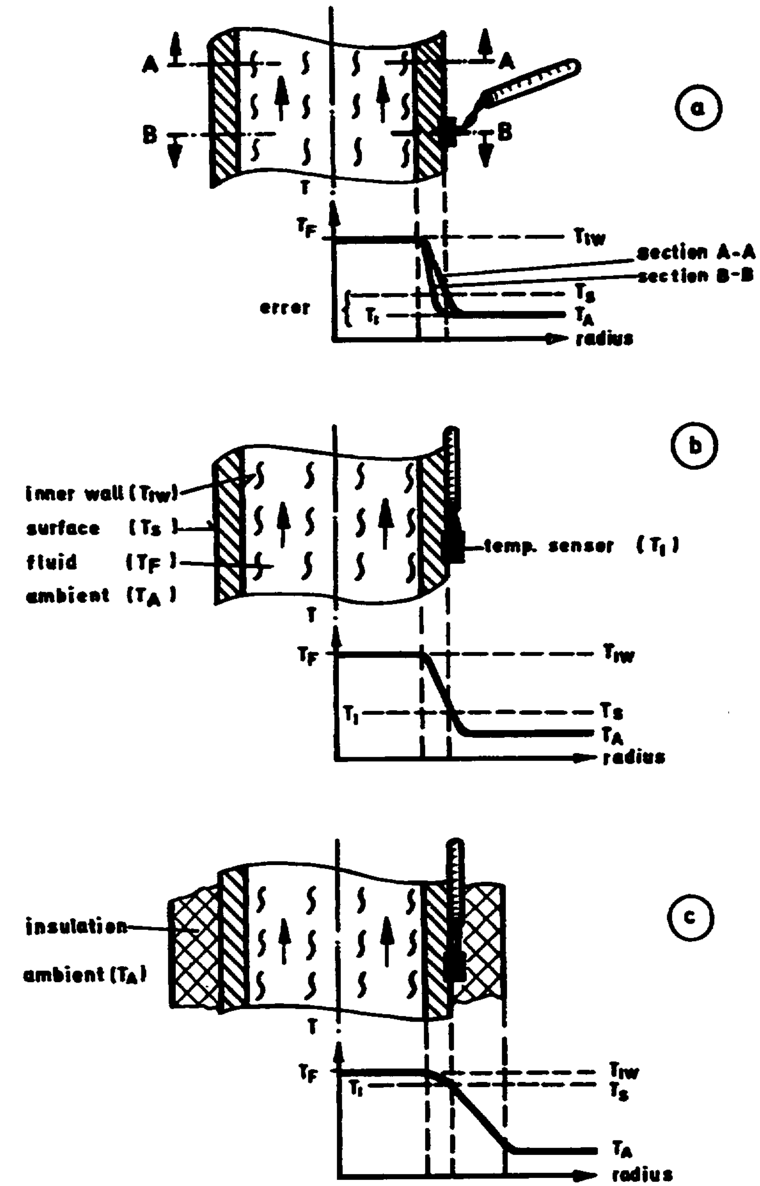


Fig.96 Relationship of surface temperature to subsurface and ambient temperature

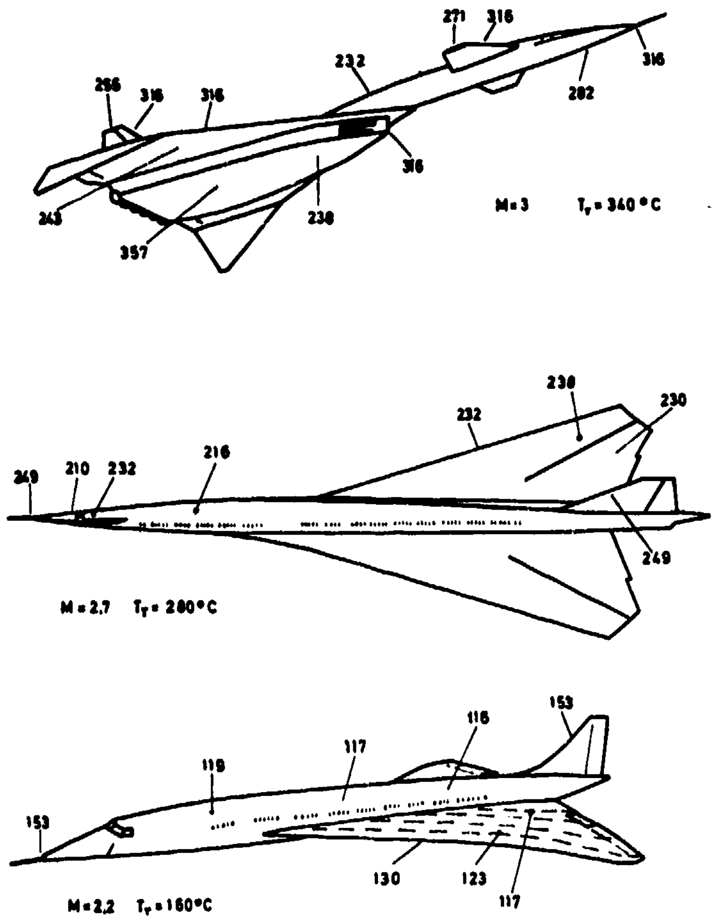


Fig.97 Skin temperatures of supersonic aircraft (°C)

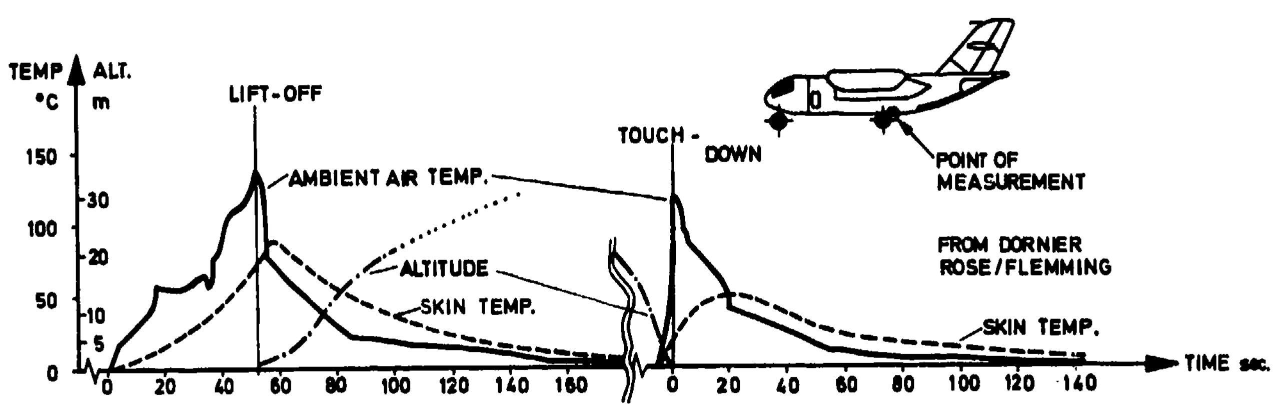


Fig.98(a) Skin temperature at vertical take-off and landing

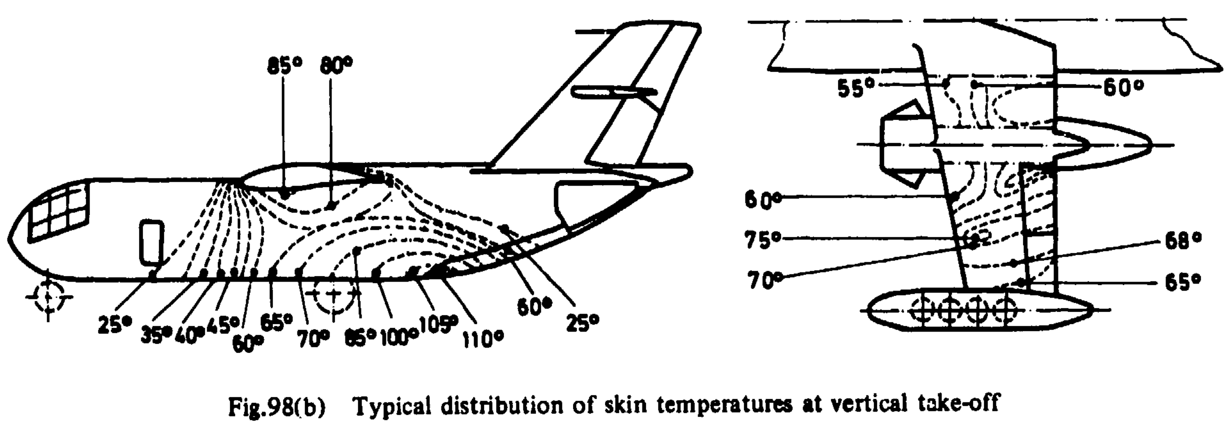


Fig.98(b) Typical distribution of skin temperatures at vertical take-off

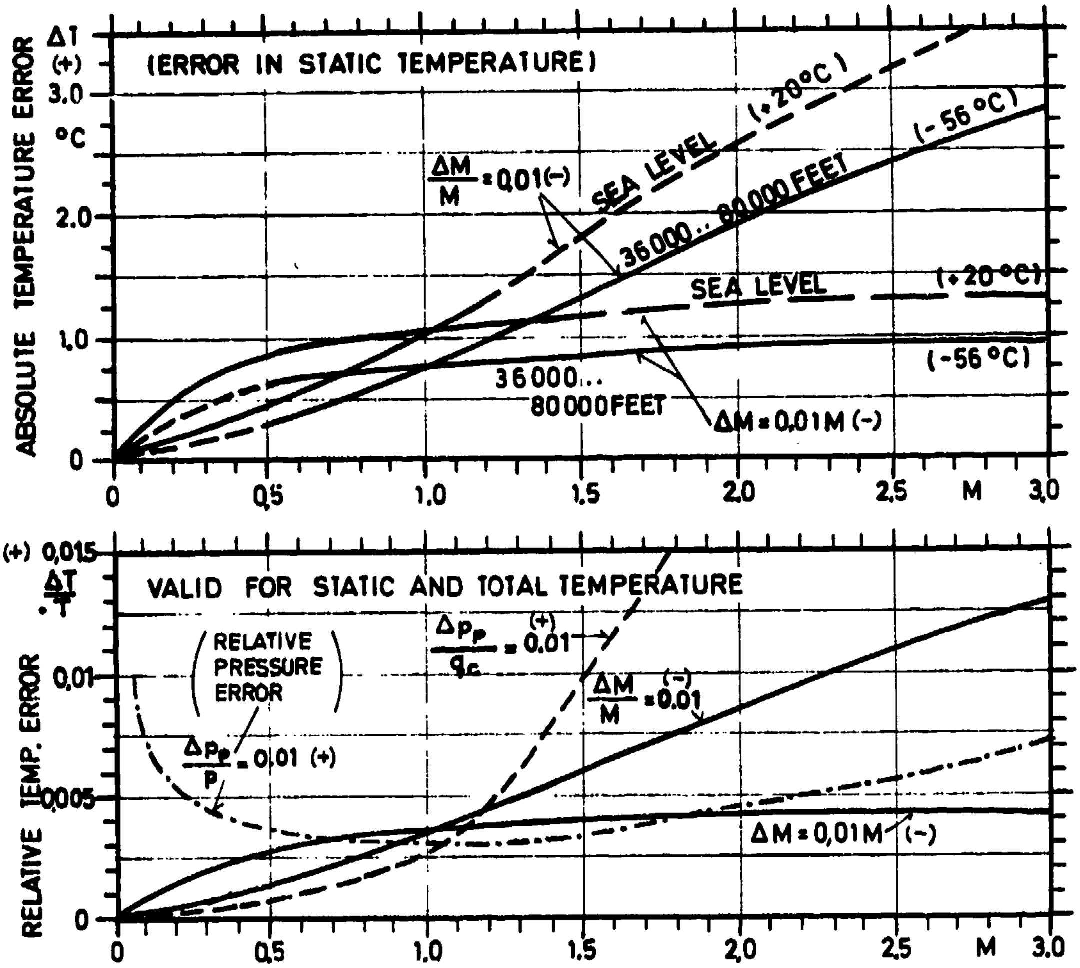


Fig.99 Absolute and relative error in static temperature vs. Mach number as functions of given values of absolute and relative errors in Mach number

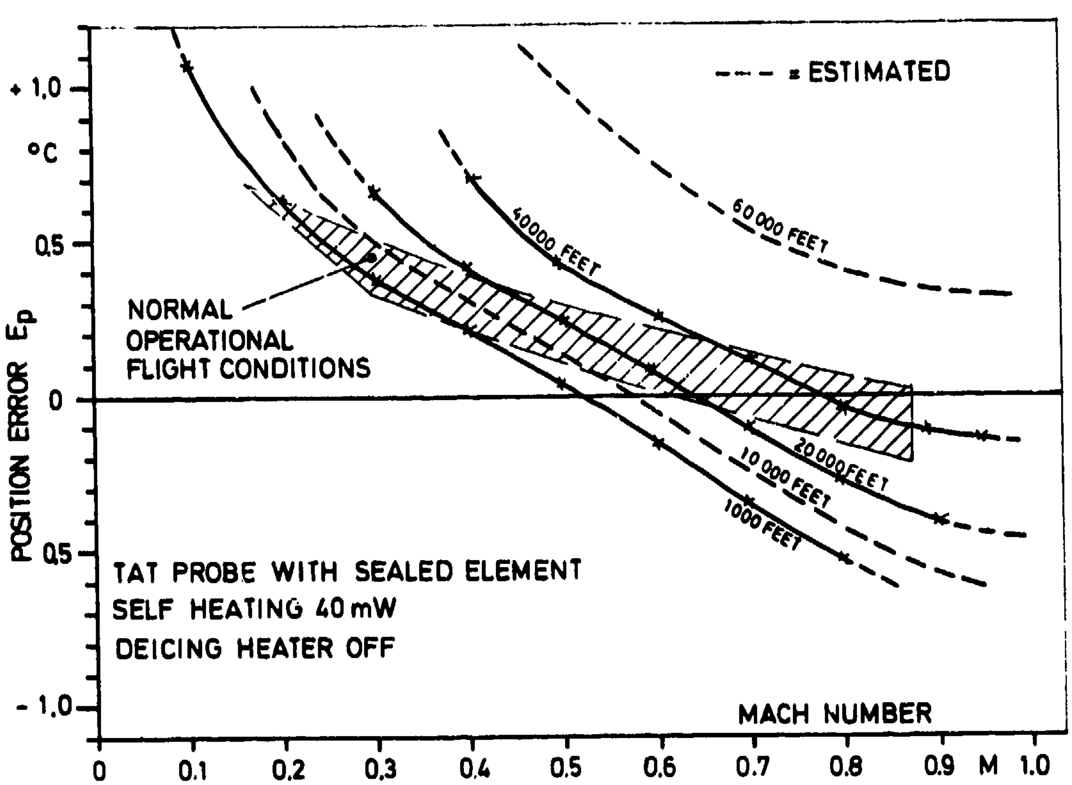


Fig.100 Position error of a TAT-probe with ratiometer (theoretical values)

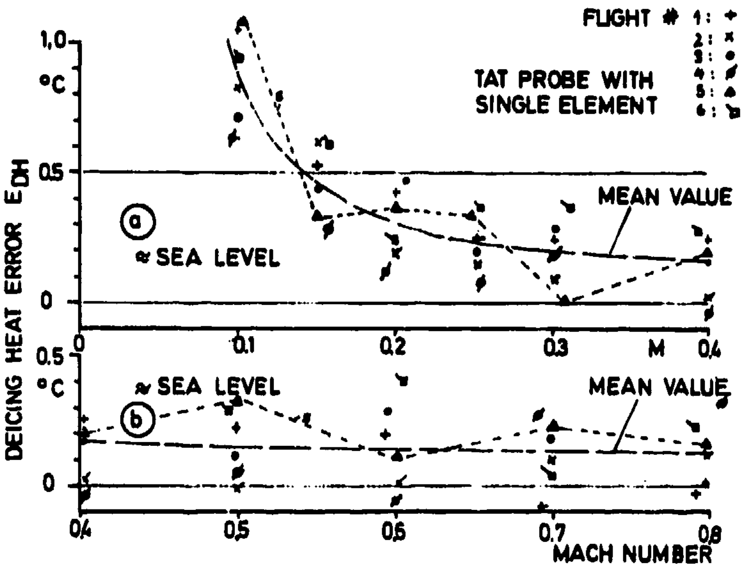
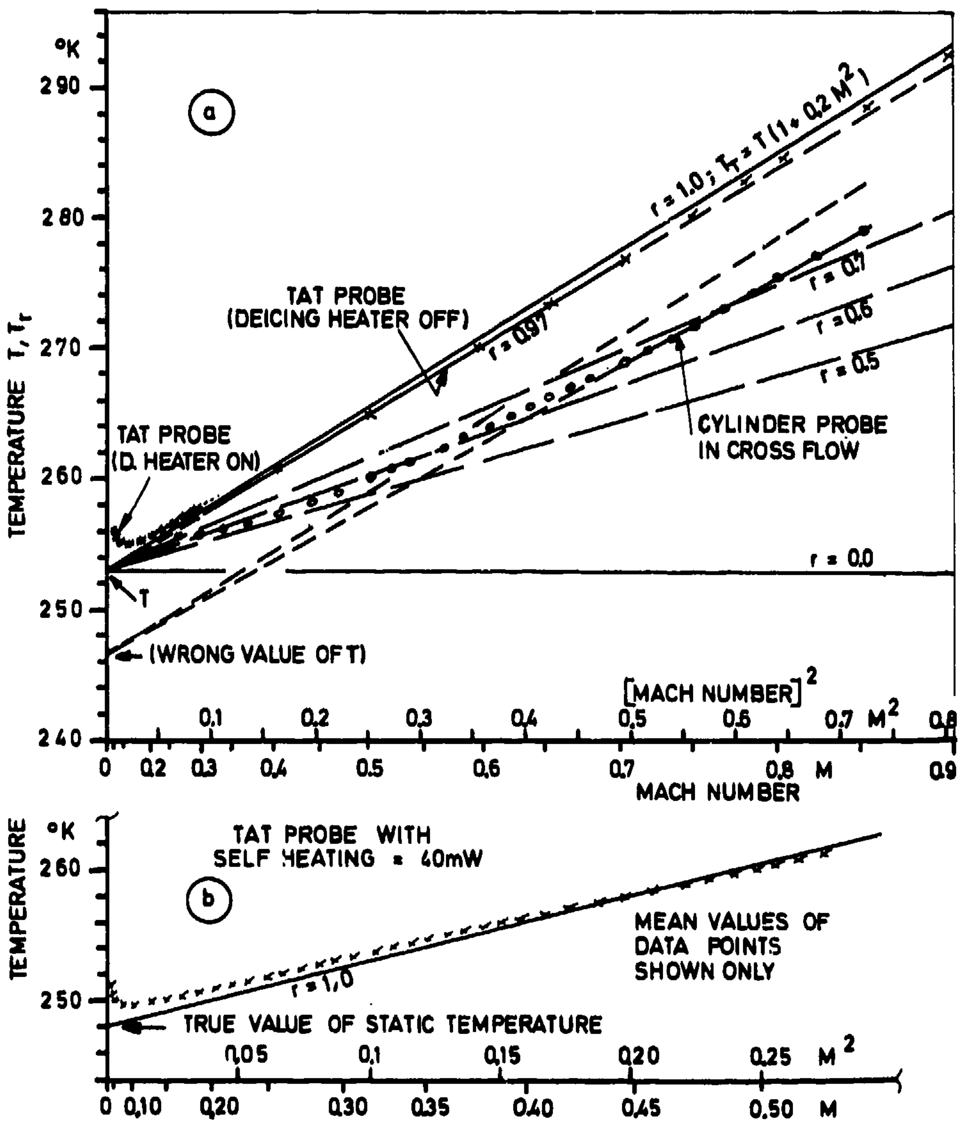
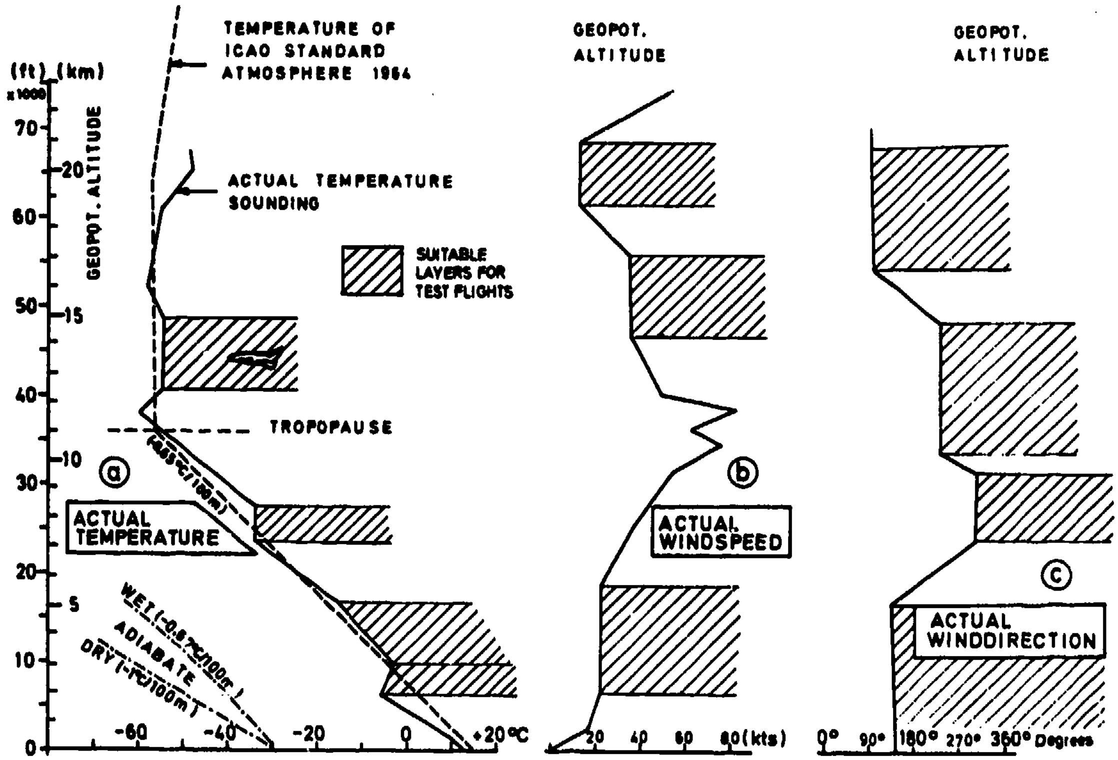


Fig.101 Deicing heat error of a TAT-probe (typical test results)



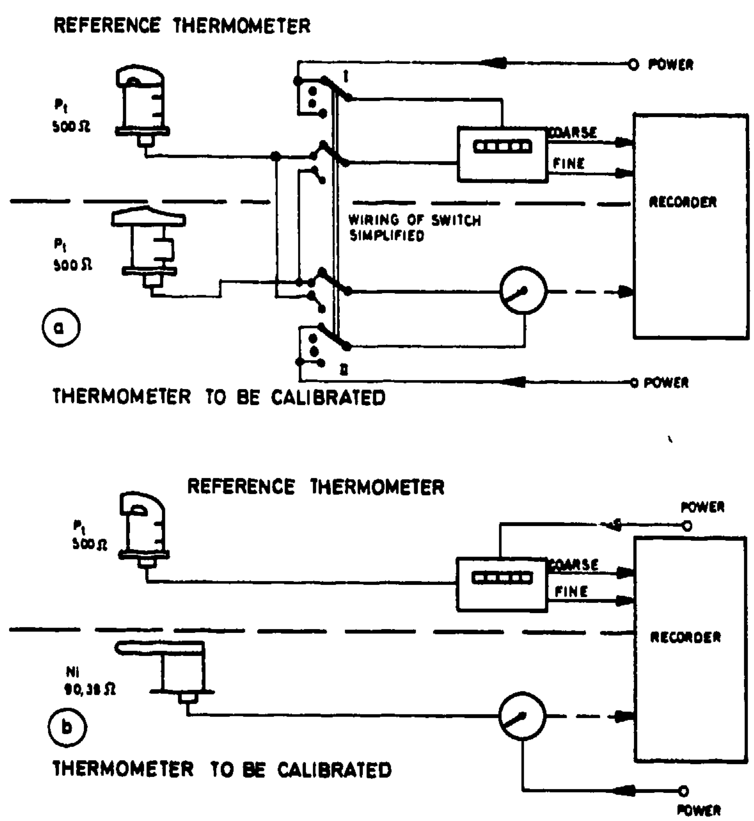
(a) measurement without self-heating
(b) measurement with self-heating

Fig.102 Evaluation of the recovery factor



(a) temperature
(b) windspeed
(c) winddirection

Fig.103 Suitable meteorological conditions for temperature calibration flights



(a) same
(b) different } probe resistance and/or characteristics

Fig.104 Connection of reference thermometer and thermometer to be calibrated

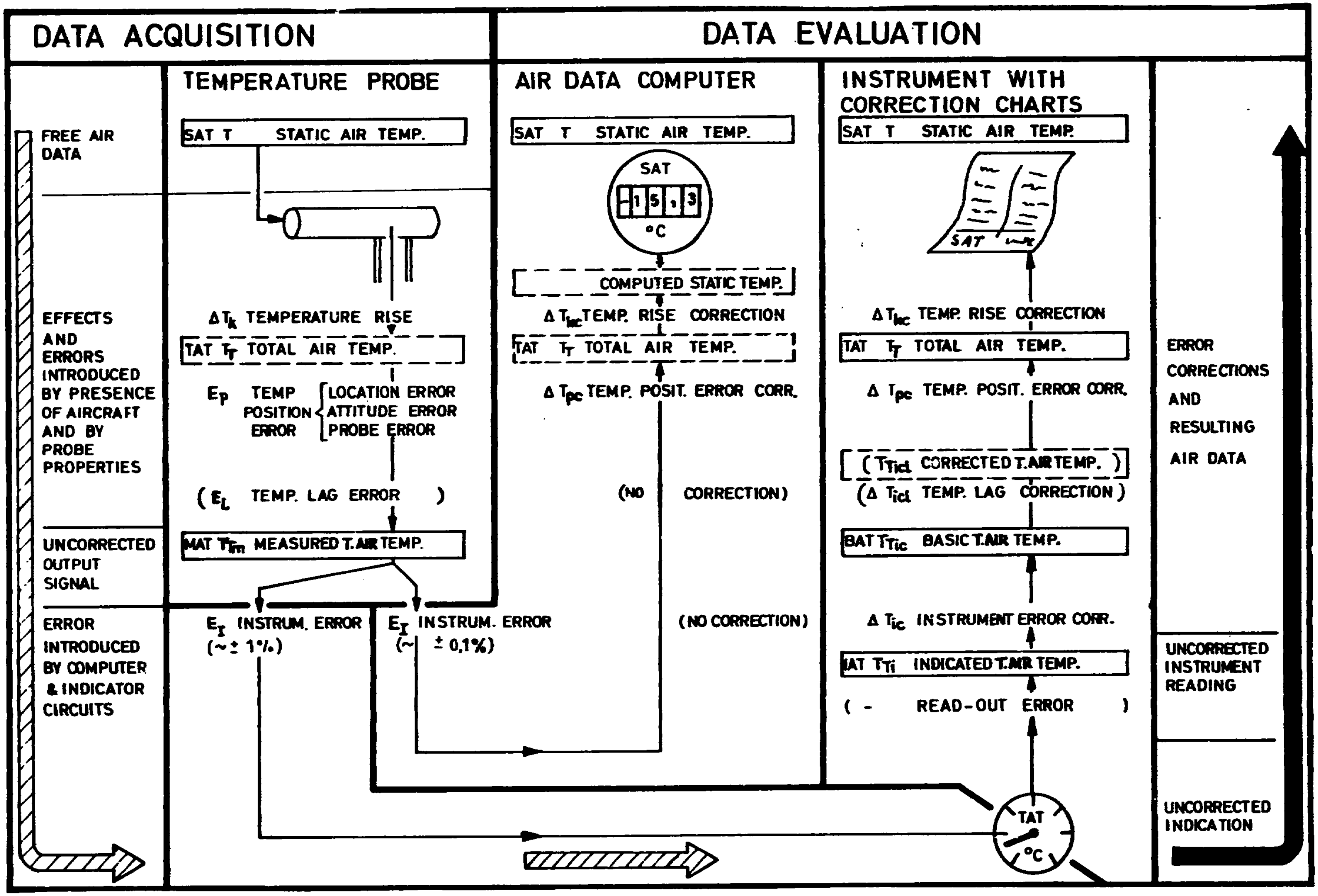


Fig.105 Data acquisition and evaluation

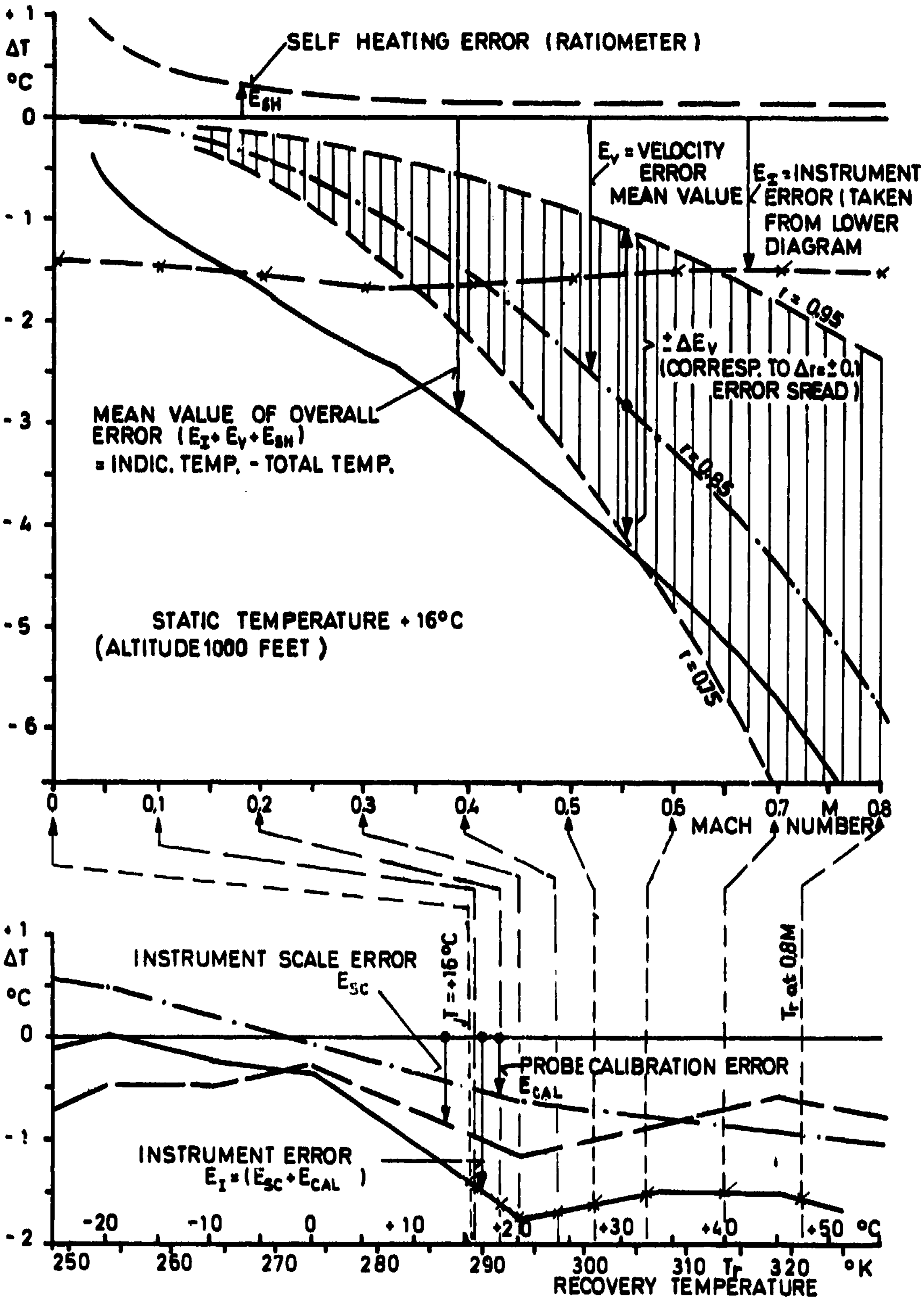


Fig.106 Temperature error components of simple probe (120Ω, Ni) with ratiometer indicator

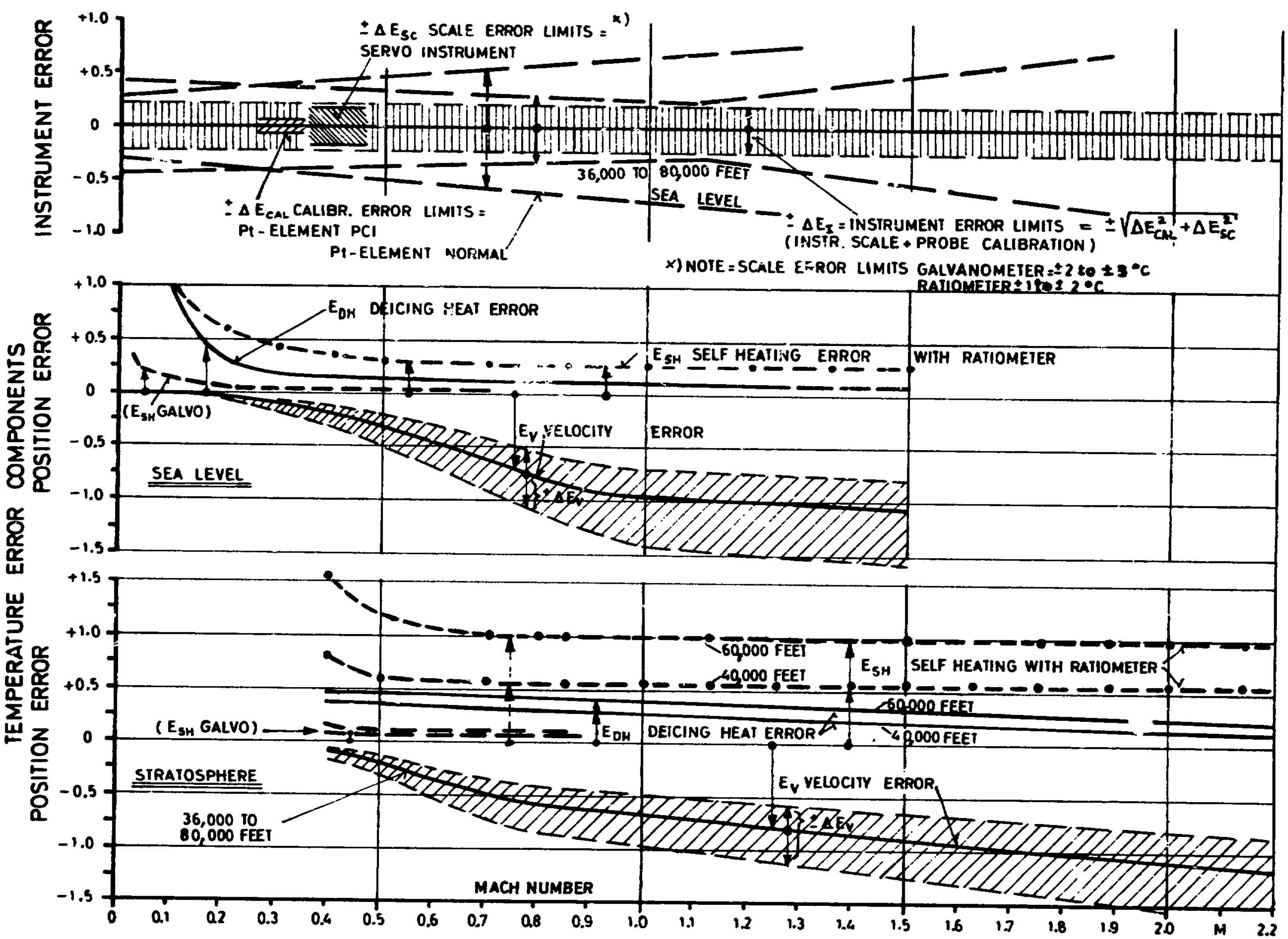


Fig.107 Temperature error components of a TAT-probe (500Ω, PCI) with servoed indicator

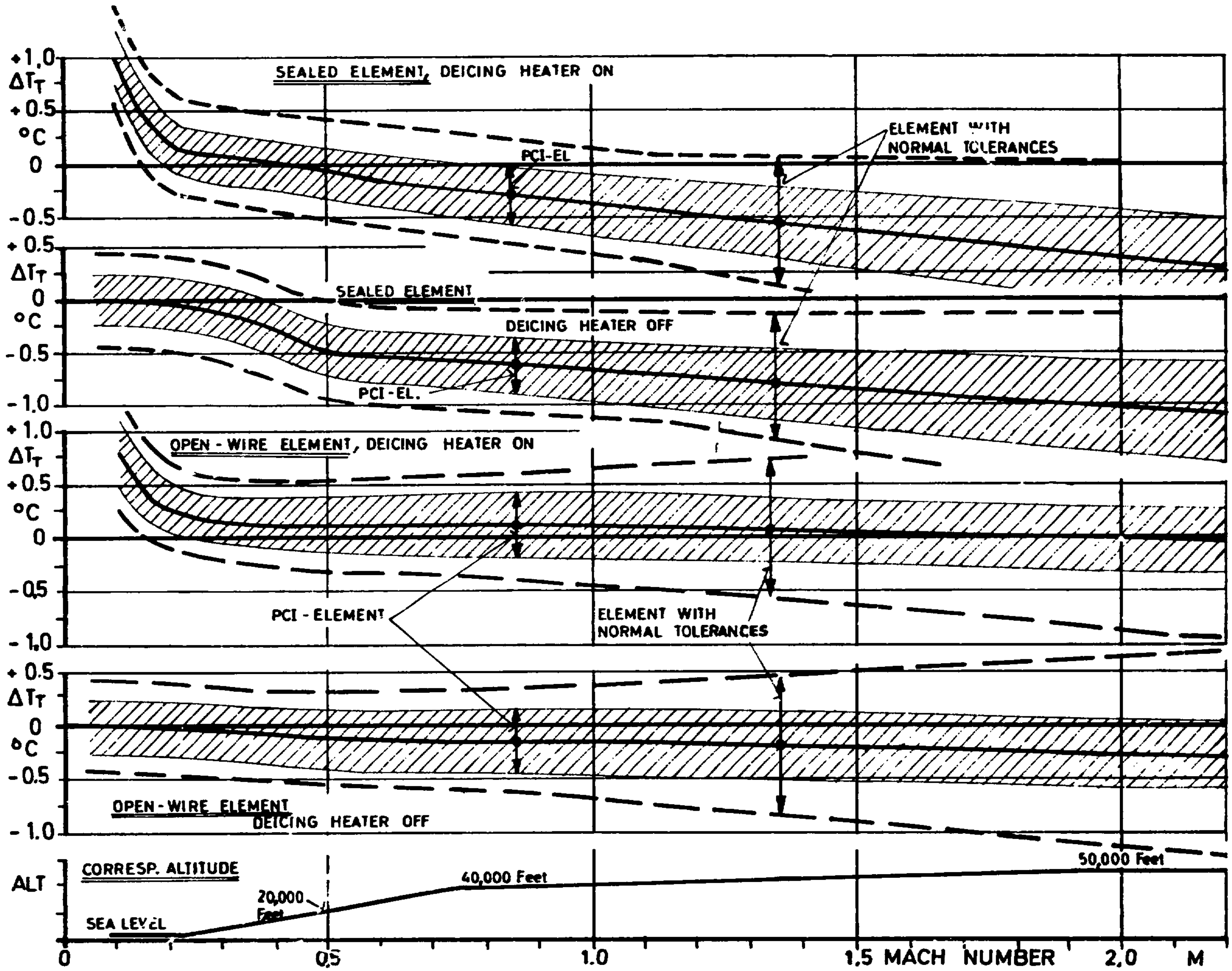


Fig.108 Overall temperature error of a TAT-probe (sealed/open-wire element) with servo-indicator

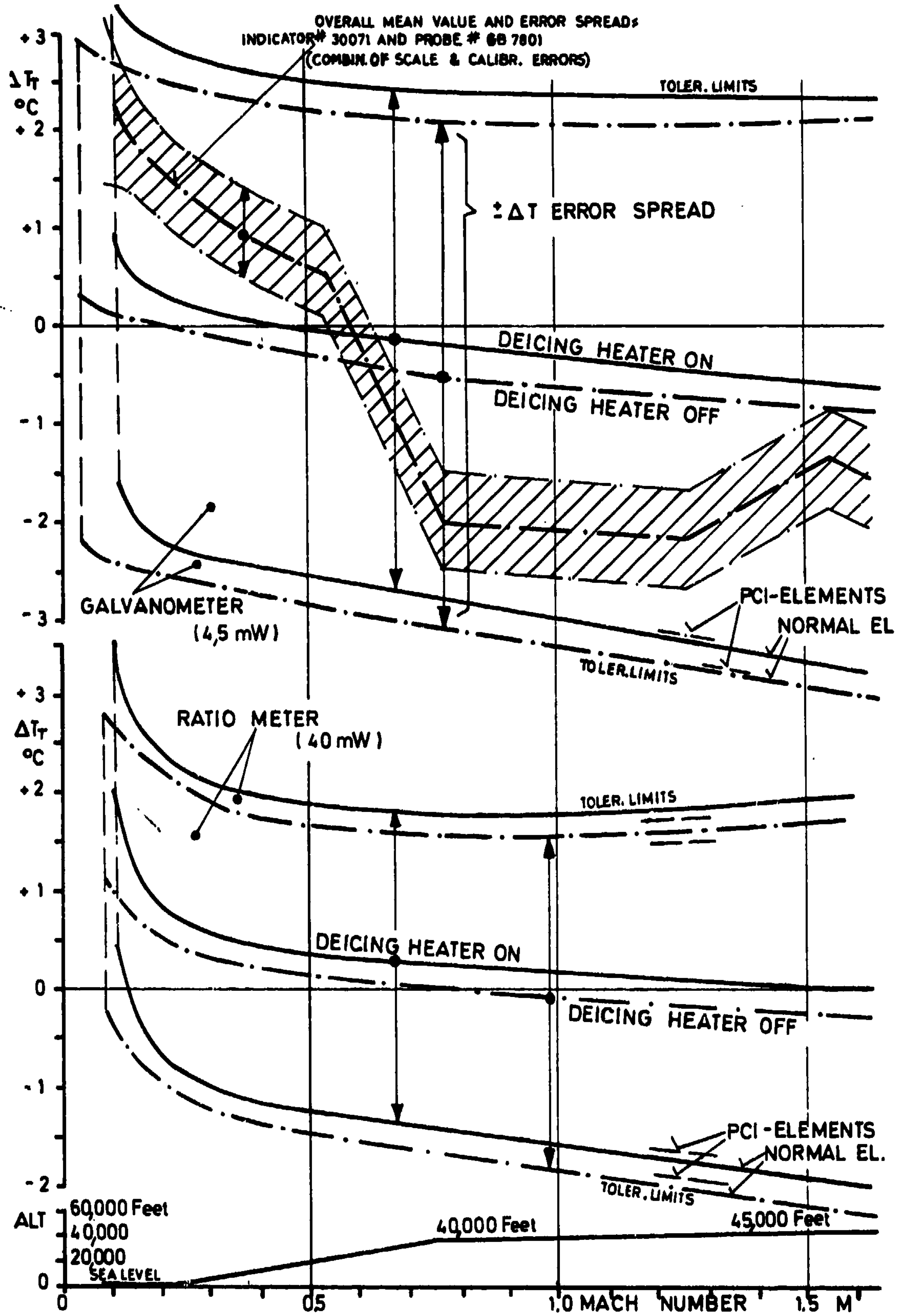


Fig.109 Overall temperature error of a TAT-probe with ratiometer or galvanometer

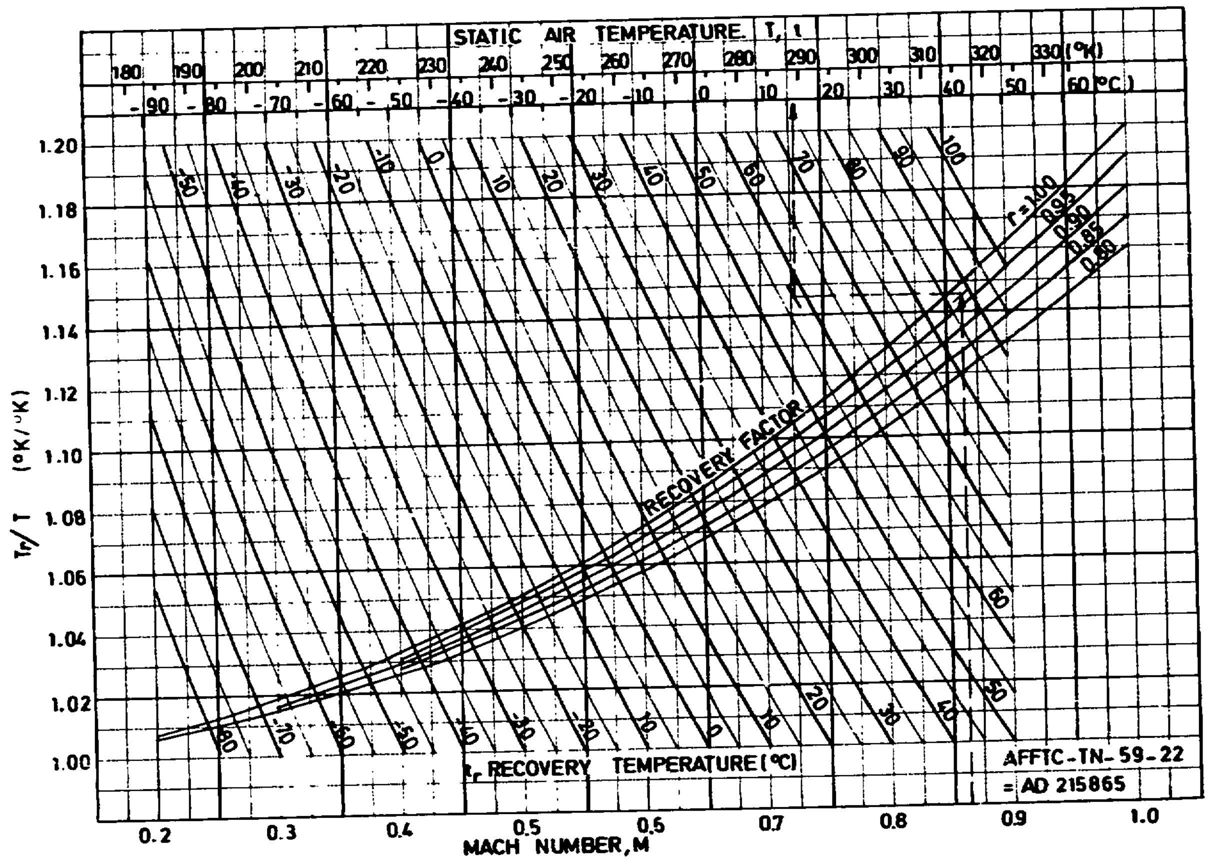


Fig.110(a) Static temperature as a function of a Mach number, recovery factor and recovery temperature (M 0.2 to 0.9)

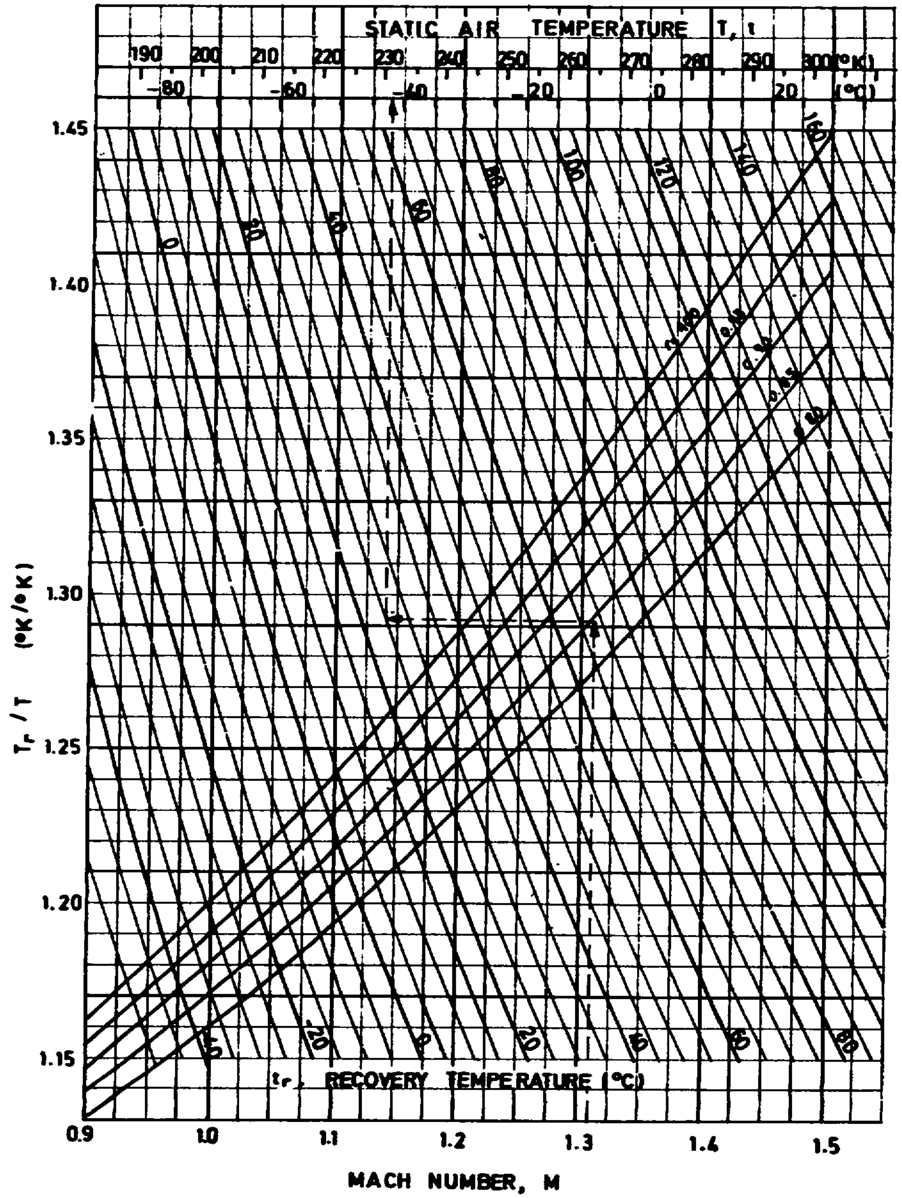


Fig.110(b) (Continued) M 1.15 to 1.45

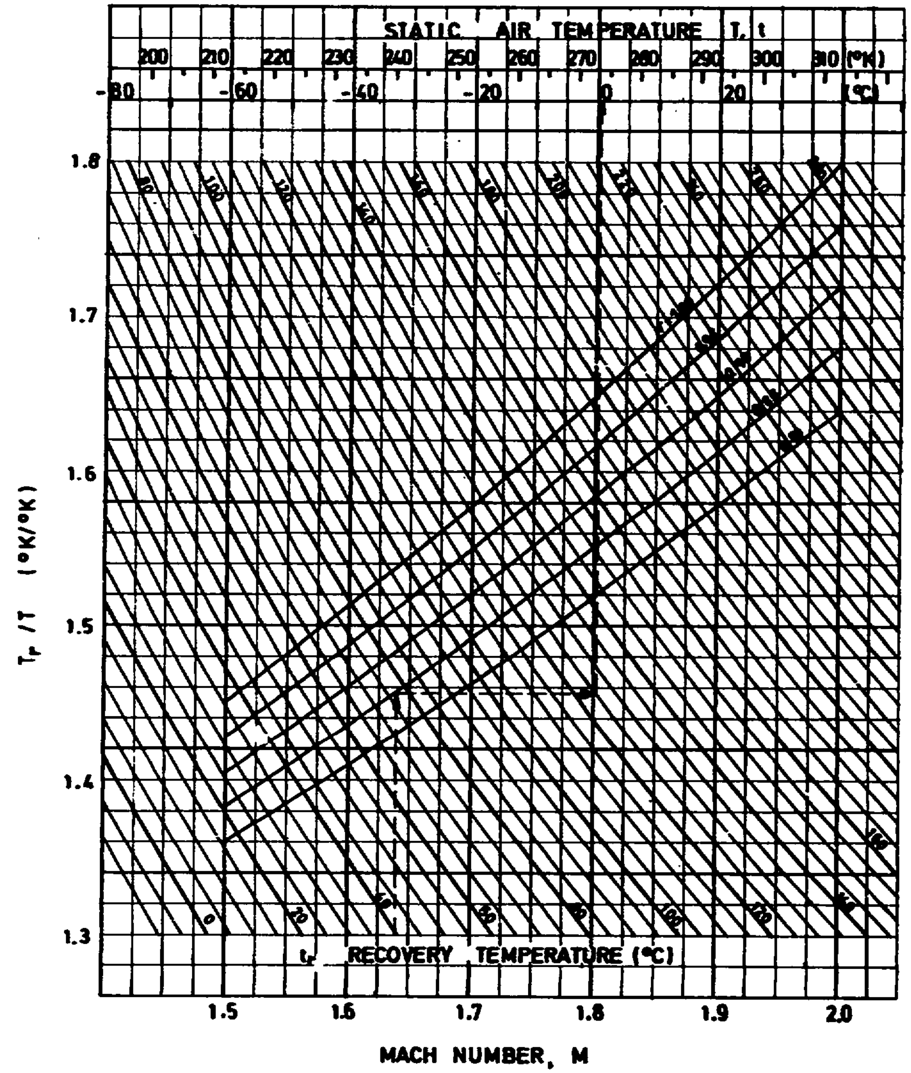


Fig.110(c) (Continued) M 1.3 to 1.8

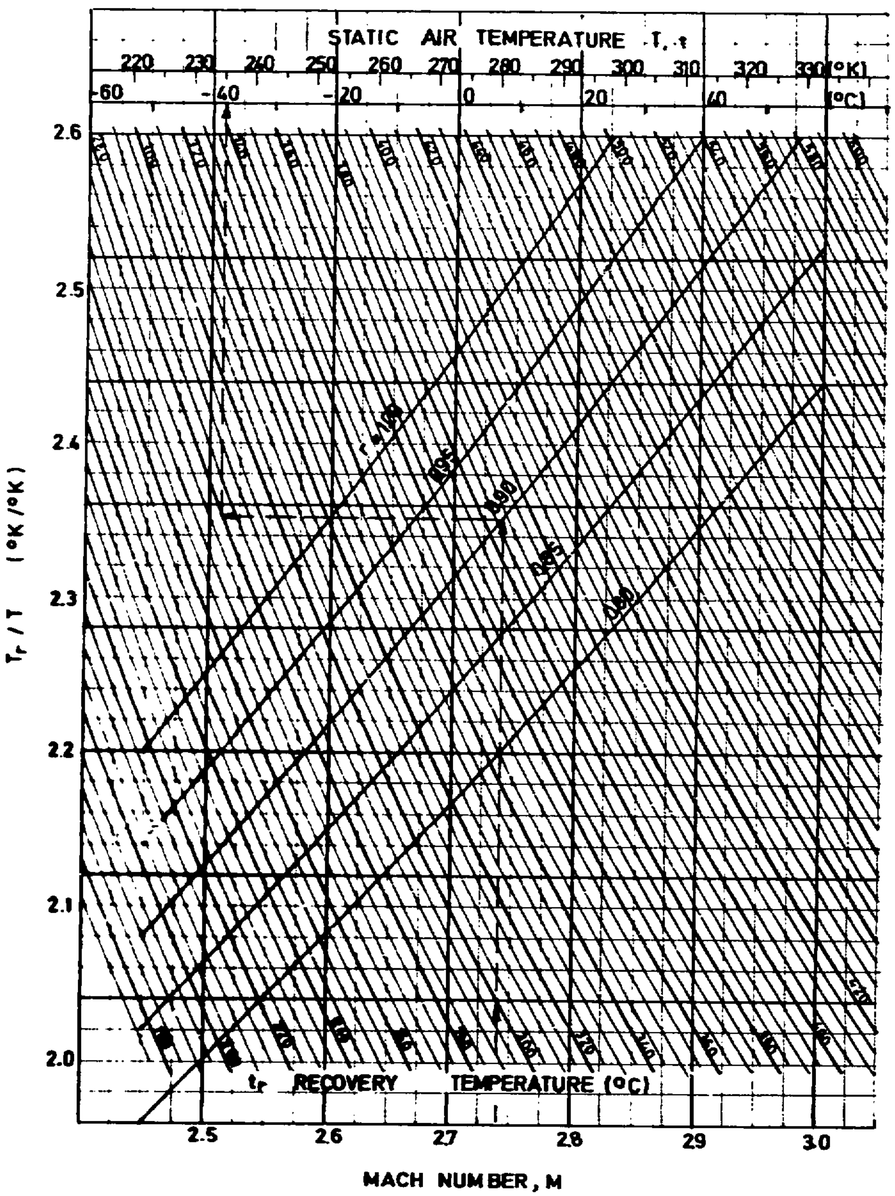


Fig.110(d) (Continued) M 1.6 to 2.2

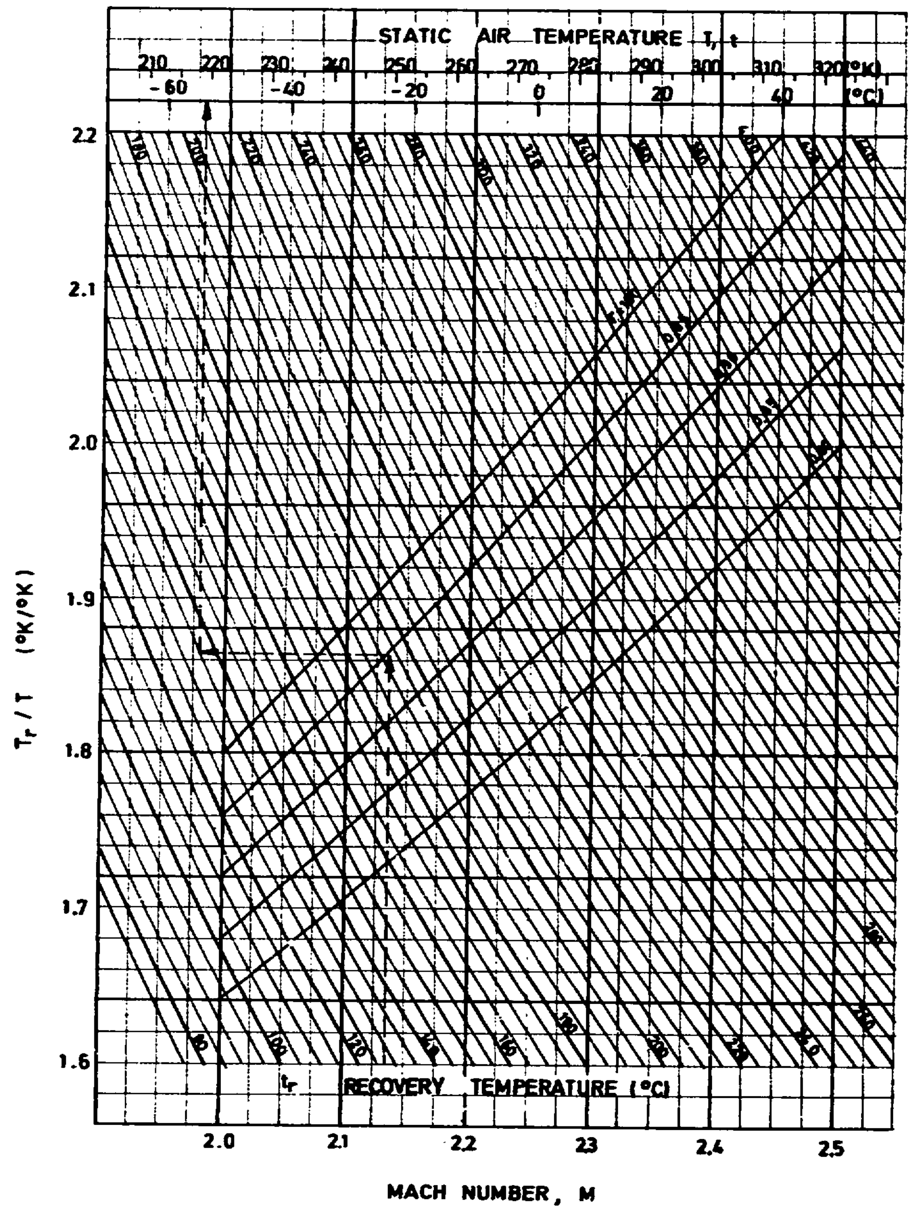


Fig.110(e) (Concluded) M 2.0 to 2.6

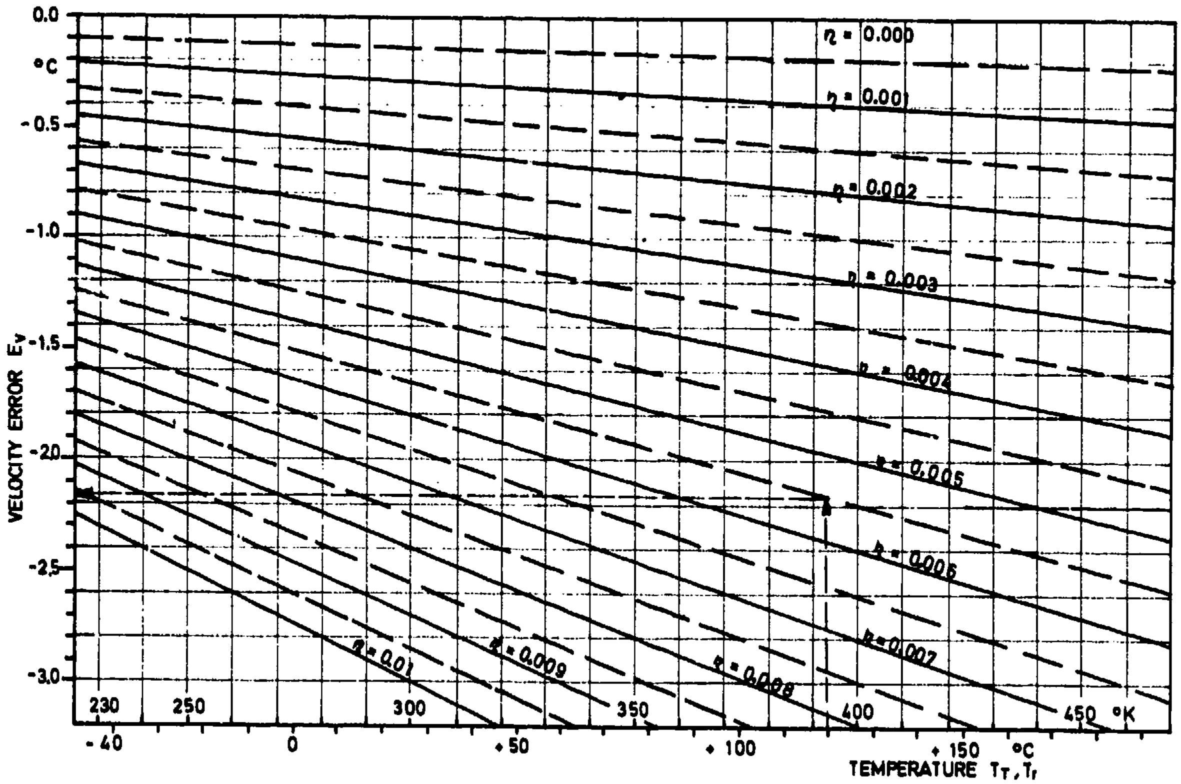


Fig.111 Velocity error E_V as a function of recovery error η and total temperature T_t $\approx$ recovery temperature T_r

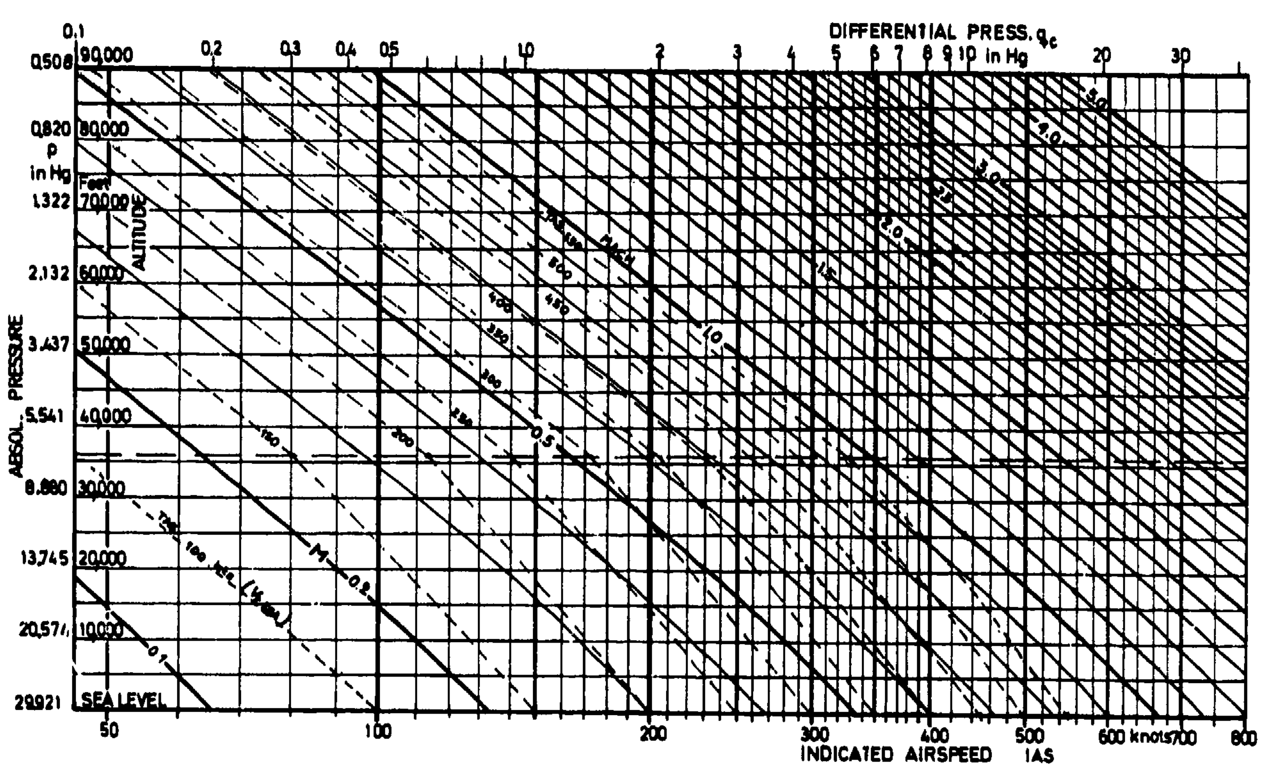


Fig.112 Mach number vs. indicated airspeed and pressure altitude

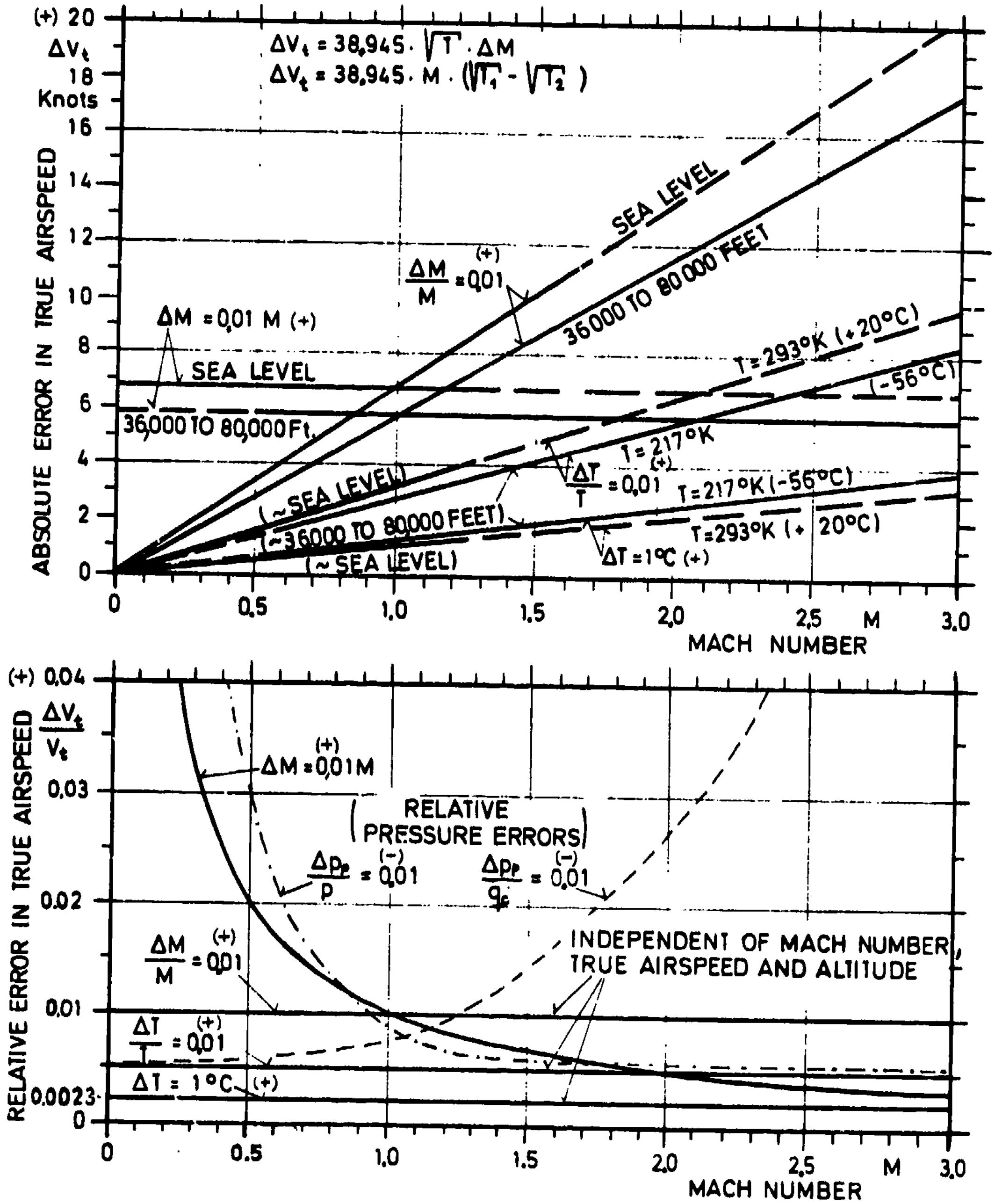


Fig.113 Absolute and relative error in the airspeed vs. Mach number as functions of given values of abs. and rel. errors in Mach number and static temperature