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MECHANICAL ENGINEERING REPORT 159

A REVIEW OF AIRCRAFT CABIN CONDITIONING FOR OPERATIONS IN AUSTRALIA

by

BRIAN REBBECHI

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SUMMARY

A review is presented of aircraft cabin conditioning. This review has been undertaken because of the inadequate performance of many aircraft air conditioning systems in the hottest conditions encountered in Australia. The factors included in this study were the climatic conditions (both Australia and world-wide), human performance in hot conditions, the heat balance of aircraft cabins, cooling system performance, and specification of cabin environment control systems.

It is concluded that climatic conditions in Australia are not severe in a world-wide context, and that there is no technological reason why the cabin conditioning systems of aircraft should be inadequate. Compliance with present RAAF specifications will provide an acceptable cabin environment for operation of aircraft in Australia.

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16. *ABSTRACT*

A review is presented of aircraft cabin conditioning. This review has been undertaken because of the inadequate performance of many aircraft air conditioning systems in the hottest conditions encountered in Australia. The factors included in this study were the climatic conditions (both Australia and world-wide), human performance in hot conditions, the heat balance of aircraft cabins, cooling system performance, and specification of cabin environment control systems.

It is concluded that climatic conditions in Australia are not severe in a world-wide context, and that there is no technological reason why the cabin conditioning systems of aircraft should be inadequate. Compliance with present RAAF specifications will provide an acceptable cabin environment for operation of aircraft in Australia.

CONTENTS

NOMENCLATURE

A	Plan projected area of transparencies (m^2)
COP	Coefficient of performance
C_p	Specific heat of air at constant pressure (kJ/kg °C)
D	Diameter of a black globe (m)
ET	Effective temperature (°C)
h_c	Convective heat transfer coefficient (air–clothing surface) (W/m^2 °C)
h_{cg}	Convective heat transfer coefficient of a sphere (W/m^2 °C)
$h_{cg(150)}$	Convective heat transfer coefficient of a 150 mm black globe (W/m^2 °C)
$h_{cg(50)}$	Convective heat transfer coefficient of a 50 mm black globe (W/m^2 °C)
h_r	Linearised radiation exchange coefficient (W/m^2 °C)
h_t	Overall heat transfer coefficient for transparencies (W/°C)
h_w	Overall heat transfer coefficient for cabin walls (W/°C)
h_{wo}	Overall heat transfer coefficient (W/m^2 °C)
k_c	Clothing conductivity (W/m^2 °C)
M	Mach number
m_t	Cabin air mass flow (g/s)
OAT	Outside air temperature (°C)
p_i	Absolute pressure at point i (kPa)
Q_c	Heat removal rate of cooling air (W)
Q_{cg}	Convective heat transfer from the surface of a black globe (W)
Q_{cl}	Heat lost by convective heat transfer from persons clothing surface (W)
Q_{ij}	Heat (or work) transfer when cycle process moves from point i to point j (W)
Q_m	Pilot's metabolic heat output (W)
Q_r	Radiation exchange of pilot with surroundings (W)
Q_{rg}	Radiative heat transfer from a black globe (W)
Q_s	Solar radiation heat transfer through transparencies (W)
Q_{si}	Solar radiation heat flux (W/m^2)
Q_{sp}	Solar radiation heat input rate to a person (W)
Q_{sg}	Heating of a black globe by direct solar radiation (W)
Q_t	Total heat input to an aircraft cabin (W)
q_m	Metabolic heat rate of a person (W/m^2)
q_l	Latent heat of condensation of water (kJ/kg)
r_e	Recovery factor

r_c	Pressure ratio of cooling unit compressor
r_p	Engine compressor bleed air pressure ratio
r_t	Cooling system expansion turbine pressure ratio
T_a	Ambient air temperature (°C)
T_e	Mean temperature of air in vicinity of the pilot (°C)
T_g	Temperature of a black globe (°C)
$T_{g(150)}$	Temperature of 150 mm black globe (°C)
$T_{g(50)}$	Temperature of a 50 mm black globe (°C)
T_{in}	Cooling air inlet temperature (°C)
T_m	Cabin mean air temperature (°C)
T_s	Aircraft external skin temperature (°C)
T_{sk}	Pilot's mean skin temperature (°C)
T_w	Wall surface temperature (°C)
T_{wb}	Psychrometric wet bulb temperature (°C)
T'_{wb}	Naturally convected wet bulb temperature (°C)
V	Air velocity (m/s)
WBGT	Wet bulb globe temperature (°C)
α_{cs}	Clothing absorptivity to solar radiation
γ	Ratio of specific heats of air
δ_r	Decrease in absolute humidity (kg moisture/kg dry air)
ϵ_{eff}	Effective emissivity from wall surface to pilot's clothing
θ	Angle of sun above horizon (radians)
η_t	Cooling system turbine efficiency
η_c	Engine compressor efficiency
η_{cc}	Cooling system compressor efficiency
τ	Transmittance of transparencies

1. INTRODUCTION

The cabin conditioning systems of military aircraft in service in Australia have frequently proved inadequate, particularly with regard to their performance in the hottest conditions encountered in this country. As the Services are anxious to avoid a recurrence of this problem with the purchase of new aircraft, ARL were asked to investigate all aspects of aircraft cockpit conditioning. This request was duly formalised as an Air Force Task.

The primary emphasis of this task was to avoid problems with future aircraft, rather than to suggest palliatives for existing aircraft. Some examination of the deficiencies in existing RAAF aircraft, and their causes, is of course inherent in this study. The original task requirement was as follows:

> "Study of aircraft cockpit environment and methods of conditioning of crew and equipment, with the object of specifying cockpit conditioning requirements for aircraft operating in the Australian climatic environment. The study to include:
>
> (*a*) Survey of climatic conditions for foreseeable RAAF operations.
>
> (*b*) Determination of cockpit conditioning design criteria for specification by RAAF, based on an assessment of heat balances in typical cockpit structures.
>
> (*c*) Examination of present or new techniques of conditioning to enable design criteria to be met with minimum weight, power and space requirements.
>
> (*d*) Preparation of outline system specifications based on the most suitable proposed techniques."

This is the final report on the Cockpit Conditioning Task, there are areas where further work can be carried out, as indicated in the body of the report; however, the main objectives of the task have been fulfilled. It is intended that this report will provide assistance to the RAAF and other services in specifying cockpit conditioning systems for new aircraft, or reviewing problems in existing aircraft; there is summarised a large body of literature pertaining to cockpit conditioning, both Australian and overseas.

The severity of the problem, as it affects the RAAF, is that all of their jet aircraft (with the exception of the F-111C), have exhibited serious deficiencies in the performance of the cockpit conditioning system. Two of the earlier RAAF jet aircraft, the Vampire Mk 30 (Jost, Noble and Rowland 1953*), and the Canberra Mk 20 (Sutherland and Scotland 1958), were not equipped with cooling systems when they arrived in Australia. Fitment of cold air units to those aircraft was subsequently carried out in Australia, when it became apparent that operation in Australian summer conditions was adversely affected by crew heat stress. Another early RAAF jet aircraft, the Sabre Mk 31, was equipped with a relatively effective cooling system. All of these aircraft, and the later Mirage IIIO have, however, experienced severe cockpit transparency misting, and fog, under some flight conditions, particularly in the tropics. This problem can also occur during landing approach or take-off, and can thus be hazardous.

Preliminary investigations of the problem showed that there is no technological reason why the performance of cooling systems should continue to be so poor, the problem appears to have remained primarily because of two interrelated factors:

(*a*) neglect of cabin conditioning by aircraft manufacturers (particularly European), even though the technology has existed to provide adequate solutions; and

(*b*) inadequate specification, or follow-up acceptance tests, by the RAAF.

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* References are listed alphabetically at the end of this report.

In support of these conclusions it is of interest to note that the Vampire Mk 30 prototype cooling system evaluated by the RAAF Aircraft Research and Development Unit in 1951 (see Jost *et al.* 1953) brought about cabin temperatures no higher than that of the Macchi MB326H introduced into Australia some 15 years later (see Aircraft Research and Development Unit 1969).

The most recent RAAF combat aircraft, the F-111C, reportedly has no problems with the cockpit conditioning system (ARL file M2/262, folios 38, 39).

2. THE EXTERNAL ENVIRONMENT

2.1 General Comments

The external environment is relevant to cockpit conditioning for two reasons:

(*a*) its effect on the transfer of heat to the cabin and the occupants by conduction, convection and radiation; and

(*b*) its effect on the condition of the ventilating air at entry to the cabin, and on the performance of the cooling unit.

The external environmental factors of primary interest are temperature, humidity and solar radiation. Other climatic factors such as wind, rain, snow, hail, air density, ozone concentration, sand and dust, while important with regard to other aspects of aircraft operation, are not directly relevant to this study.

The Australian climate, with regard to temperature and humidity, is not unusually severe in a world-wide context. This has been demonstrated by McRae (1977) in a comparison of Australian upper air conditions with the world-wide environment described in the United States MIL-STD-210B (Military Standard Climatic Extremes for Military Aircraft). This report by McRae is to be incorporated in a revision of Australian Defence Standard 5168 'Climatic Extremes for Service Equipment'; this revision will incorporate ground, sea and air environments. The work is being carried out by a Defence Standardisation Committee working party.

2.2 Temperature

A comparison is given in Figure 1 between the United States MIL-STD-210B, the British Av. P. 970, and the RAAF atmospheric environment (RAAFAE). A comparison is also given in Figure 2 between the RAAFAE, and the recorded 1% extreme* high and low temperatures for Australia and nearby islands (from McRae 1977; Redman and McRae 1975).

The primary United States climatic design standard is MIL-STD-210B. This standard separately describes the climatic environment for world-wide ground, naval and air (up to 80 km) operations. The maximum recorded, 1, 5, 10 and 20% extremes are given for the most severe geographical locations; the 1% extreme being chosen in this standard as an appropriate design criterion for operations. Other United States climatic design information (not shown in Fig. 1) is provided by MIL-E-38453A, and an Air Force–Navy Aeronautical Bulletin 421. Detailed background information on the formulation of MIL-STD-210B is given by Sissenwine and Cormier (1974).

From Figure 1 it can be seen that the RAAFAE is less severe at most altitudes than the world-wide MIL-STD-210B; at ground level the RAAFAE, at 43°C, accords closely with the ground-level 1% extreme for Australia (Redman and McRae 1975) of 43·7°C. The 1% extreme ground-level temperature for world-wide operations from MIL-STD-210B is 49°C.

2.3 Humidity

A comparison is given in Figure 3 between absolute humidity levels (kg moisture/kg dry air) as specified in the United States MIL-STD-210B, the British Av. P. 970, and the RAAFAE.

* The 1% extreme is that temperature (or humidity) that is equalled or surpassed for 1% of time (7·5 h) in the most severe month.

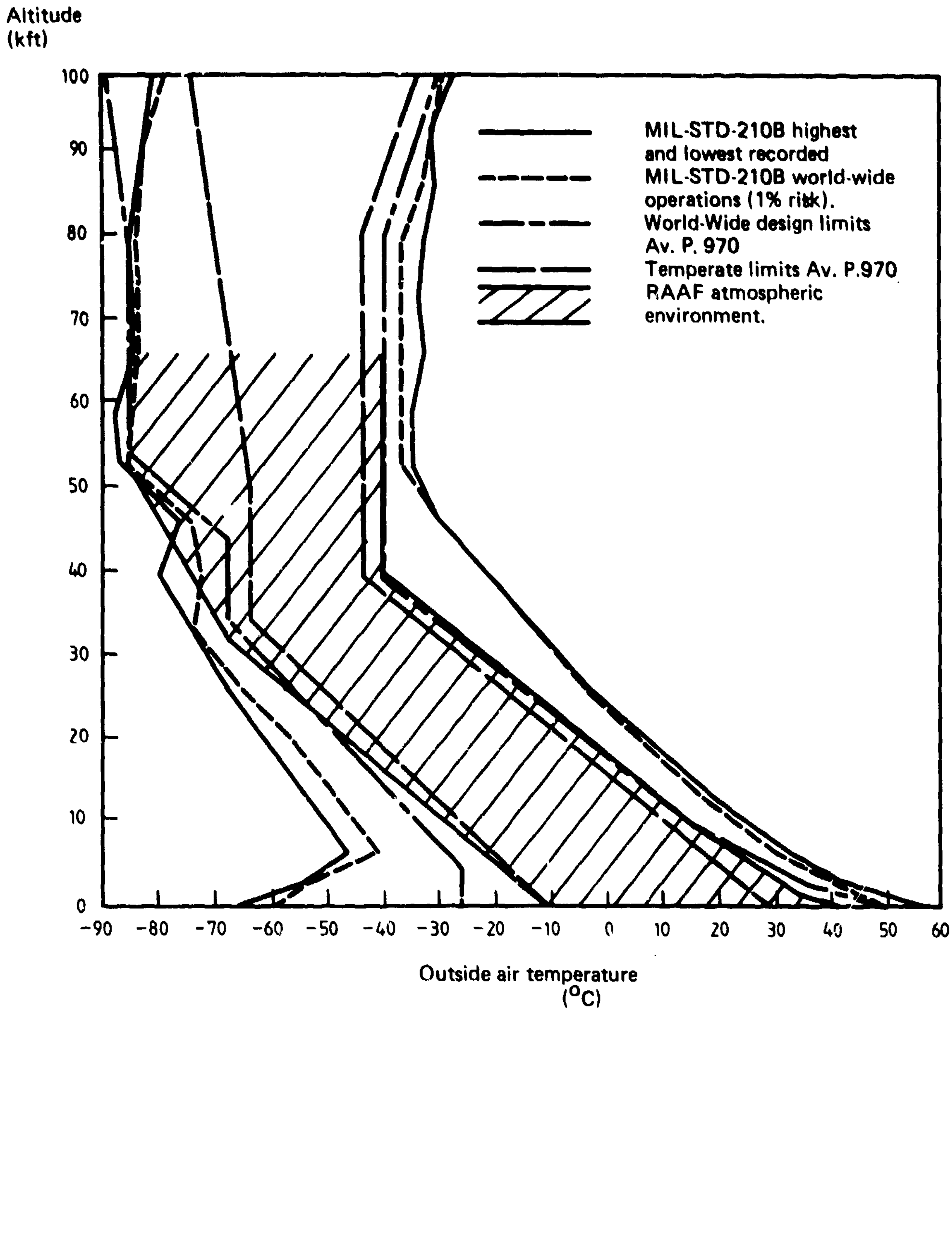

FIG. 1 GRAPH OF TEMPERATURE VERSUS ALTITUDE FOR SEVERAL DESIGN STANDARDS

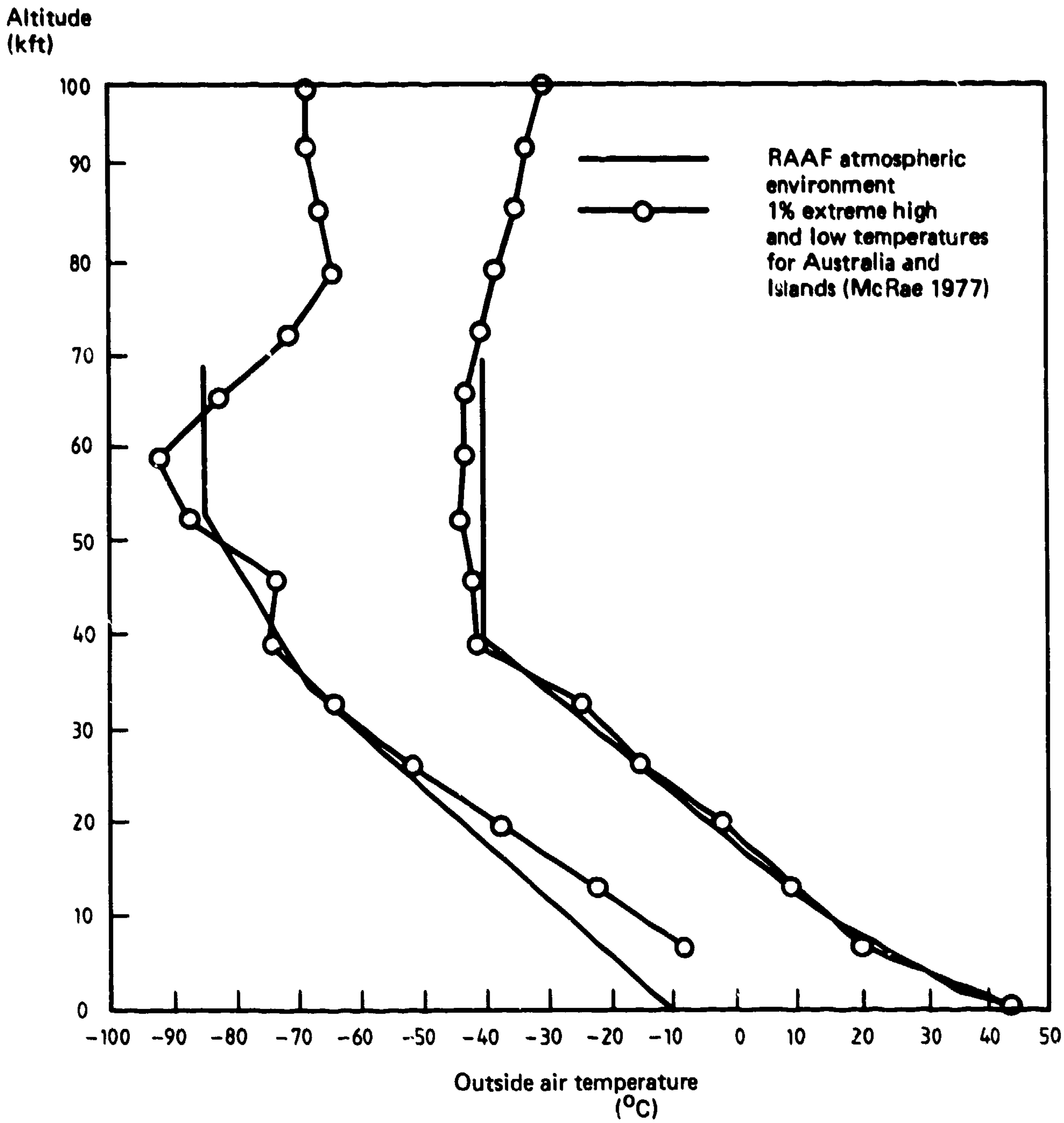

FIG. 2 COMPARISON BETWEEN RAAF ATMOSPHERIC ENVIRONMENT AND 1% EXTREME HIGH AND LOW TEMPERATURES FOR AUSTRALIA AND THE ISLANDS

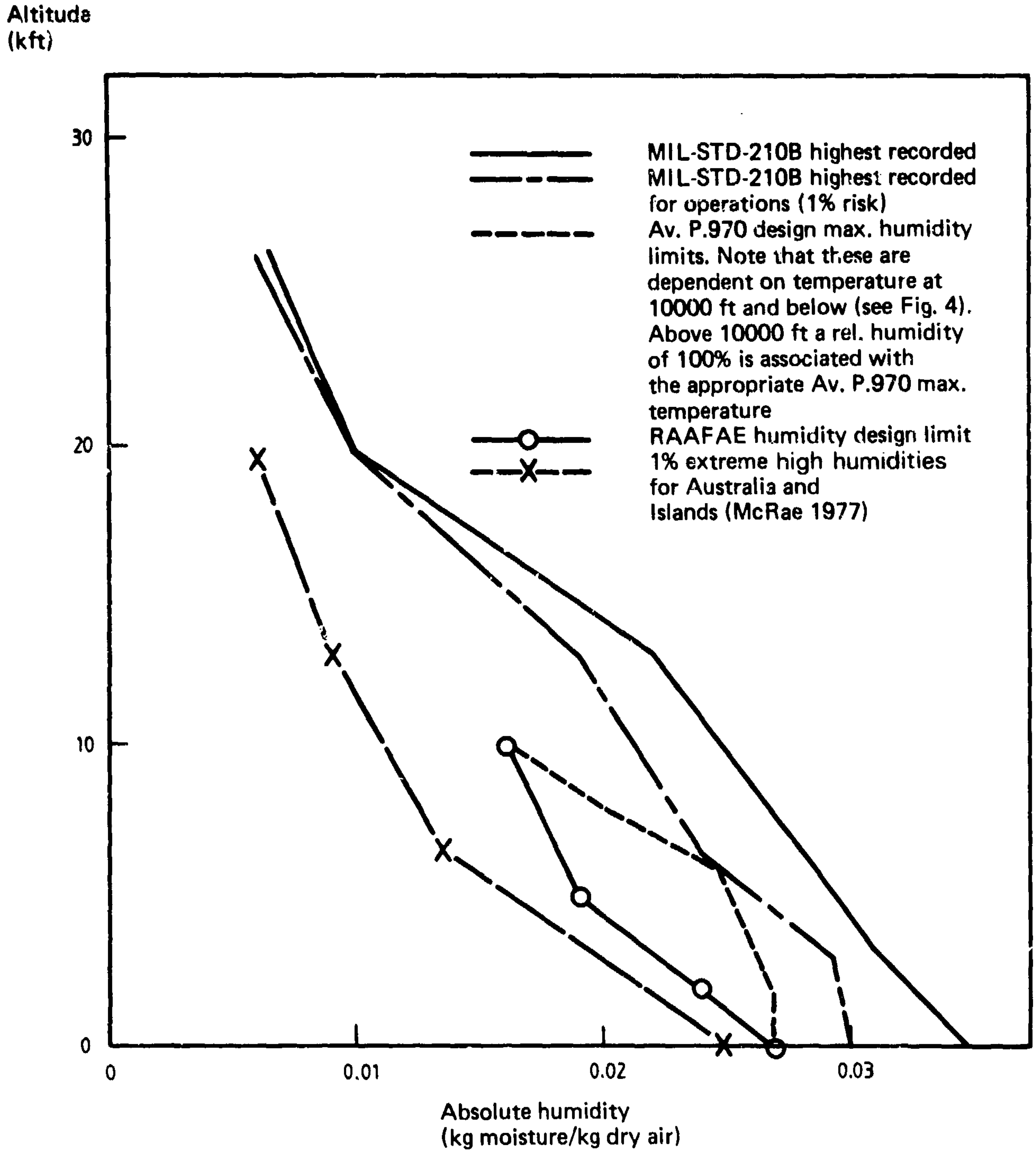

FIG. 3 ABSOLUTE HUMIDITY AS A FUNCTION OF ALTITUDE FOR SEVERAL DESIGN STANDARDS

Also shown are four data points for 1% extreme humidities, recorded in Australia and nearby islands (McRae 1977). The humidity levels for the RAAFAE have been calculated from the RAAF method of presentation, which is dewpoint vs. altitude (see Appendix 1).

The British Av. P. 970 and the United States MIL-E-38453A are the only specifications known to the author which show that the design humidity level is properly a function of temperature, as well as altitude. The United States MIL-E-38453A, however, permits this change in design humidity level with high temperatures only for the ground cooling case; the flight situation, even at very low altitude, is covered by MIL-STD-210B. The design humidity levels for Av. P. 970 and MIL-E-38453A, as a function of both temperature and altitude, are shown in Figure 4.

World wide joint extremes of temperature and humidity have been summarised by Cormier (1974). The joint values of temperature and humidity which are exceeded 0·1, 1·0 and 5·0% of the time in the most severe month are plotted in Figure 5. Also plotted on this figure are the corresponding extremes for Australia and nearby islands, from data derived by McRae (1980). In Figure 5, then, the lines of constant percentage are those lines joining a series of points at which that *joint* temperature and humidity is exceeded for a particular percentage of the time in the most severe month. For example, at the temperature and humidity level of point A in Figure 5, 0·1% of the readings (for the worst month) lie within the shaded region, and 99·9% lie anywhere outside of this region. In the design of equipment where both temperature and humidity influence performance, information on the percentage of time the performance will not meet requirements cannot necessarily be obtained from Figure 5. For example, an air-cycle cooling system may provide an equivalent cabin environment condition along the line BC in Figure 5. The design information then sought will be the percentage of time temperature–humidity points are above this line. For Australian conditions, however, very little error is introduced, for systems of similar characteristics to the line BC (Fig. 5), if the 1·0% curve is used for design purposes. The RAAFAE is shown on Figure 5; it can be seen to be well outside of the 1·0% probability of joint occurrence of temperature and humidity for Australia.

2.4 Solar Radiation

Statistics of solar radiation for Australia have not been examined; however, the maximum incident solar radiation for Australia is usually taken as 1,000 W/m^2. This solar radiation level is confirmed by reference to the Engineering Sciences Data No. 69015 on solar heating (Engineering Sciences Data Unit 1969). From these data the variation in solar radiation intensity is shown to be primarily a function of atmospheric water content and altitude, the maximum intensity of 1,360 W/m^2 being closely approached at 10,000 m and above. The USAF design standard for solar radiation as a function of altitude is shown in Figure 6; this standard is for a very dry atmosphere. Typical values of solar radiation for Australia are shown also in Figure 6, based on information from the Engineering Sciences Data Unit (1969).

3. THE COCKPIT ENVIRONMENT FOR EFFECTIVE PERFORMANCE OF AIRCREW AND EQUIPMENT

3.1 Objectives of the Environment Control System

The objectives of the environment control system are to provide adequate heating, cooling and ventilation to ensure the operational efficiency of the occupants, and the efficient functioning of all equipment. The climatic criteria for occupant and equipment compartments differ; in Sections 3.2 to 3.4 the aircrew aspects are discussed, and the requirements for equipment operation are described in Section 3.5.

3.2 Heat Stress of Aircrew

3.2.1 General Comments

It is in the definition of what constitutes the limits of an acceptable environment for occupants that considerable difficulties are encountered; this is discussed in Section 3.2.3. The cockpit environment itself is readily defined in terms of the dry bulb temperature, radiant heating from surroundings, moisture content of the air, and air velocity. All of these factors influence the rate of heat exchange between aircrew and their surroundings. From the viewpoint of heat stress

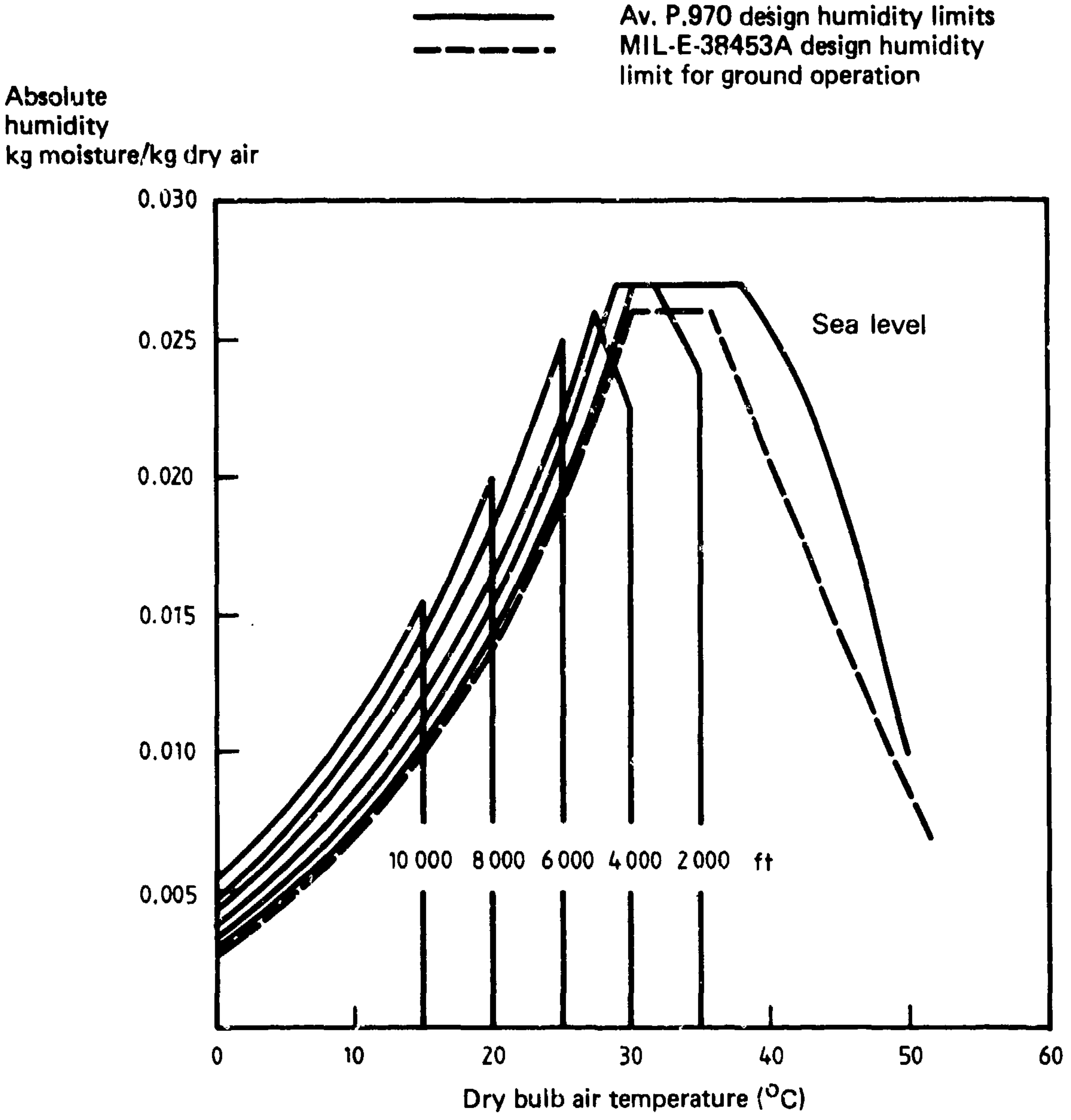

FIG 4 ABSOLUTE HUMIDITY LEVELS AS A FUNCTION OF BOTH TEMPERATURE AND ALTITUDE

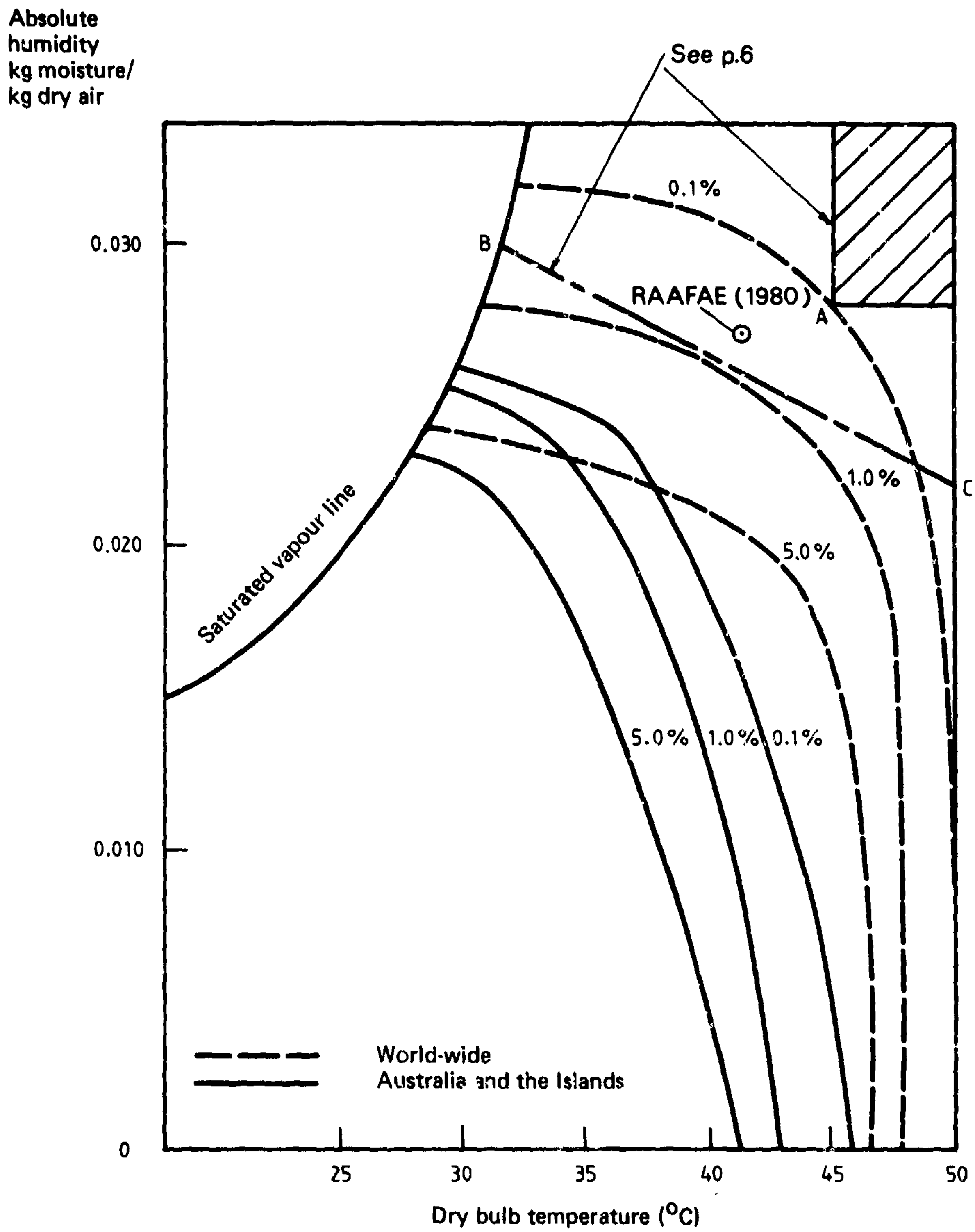

FIG. 5 JOINT VALUES OF HIGH TEMPERATURE AND HIGH HUMIDITY WHICH ARE EXCEEDED 0.1, 1.0 AND 5.0 PERCENT OF THE TIME FOR THE MOST SEVERE MONTH (AT SEA LEVEL)

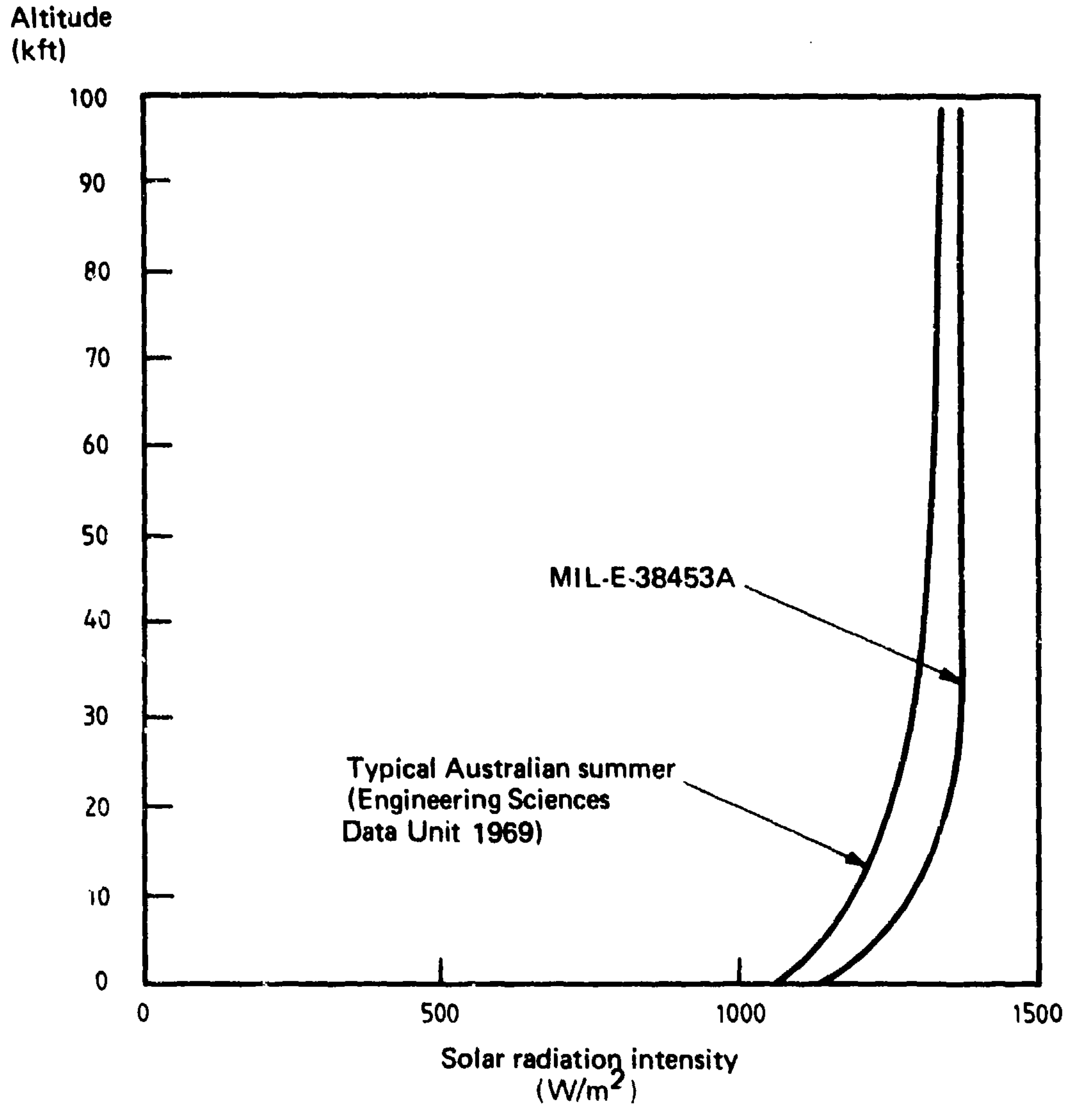

FIG. 6 SOLAR RADIATION INTENSITY AS A FUNCTION OF ALTITUDE

the work rate, which affects metabolic heat production, and clothing, which affects heat transfer from the skin, need also to be included.

There have been many attempts to develop a single quantitative scale to be used as a measure of heat stress, which would take into account all of the above factors. A detailed analysis of these many heat stress indices can be found in such references as Kerslake (1972); it is proposed to refer here to only two indices, the Effective Temperature (ET) index, and the Wet Bulb Globe Temperature (WBGT) index. These two are widely used, the former mainly in laboratory type situations, and the latter in both laboratory and field trials. A description of these two indices follows.

3.2.2 Heat Stress Indices

3.2.2.1 Effective Temperature Index

The Effective Temperature (ET) index (not to be confused with the Environmental Temperature Index, which is also used for the definition of thermal environment in aircraft cabins —United States Army Air Force 1945), is widely used in research into comfort and heat stress of humans. The ET of an environment is the temperature of still, saturated air which would give rise to an equivalent sensation; it is found from a nomogram (see Fig. 7). This nomogram is appropriate to subjects wearing normal indoor clothing. Adjustments can be made for thermal radiation; the index is then termed the Corrected Effective Temperature (CET), and is found by entering the nomogram with the observed value of the temperature of a 150 mm black globe.

Ramanathan and Belding (1973) have pointed out inconsistencies in the ET and CET scales, as has Kerslake (1972), particularly with regard to underestimation of the effects of humidity in severe conditions approaching the tolerance limit. These inconsistencies are not such as to negate the usefulness of this index—it is very widely used and, according to Kerslake (1972), works quite well.

3.2.2.2 Wet Bulb Globe Temperature Index

The Wet Bulb Globe Temperature (WBGT) index takes account of incident thermal radiation, ambient temperature, humidity and air velocity. It is defined (Kerslake 1972) as

$$\mathrm{WBGT} = 0{\cdot}7T'_{\mathrm{wb}} + 0{\cdot}2T_{\mathrm{g}} + 0{\cdot}1T_{\mathrm{a}}, \tag{1}$$

where T_{a} = ambient shade temperature (°C),

T'_{wb} = temperature of a naturally convected wet bulb (°C),

T_{g} = temperature of a 150 mm black globe (°C).

An alternative formula which can be used if the psychrometric wet bulb temperature is available, is

$$\mathrm{WBGT} = 0{\cdot}7T_{\mathrm{wb}} + 0{\cdot}3T_{\mathrm{g}}, \tag{2}$$

where T_{wb} is the psychrometric wet bulb temperature. In this index no measurement of air velocity is required; the effect of air movement is reflected in the globe temperature (through convective heat transfer) and in the naturally convected wet bulb temperature, through its effect on evaporation. The index was originally intended only for application to troops training in the open, under conditions of high radiant heat load. Because of its simplicity of use, the index has become widely used, particularly in the military sphere—for example the RAF Institute of Aviation Medicine (Harrison *et al.* 1977), the USAF School of Aerospace Medicine (Nunneley and James 1977), and the RAAF Institute of Aviation Medicine (Readett and Knights 1978).

Inconsistencies in the WBGT index are discussed by Ramanathan and Belding (1973) and Kerslake (1972), and are particularly evident when comparing humid and dry environments of equivalent WBGT. Despite these inconsistencies, the widespread use of the index is an indication of its usefulness; no index in use at the present time is free from limitation in its interpretation.

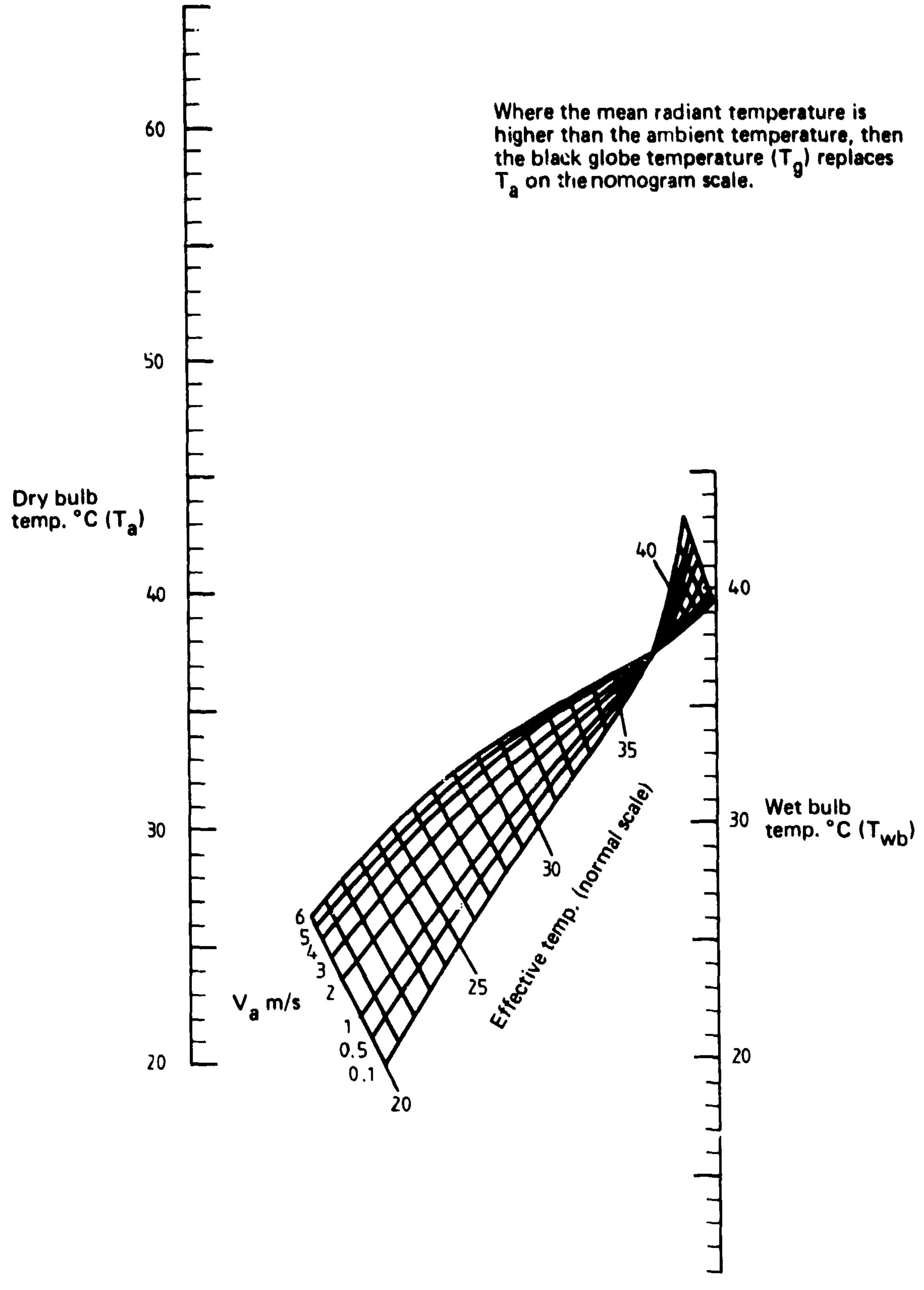

FIG. 7 NOMOGRAM FOR DETERMINATION OF EFFECTIVE TEMPERATURE (Reproduced from Kerslake 1972)

It should be noted that the constant weighting of the temperatures in the WBGT index (see Equations (1) and (2)) implies its validity only in severe heat stress conditions, where heat loss of a subject by evaporation is the predominant heat dissipation mechanism (as shown by the weighting of 0·7 given to the wet bulb temperature). The WBGT index is quite inapplicable in lower stress regions, which tend towards the comfort level. This feature of the WBGT index can be seen by reference to Figure 7; whereas the weighting given to the wet bulb and globe temperatures is similar to the ET for the high stress regions of WBGT = 30–35°C, for the low stress regions (WBGT 20°C) the appropriate weighting to the wet bulb temperature as given by the ET scale is approximately 0·3. For colder conditions still, heat dissipation by evaporation will be negligible.

A seemingly small but significant change has recently taken place in WBGT measurement; because of restricted space in modern aircraft cockpits, a black globe diameter of 40 mm is used by the USAF School of Aerospace Medicine in their Miniature Environmental Monitor* (James *et al.* 1975; Nunneley and James 1977), and a black globe diameter of 50 mm is used by the RAAF Institute of Aviation Medicine in their portable 'Minilab' instrument† (Readett and Knights 1978) and by the RAF Institute of Aviation Medicine (Harrison *et al.* 1977). The use of a smaller globe diameter results in a greater heat transfer coefficient at the same wind speed, and so its temperature is closer to the air temperature. Thus the WBGT index level calculated using the small globe will be less than that calculated using the large globe. The difference between the two WBGT levels will be approximately 0·7°C, depending upon humidity and dry bulb temperature. A general analysis of the heat balance of a globe is presented in Appendix 2; a comparison between the temperatures measured by 50 mm and 150 mm globes is given in Figure 8, and the relationship between globe temperature, surrounding wall temperature and incident radiation is given in Figure 9.

3.2.3 Response of Aircrew to Heat Stress

If the cooling of aircrew to a level which would ensure their comfort could be achieved without decrement of aircraft performance, then their performance under heat stress would not be relevant to this study. However, at the present time, with the known exception of the liquid-conditioned suit which has very low power requirements, the provision of adequate crew cooling will have an adverse effect on aircraft performance. Whilst is it not yet possible to quantify human performance under heat stress, to the extent that it could be used in a trade-off study against aircraft performance, the level of heat stress at which measurable performance degradation occurs can be established. The results of a number of studies of human performance under heat stress are discussed in the following sections, 3.2.3.1. to 3.2.4.

3.2.3.1 Measured Psychological and Physiological Effects of Heat Stress on Crew while in Flight

There is only one account known to the author where both physiological and psychological stress have been measured in flight. This was by Bollinger and Carwell (1975) who measured performance decrement of aircrew performing in-flight tasks. This was achieved by assessing mission performance using photo target acquisition scores. While specific details were not given, the crew apparently were wearing normal flying clothing, helmets and g-suits. A wide range of physiological parameters was also measured. The results from missions on which the crew were subjected to WBGTs of up to 31°C were compared with results from winter flights, this showed that nearly twice the percentage of targets were missed in the summer as in the winter (although the majority, greater than 90%, were photographed in both seasons). The misses due to pilot error were spread over 24% of the summer and 10% of the winter flights.

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* The USAF Miniature Environmental Monitor is a four-channel recording system, which can be located in the crew seat of an aircraft, and records dry bulb temperature, black globe temperature, dewpoint and air velocity.

† The Minilab instrument is carried on the observer's lap; the WBGT is available as a direct read-out.

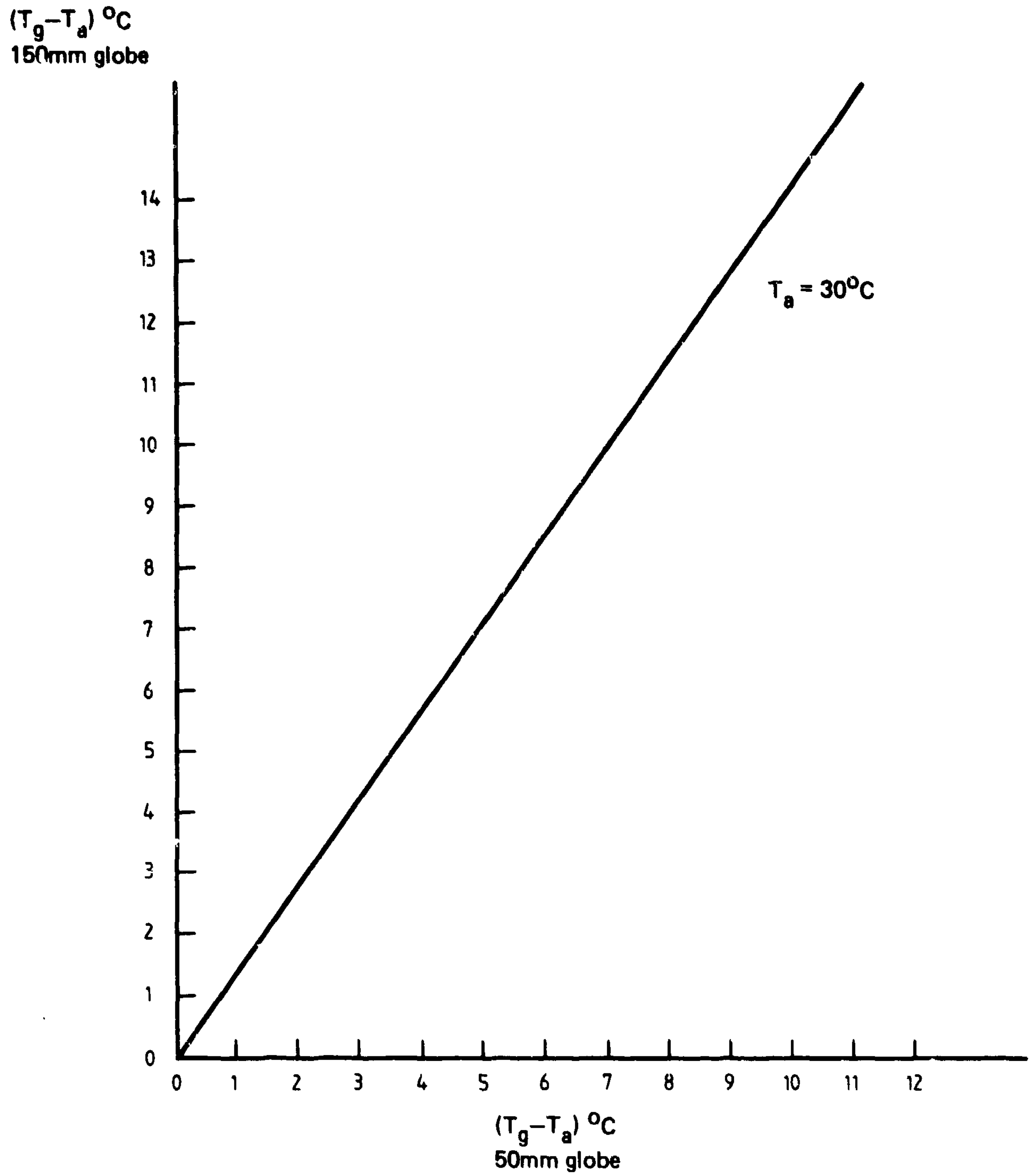

FIG. 8 GRAPH RELATING LARGE AND SMALL BLACK GLOBE TEMPERATURES UNDER VARIOUS RADIANT LOADS, FOR WIND SPEEDS IN THE REGION 0.5 TO 3.0m/s

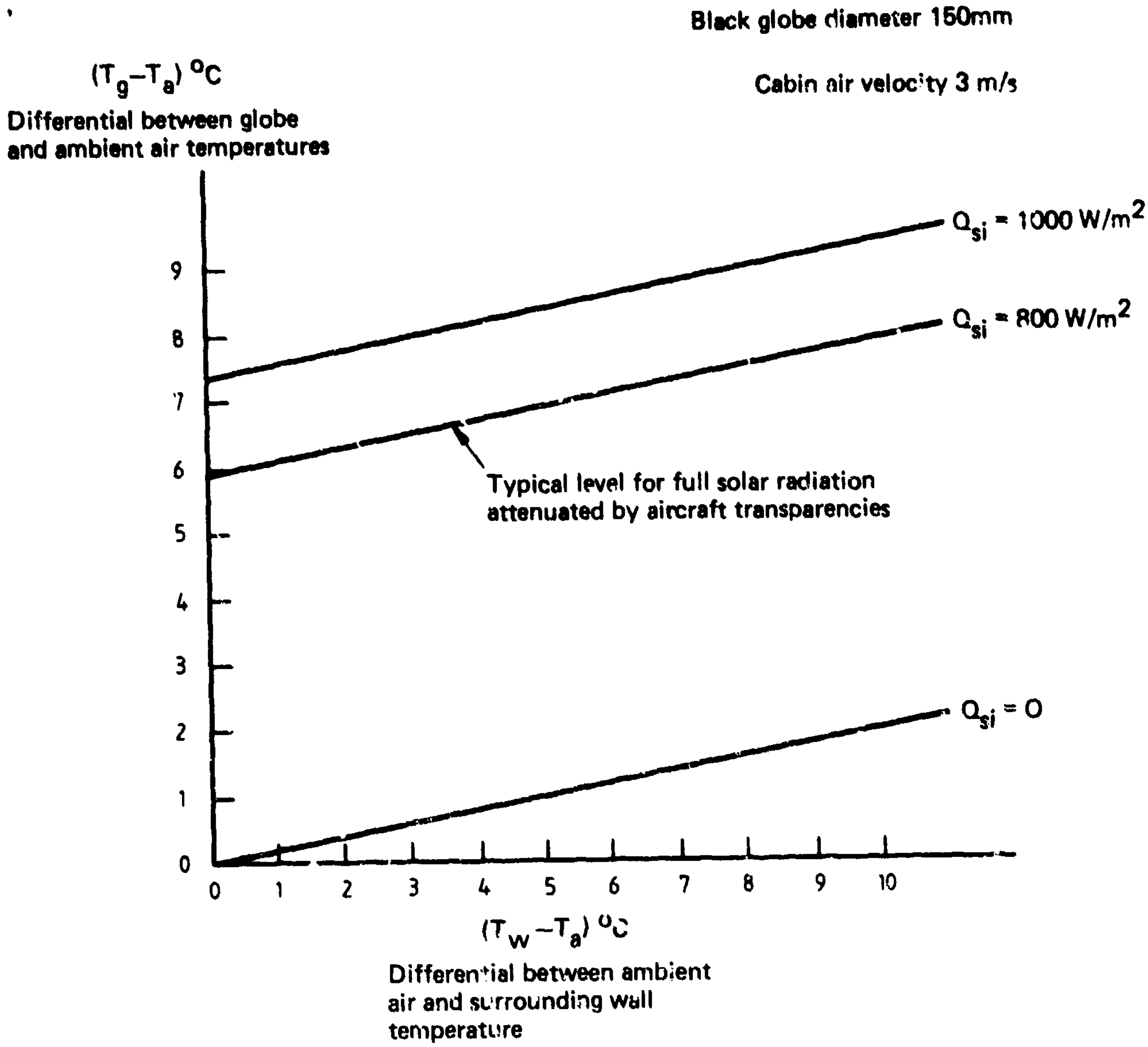

FIG. 9 TEMPERATURES OF A 150mm BLACK GLOBE

3.2.3.2 Measured Physiological Effect Only, of Heat Stress on Crew while in Flight and on Ground Standby

Nunneley and James (1977) measured cockpit air temperatures and crew skin temperatures during low altitude, high speed flights in a hot desert environment, aboard an F-111A aircraft. Conditions often fell outside the comfort zone (as designated by a mean crew skin temperature above 34°C). Particularly high head temperatures were noted.

Nunneley and Myhre (1976) measured heat stress on crew seated in an F-15 aircraft parked so that the cockpit was fully exposed to sunlight, or could be shaded. Cooling air from ground carts was available. Subjects wore a Nomex flying suit and helmet with lowered visor, but not a 'g' suit. The tests were conducted for a duration of 60 min. At a WBGT of 34·4°C the body temperature reached equilibrium, but the heart rate steadily rose throughout the test, demonstrating the existence of heat strain.

3.2.3.3 Measured Psychological and Physiological Effects of Heat Stress on Pilot in Simulated Flight

Iampietro *et al.* (1972) conducted experiments measuring physiological stress and performance in a general aviation trainer. Pilots were attired in light clothing, without helmets. Significant decrements in performance occurred at WBGTs of 30·3°C and 38·7°C. Duration of heat stress was 50 min. Routine tasks appeared to be unaffected by these WBGT levels, whereas complex tasks were adversely affected.

3.2.3.4 Measured Psychological Effect of Heat Stress on Subjects under Laboratory Conditions

The effects of heat on mental and perceptual performance have been studied widely. Grether (1973) and Johansson (1975) provide summaries of the considerable activity in this field. There are several difficulties in applying this work to aircrew performance, namely:

(*a*) the tasks are not flight-related;

(*b*) the effects of motivation as may exist for aircrew cannot be quantified, and laboratory tests may not represent this;

(*c*) the clothing worn for laboratory tests is almost without exception of a lower insulation value than the flying suit/'g' suit/'mae west'/helmet ensemble worn by aircrew;

(*d*) the work rate for aircrew in some instances will be greater than in most of the laboratory tests.

Grether (1973) concluded that mental and perceptual performance decreased at an ET above 29·4°C. However, there was a wide scatter in data points. Johansson (1975) concluded from his wide survey that ETs over 30°C will most likely impair performance of mental, perceptual and psychomotor tasks for subjects stripped to the waist. He found a critical zone between ETs of 27° and 30° where impairment of performance *may* occur.

Wing (1965) calls attention to the exposure duration. He suggests the existence of an inverse exponential relation between heat exposure duration and the lowest temperature that yields significant impairment of mental performance. The relationship he proposes between performance impairment (as a function of ET) and exposure time is presented graphically in Figure 10.

Grether *et al.* (1971) examined the effects of combined heat, noise and vibration on human performance and physiological functions. Subjects were exposed to 31°C ET for 95 min, dressed in lightweight flight clothing. A performance decrement was observed at this ET compared with an ambient of 20°C ET; the addition of noise and vibration did not worsen this performance decrement.

The most recent research cited here is that of Nunneley *et al.* (1978) who examined subjects exposed to WBGTs of 28·7 and 31·1°C, for physiological and psychological effects. Subjects wore lightweight flight suits and helmets. A decrement in performance at both WBGTs occurred for

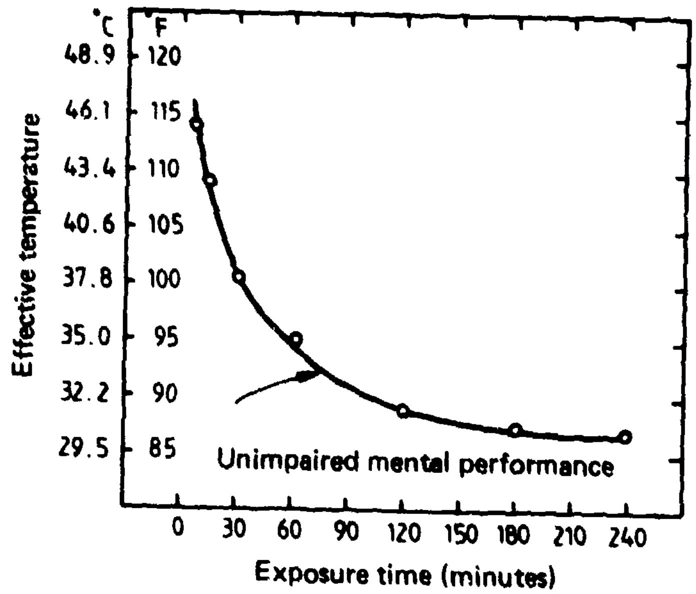

FIG. 10 PERFORMANCE LIMIT IN HOT ENVIRONMENTS
(reproduced from Wing (1965)).

most tasks except the number facility (simple arithmetic-type tasks) with which a person is familiar. In particular these tests indicated that learning may be highly sensitive to environmental conditions.

3.2.3.5 Australian Aircrew Experience of Heat Stress

There are several instances where the effects of a particular thermal environment on Australian military aircrew have been documented--for example flight trials in a Nomad aircraft (RAAF Institute of Aviation Medicine 1976), a Macchi aircraft (Aircraft Research and Development Unit 1969), and the CT4-A Airtrainer (Readett and Knights 1978).

From the Nomad flight trials it was subjectively judged that a WBGT of 28–30°C was the maximum level for effective aircraft operation. During one flight at 34–35°C WBGT, of duration 50 min under high humidity, a passenger was very close to collapse from heat exhaustion. Crew were dressed in light clothing.

During the Macchi flight trials, crew were dressed in the usual flying equipment, which includes anti-'g' suit, 'mae west', oxygen mask, and leather gloves. It was subjectively assessed that a WBGT of 25·5–27·0°C was the *maximum* comfortable for extended flight periods. WBGTs of 30°C and above were considered unacceptable for safe operation of the aircraft.

The conclusion reached from the CT4-A flight trials was that a degradation in pilot performance occurred when the cockpit WBGT exceeded 27°C. The CT4-A flight trials were carried out using a 50 mm black globe for measurement of the WBGT index, whereas the two earlier trials of the Macchi and Nomad used a 150 mm black globe. Thus, the WBGT of 27°C for the CT4 trials would be equivalent to approximately 28°C for the Nomad and Macchi trials.

3.2.4 Summary of Information on Heat Stress

The foregoing sections, 3.2.3.1 to 3.2.3.5, provide a considerable body of evidence that a reduction in mental and perceptual performance occurs at moderately high WBGTs of 27°C and above. In many of the instances subjects were wearing light clothing, in comparison with the ensemble worn by Australian aircrew (which frequently includes helmet, 'mae west', 'g' suit, oxygen mask, Nomex flying suit, and gloves). The body surface area available for evaporative cooling is thus very much reduced compared to some of the laboratory subjects. It is therefore likely that a reduction in human performance would occur before the level of 28–30°C WBGT found for most of the laboratory tests.

There is a dissenting viewpoint from the above, presented by Poulton (1976). He states that heat stress can reliably improve performance, and provides some evidence in support of his claim. However, the cases he cites were experiments of relatively short duration (less than 1 h) and, in most cases, for an ET of 30°C or less. Learning tasks were not included.

Grether (1973) suggests that an *initial* exposure to elevated temperatures would cause an increase in the general level of arousal, resulting in improved performance. With continued exposure impairment would be expected to occur. In view of the large body of evidence which shows both performance and physiological decrement at elevated temperatures, the view of Grether (1973) seems most probable.

An aspect of heat stress which is very relevant to aircrew is the decrease in acceleration tolerance and exaggeration of the effects of hypoxia. This aspect has not been explored here because of the limited scope of this report; it is, however, one which could be very important in high-performance aircraft and should be studied further. References to these two aspects are given by Nunneley and James (1977).

3.3 Physiological Heat Balance for Thermal Comfort

The heat balance for thermal comfort (Hughes 1968) can be basically expressed as

$$\underbrace{Q_r}_{\substack{\text{Radiation}\\ \text{exchange}\\ \text{with}\\ \text{surroundings}}} + \underbrace{Q_m}_{\substack{\text{Metabolic}\\ \text{heat output}}} + \underbrace{Q_{sp}}_{\substack{\text{Solar}\\ \text{radiation}}} = \underbrace{Q_{cl}}_{\substack{\text{Heat lost}\\ \text{by convection}\\ \text{from clothing}\\ \text{surface}}} \qquad (3)$$

and analytically as

$$3{\cdot}96 \times 10^{-8}\, \epsilon_{eff}\,[(T_w + 273)^4 - (T_{sk} + 273 - q_m/k_c)^4] + q_m + 0{\cdot}31\alpha_{cs}\,\tau\, Q_{st}$$
$$= (T_{sk} - q_m/k_c - T_e)\, h_a \qquad (4)$$

where ϵ_{eff} = effective emissivity from wall surface to pilot's clothing,

T_w = temperature of inner surface of cockpit wall (°C),

T_{sk} = pilot's mean skin temperature (°C),

T_e = mean temperature of the air in the immediate vicinity of the pilot (°C),

τ = mean transmissivity of the transparencies,

h_a = convective heat transfer coefficient (air–clothing surface) (W/m² °C),

k_c = clothing conductivity (W/m² °C),

q_m = pilot's metabolic heat load (W/m²),

α_{cs} = pilot's clothing absorptivity to solar radiation, and

Q_{st} = solar heat flux external to the cockpit (W/m²).

Aircrew are assumed to be thermally comfortable when their mean skin temperature is 33°C; representative values for h_a, k_c, and q_m can be found in Hughes (1968), and Billingham and Kerslake (1960*a*,*b*). Typical values for aircrew are $k_c = 11{\cdot}6$ W/m² °C (medium weight clothing), $h_a = 21{\cdot}5$ W/m² °C (where cabin air velocity is 3 m/s) and $q_m = 87$ W/m²; α_{cs} can be taken as approximately 0·5.

This equation does not take into consideration humidity, as it is a balance of *sensible* heats. Whilst evaporation contributes only approximately 20% to heat loss from a subject in comfort conditions (Billingham and Kerslake 1960*a*) this is none the less significant; use of Equation (3) will lead then to conditions cooler than the comfort level.

Equations (3) and (4) are quite inappropriate for conditions above the comfort level, where heat loss by evaporation is the dominant heat transfer mechanism.

This heat balance equation can be used in the design of an environmental control system, as demonstrated by Hughes (1968); an example of this is given in Appendix 3.

3.4 Liquid Conditioned Garments

A typical liquid conditioned garment for aircrew use is a close fitting full length undergarment. The heating or cooling fluid is circulated through tubes threaded through tunnels in the garment. The liquid conditioned garment is very much more effective than an air ventilated suit; because the heat transfer capacity of liquids is higher (for a given volume) than for air, the garment is less bulky and pumping power requirements to circulate the working fluid are smaller.

The overall cockpit cooling requirements when a person is wearing a liquid cooled suit are very much smaller than when using the usual cockpit cooling air for crew cooling; for example the conditioned suit cooling requirement at 60°C ambient temperature for aircrew dressed in normal flight ensemble, and working steadily, is 0·45 kW. The cockpit air cooling requirements, in comparison, would range from 5 to 10 kW. In the UK, liquid conditioned suits have been developed by the Royal Aircraft Establishment in conjunction with Beaufort (Air–Sea) Equipment Ltd., the latter organisation manufacturing the suits. A liquid conditioning system for the suits has been developed by Delaney Gallay (UK); it provides heating and cooling, weights 7 kg, and has dimensions 130 mm by 90 mm by 326 mm. It can be fastened to the side of an ejection seat.

The liquid cooled suit cannot be seen as a complete substitute for cockpit air conditioning. Hughes (1974) outlines some of the limitations of a liquid cooled suit; these are:

(*a*) surfaces that are to be touched still need to be maintained at less than 45°C;

(*b*) pressurization and ventilation of the cockpit are still required;

(*c*) cooling is required for cockpit-mounted electronic components;

(*d*) canopy materials for an uncooled cockpit may have temperatures so high that the maximum allowable stress has to be reduced;

(*e*) cabin discharge air is not available to cool equipment compartments.

It is most likely then that the usefulness of a liquid cooled suit is in bringing about a sizeable *reduction* in the cooling air requirements.

There is very much more which could be said about liquid conditioned suits, as a considerable body of experimental data has been amassed by the Royal Aircraft Establishment. However, the human behaviour aspects of these garments are currently being studied by Cybernetics Group (Systems Division) of ARL, and it is therefore appropriate to leave further comment to them.

3.5 Environmental Requirements for Electronic Equipment

At a symposium on the cooling of high performance aircraft, several papers were presented on the cooling of electronic equipment—German (1974), Freeman and Price (1974), and Hughes (1974). The reliability of electronic components was shown to be heavily dependent on temperature; reliability increases significantly down to 0°C—see Figures 11(a) and 11(b). However, because of limitations that exist on cooling air flow, in practice an upper limit to the outlet temperature of the equipment cooling air is set usually in the region of 60°C.

Temperature-pressure specifications for airborne electronic equipment are contained in the United States Navy specification MIL-E-5400P, and the British Standard for equipment in aircraft—3G.100: Part 2: Section 3: Subsection 3.2, October 1970.

Discussion of design problems in cooling electronic equipment is outside the scope of this report—the three papers referred to above provide some insight into current work.

4. THERMAL LOAD ON AIRCRAFT CABINS

4.1 Cabin Heat Balance in Flight

The cabin heat balance determines the crew environment, which, as discussed in Section 3, can then be expressed as an index such as the WBGT or ET, or can be used in conjunction with the crew heat balance (Section 3.3) to determine cooling requirements.

The heat balance for avionic compartments is not specifically discussed here; the same principles will apply, except that altitude effects may differ, and the dry bulb air temperature only may be of interest (presuming the absence of liquid water).

4.1.1 Fixed Wing Military Aircraft

The heat balance for fixed wing military aircraft has been analysed in depth by several workers—two prominent examples are Torgeson *et al.* (1955), and Hughes (1961, 1963*a*,*b*, 1968).

Torgeson *et al.* (1955) describe, and illustrate by example, a method of calculating heat flow into a cabin, where a detailed knowledge of the aircraft structure is available. The analysis considers both steady-state air conditioning load, and transient heating of an aircraft cabin.

The work of Hughes was carried out at the Royal Aircraft Establishment and relates to cooling tests of full-scale aircraft cabin mock-ups in a closed-circuit hot-air tunnel. The reports

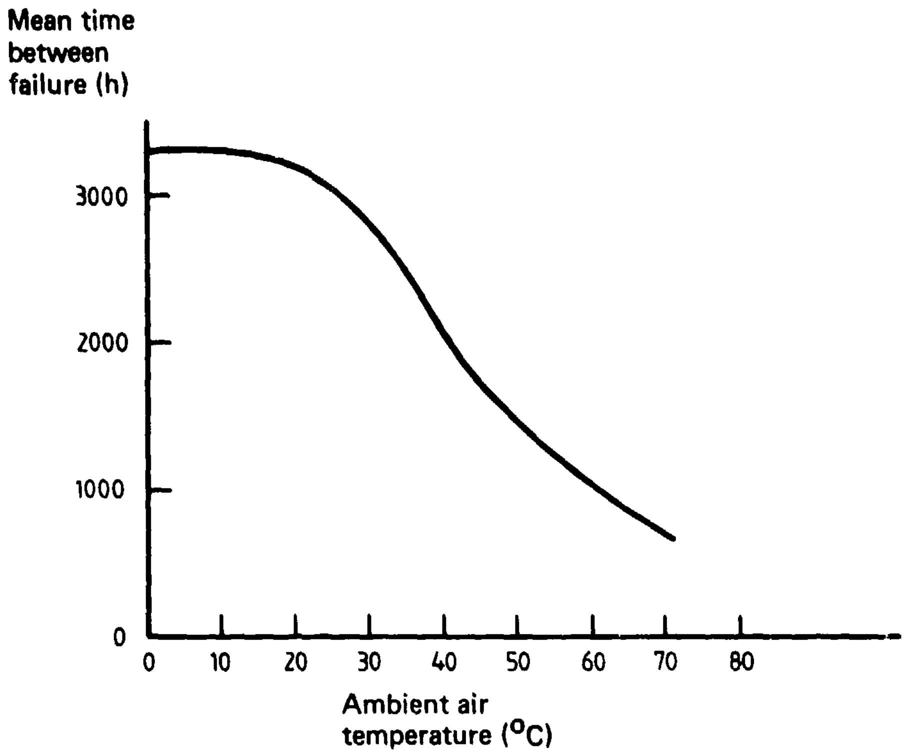

FIG. 11(a) TYPICAL CURVE OF MEAN TIME BETWEEN FAILURE OF ELECTRONIC COMPONENTS VERSUS OPERATING AMBIENT TEMPERATURE. REPRODUCED FROM HUGHES (1974)

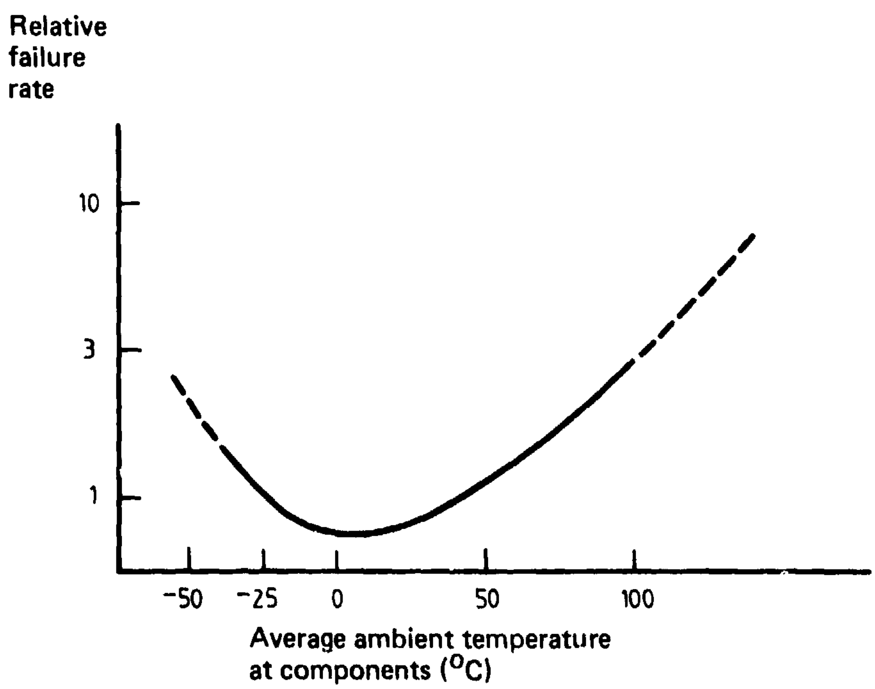

FIG. 11(b) RELATIVE FAILURE RATE OF TYPICAL ELECTRONIC EQUIPMENT WITH TEMPERATURE. REPRODUCED FROM GERMAN (1974) of Hughes (1961, 1963*a*) described tests of a Bristol 188 Research Aircraft cockpit and cooling system, Hughes (1963*b*) described the development of an air distribution system for the TSR-2 cockpit, including an analysis of cabin heat loads for these tests. Hughes (1968) gave an overall summary of cabin air requirements for crew comfort in military aircraft. In this work the earlier analyses of heat balance had been refined to the point where the design curves and computer program presented enabled estimation of the cabin cooling air requirements under various flight conditions, without requiring a detailed knowledge of the aircraft structure.

Earlier studies on cabin air conditioning in high speed flight, carried out at the Royal Aircraft Establishment, were by Davies (1953), on general heat balance considerations for flight between M 0·5 and M 2·0; the Ministry of Supply Working Party (1956) on the internal cooling of high speed supersonic military aircraft; and by Rudman *et al.* (1959) on cooling problems (including heat balance considerations) for supersonic civil transport aircraft. More recently Hughes and Timby (1968) described cabin conditioning tests on a simulated M 2·2 aircraft cabin, which included evaluation of air distribution schemes, insulation and cooling requirements.

Other studies at the Royal Aircraft Establishment related to cabin heat balance, were the reports on skin temperatures reached in high speed flight by Davies and Monaghan (1952), Monaghan (1954), and Rendel *et al.* (1954); a study of the thermal properties of aircraft cabin insulation using electrical analogue techniques by McNaughton and Hughes (1962); and development of a method of simulating solar radiation through transparencies in cabin conditioning tests by Hughes (1969).

Examples of heat balance calculations applicable to aircraft in Australian military service are the analysis for the F86E Sabre aircraft by North American Aviation (1951) and that for the Macchi MB326H jet trainer by Lynch and Beenham (1968), and Beenham (1969). The studies relating to the Macchi were carried out in response to RAAF proposals to improve the effectiveness of the Macchi cooling system.

From a paper presented by Lowery and Howells (1974) at a symposium on the cooling of high performance aircraft, held at the Royal Aircraft Establishment, Farnborough, the total cooling requirements for the modern generation of military aircraft (11 000–23 000 kg) is approximately 30 kW, of which approximately 10 kW would be required for the cabin. This figure of 10 kW is close to advice received from the Royal Aircraft Establishment (Rebbechi 1976) where the heat input for a standard two-man cockpit of a military aircraft, flying at M 0·9, is estimated to be:

(*a*) skin heat transfer 6–8 kW;

(*b*) solar heating (canopy) 2 kW; and

(*c*) avionic equipment (inside cockpit) 1·5 kW.

To give a practical example of an aircraft heat balance, the results are given here of a calculation based on the Macchi MB326H jet trainer, at the following conditions:

(1) outside air temperature 43°C;

(2) cloudless sky;

(3) Mach number 0·6;

(4) low altitude;

(5) two crew members;

(6) cabin WBGT 28°C.

The details of the calculations are given in Appendix 3. The relative magnitudes of the various terms in the heat balance equation are:

Heat flow through aircraft skin	+	Solar radiation	+	Pilot's metabolic output	+	Electrical load	=	Heat out in cooling air	
1950	+	1570	+	240	+	500	=	4260,	(5)

where the units are watts. Also in Appendix 3 a cabin heat balance is carried out for the crew in a state of thermal comfort, where the physiological heat balance equation, (4), is also utilised. For this latter case, the cooling requirements indicated by Equation (5) are greater, the estimate from Appendix 3 being 4960 W.

4.1.2 Fixed Wing Light Aircraft

Trials by the RAAF Aircraft Research and Development Unit (Readett and Knights 1978) of the CT4-A Airtrainer have shown that a crew heat stress problem can exist in light aircraft, even in quite moderate ambient temperatures. The severity of the problem depends considerably on the configuration of the aircraft. The CT4-A, with a 'bubble' type canopy which is not intended to be opened in routine flight, is very unfavourable with regard to heat stress as the crew receive direct solar radiation, and ventilation is limited. Other aircraft types, in which the canopy can be slid back and latched whilst in flight, or with a high-wing configuration where the crew may be shielded from direct solar radiation, will have appreciably lower cockpit temperatures. The cabin cooling requirements of light aircraft can be assessed by the methods described in the previous section, 4.1.1.

Various aspects of the cooling of light aircraft are discussed by Aarons (1970). As the majority of light aircraft are powered by piston engines, and are unpressurised, the usual choice of cooling system is a vapour cycle system based on automotive-type components, the refrigerant compressor being either directly driven from the engine. or from an electric motor.

4.1.3 Rotary Wing Aircraft

As with fixed wing light aircraft, the heat balance would follow the general principles of Section 4.1.1, but with several important differences as evidenced by recent air conditioning trials with the Royal Australian Navy Sea King. The heat balance for the RAN Sea King is analysed in detail by Rebbechi (1979, 1980); it differed significantly from that for fixed wing aircraft with regard to heating of the cabin due to hot exhaust gas impinging on the fuselage skin (the engines are situated above the cabin), direct conduction from the engine and main gearbox area, and the presence of strong hot air inflows to the cabin from the engine/gearbox area—discussed by Rebbechi (1977). The cabin cooling requirements to bring the crew environment to an acceptable level in high ambient temperatures were estimated to be between 12–15 kW, depending on electrical heating from cabin avionics. A cabin environment survey of RAN Sea King helicopters by Rebbechi and Edwards (1979), and of Royal Navy Sea Kings by Lovesey *et al.* (1976) showed that where the cabin is ventilated only by ambient air, cabin temperatures vary between 3°C and 12°C above outside air temperature.

An extensive survey to estimate cabin heat transfer coefficients of radiation, conduction and convection in rotary wing aircraft has been carried out by Laing (1974). The object of this survey was to determine static and in-flight transient and steady-state interior temperatures of an uncooled cabin. A number of United States helicopters was surveyed—the CH-47C, UH-1H, CH-54B, OH6A and AH1G. The CH-47 (Chinook) survey is described by Laing *et al.* (1975); references to the other helicopter surveys are contained within that report. The heat transfer coefficients derived in the surveys of Laing are not directly applicable to the heat balance of a *cooled* helicopter.

4.2 Ground Cooling of Aircraft

4.2.1 General Comments

When the aircraft is parked without the main engines operating, the cabin environment is dependent either on natural ventilation with ambient air, electrical/hydraulic power to an in-built cooling unit, or cooling air supplies from an external portable cooling unit. Without cooling, cabin temperatures can be quite high; for example, 22°C above outside air temperature (OAT) for the Buccaneer aircraft (Harrison and Higenbottom 1977), 20°C above OAT for the Sabre aircraft (Cameron and Cumming 1963) and in the range 25–33°C above OAT for rotary wing aircraft (Laing 1974).

It is obviously advantageous to shade the aircraft when it is parked and exposed to solar radiation. Although it is largely self evident that shading will bring the cockpit closely to ambient temperature, quantitative verification of the merits of shading are given by Shpilev and Kruglov (1974) and Repacholi and Rebbechi (1980).

Where the engines are operating at idle power settings a limited degree of cooling is available in the case of an air cycle cooling system, and (usually) full cooling is available in the case of electrically or hydraulically powered vapour cycle systems. It is highly desirable that the canopy can be latched partially open when the aircraft is taxiing, as the cooling effect provided by an air cycle system may be very small with the aircraft engine operating at low taxiing power.

4.2.2 Cooling Requirements for a Parked Aircraft

4.2.2.1 Fixed Wing Aircraft

The heat balance for a typical parked fixed-wing aircraft, one to two-crew cabin, bubble-type canopy, can be written in a simplified form as

$$h_{wo}(T_a - T_m) + Q_s = Q_c, \tag{6}$$

where h_{wo} = overall heat transfer coefficient (W/°C),

T_a = outside air temperature (°C),

T_m = cockpit mean air temperature (°C),

Q_s = solar radiation transmitted through transparencies (W),

Q_c = cooling effect (W).

The coefficient h_{wo} in Equation (6) includes long wavelength (near-ambient temperature) radiation exchange with surroundings, and the outer skin convective heat transfer coefficient (which is dependent on wind velocity and direction). The solar radiation transmitted through the transparencies is given by

$$Q_s = Q_{si}\, A\, \tau \sin \theta, \tag{7}$$

where Q_{si} = incident solar radiation (W/m²),

A = plan projected area of transparencies (m²),

τ = transmittance of the transparencies,

θ = altitude of the sun (angle above the horizontal plane).

Tests, carried out by Repacholi and Rebbechi (1980) on a parked Sabre aircraft, have shown that Equation (6) can be written, for a Sabre aircraft parked in light winds, as

$$57{\cdot}5(T_a - T_m) + 1020 = Q_c. \tag{8}$$

Higher wind velocities will increase cooling requirements up to a maximum of

$$96(T_a - T_m) + 1020 = Q_c; \tag{9}$$

the value of $h_{wo} = 96$ in Equation (9) being obtained from an analysis of the Sabre cabin heat balance by North American Aviation (1951).

4.2.2.2 Rotary Wing Aircraft

The heat balance for parked rotary wing aircraft differs a little from that for fixed wing aircraft (Equation (6)). Depending on the helicopter type the cabin may be largely encased by transparencies, as in the case of the Iroquois UH1-G and Bell Jet Ranger, or a sizeable part of the cabin may be covered by an opaque fuselage skin as in the Sea King Mk 50, Wessex and CH-47 (Chinook). In these latter aircraft the solar heat loads on the fuselage skin can then be a significant part of the aircraft heat balance. This case was analysed by Rebbechi (1980) where the heat balance and cooling requirements were obtained for a parked RAN Sea King Mk 50 helicopter. The heat balance is sufficiently general for use in analysis of other aircraft types where transparency areas are relatively greater.

4.3 Air Distribution and Velocity

The air distribution, ventilation and velocity *requirements* are well described by such specifications as the United States Air Force Systems Command Design Handbook 2–3 (Design Note 4A1), and the Military Specifications MIL-E-38453A (USAF), and MIL-E-18927D. These requirements are quite exacting, particularly with regard to temperature differences throughout the cabin; for example MIL-E-38453A requires that the temperature variation between any two points in the envelope occupied by seated personnel shall not deviate more than 2·8°C from the average compartment temperature. In practice it is unlikely that a requirement as stringent as this could be met where there are overhead transparencies (in view of the high solar heat load on the upper areas of the crew), although it is a desirable aim.

A comprehensive description of the development of a cooling air distribution system is given by Hughes (1963*b*) where he describes the development at the Royal Aircraft Establishment of an air distribution system for the TSR-2 pilot's cockpit. This report gives considerable insight into the problems encountered in developing an effective air distribution scheme. Eight different schemes were tested; in Figure 12 the initial and final schemes are shown, and in Figure 13 another illustration of the final version (scheme 8) is shown. Figure 13 gives a good indication of the air distribution necessary in a single-seat cockpit to optimise the crew cooling available from the cooling air supplies, and to minimise temperature gradients in the cockpit. In contrast to the TSR-2 distribution system of Figure 13, the air distribution system for the Macchi MB326H is sketched in Figure 14. The lack of cooling air outlets at head and shoulder level could be expected to result in high head level temperatures, and, because the cabin air exit is located in the forward part of the front cockpit area, the rear crew member is seated in largely stagnant air. The air temperature in the rear cockpit will then be appreciably greater than in the front cockpit.*

An important feature of the tests carried out by Hughes (1963*b*) on the TSR-2 cockpit air distribution was that a full-size replica of the cabin was placed in a hot-air tunnel. The outer skin of the simulated cockpit could then be brought to a representative temperature for a particular flight Mach number. The placement of a simulated cockpit in an environmental test chamber in which the outside air is *stationary* will not yield a skin temperature representative of flight conditions. Despite this fundamental error in the use of still ambient air for cabin environment tests, this procedure has been used in acceptance tests carried out for the F-111A environmental control system (McSwain 1971).

Various recommendations exist for cabin air velocities. There is general agreement that air velocities over face and exposed areas of skin should be no greater than 1 m/s; however, over clothed regions of the crew, air velocities up to 3 m/s are acceptable (Hughes 1968). The specification MIL-E-38453A requires that the airflow moving past crew is not to exceed 1·5 m/s.

Fresh air ventilation requirements per crew member range from 4 g/s (British Civil Airworthiness Requirements, Chapter D6-11, 1976), to 11 g/s (MIL-E-38453A, AFSC DH 2–3 (DN 4A1)). In a report on airflow rate requirements for passenger aircraft, Timby (1969) states

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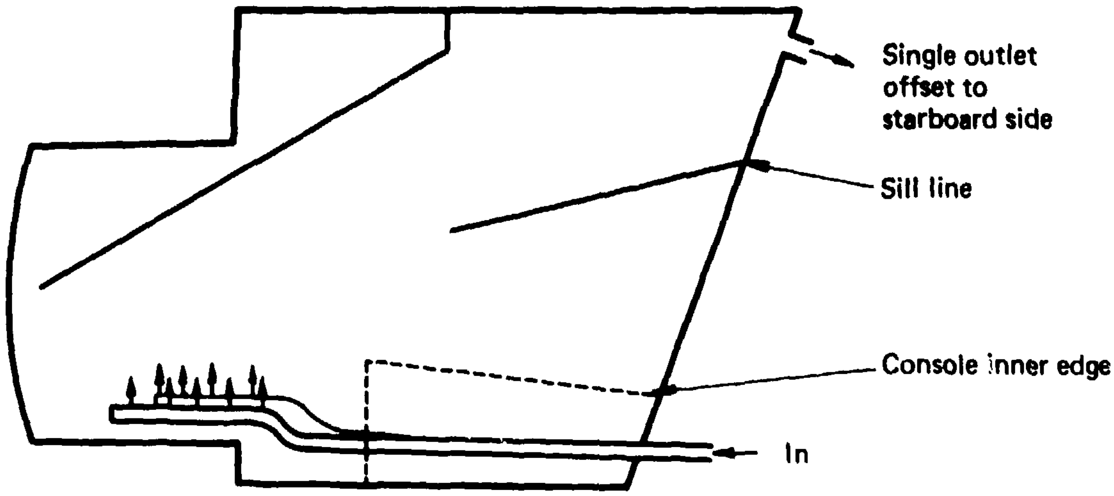
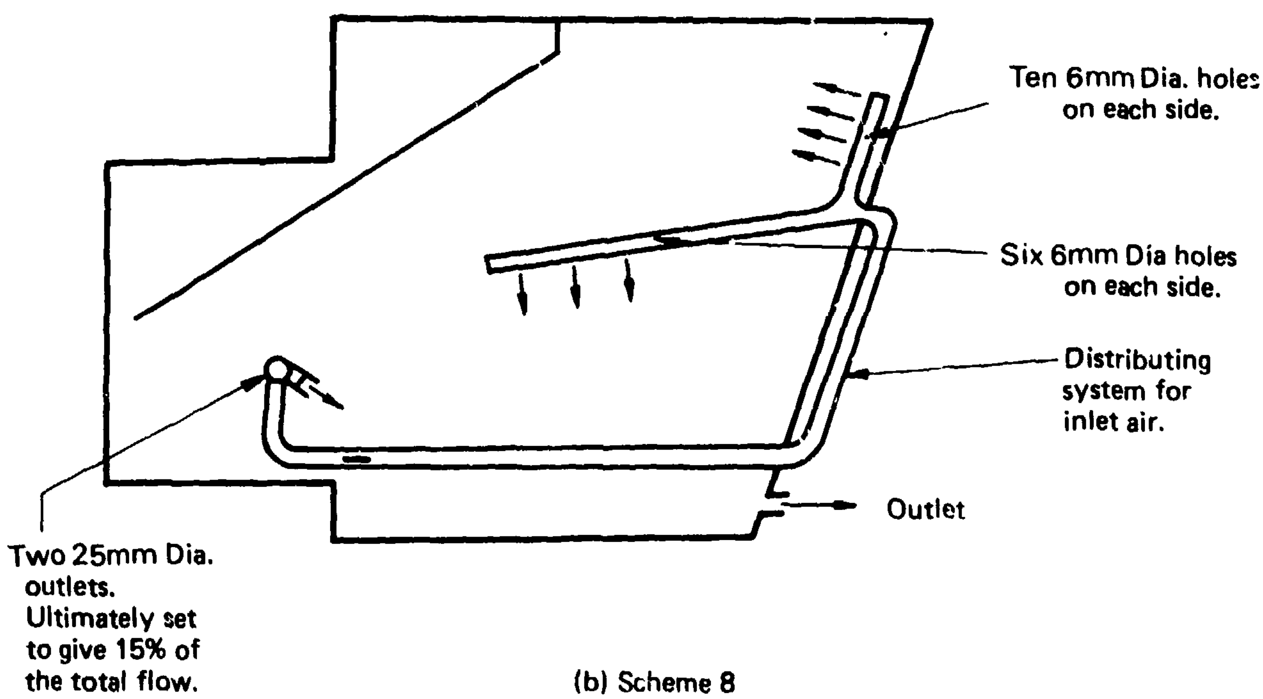
* Flight trials carried out by the Aircraft Research and Development Unit (1969) confirm this differential between front and rear cockpit air temperatures, however improvement in the air distribution would only be a partial solution to this particular cooling system problem, as increased cooling air mass flow is also required.



(a) Scheme 1



(b) Scheme 8

FIG. 12 INITIAL AND FINAL COCKPIT AIR DISTRIBUTION SCHEMES FOR TSR-2 COCKPIT (REPRODUCED FROM HUGHES 1963B)

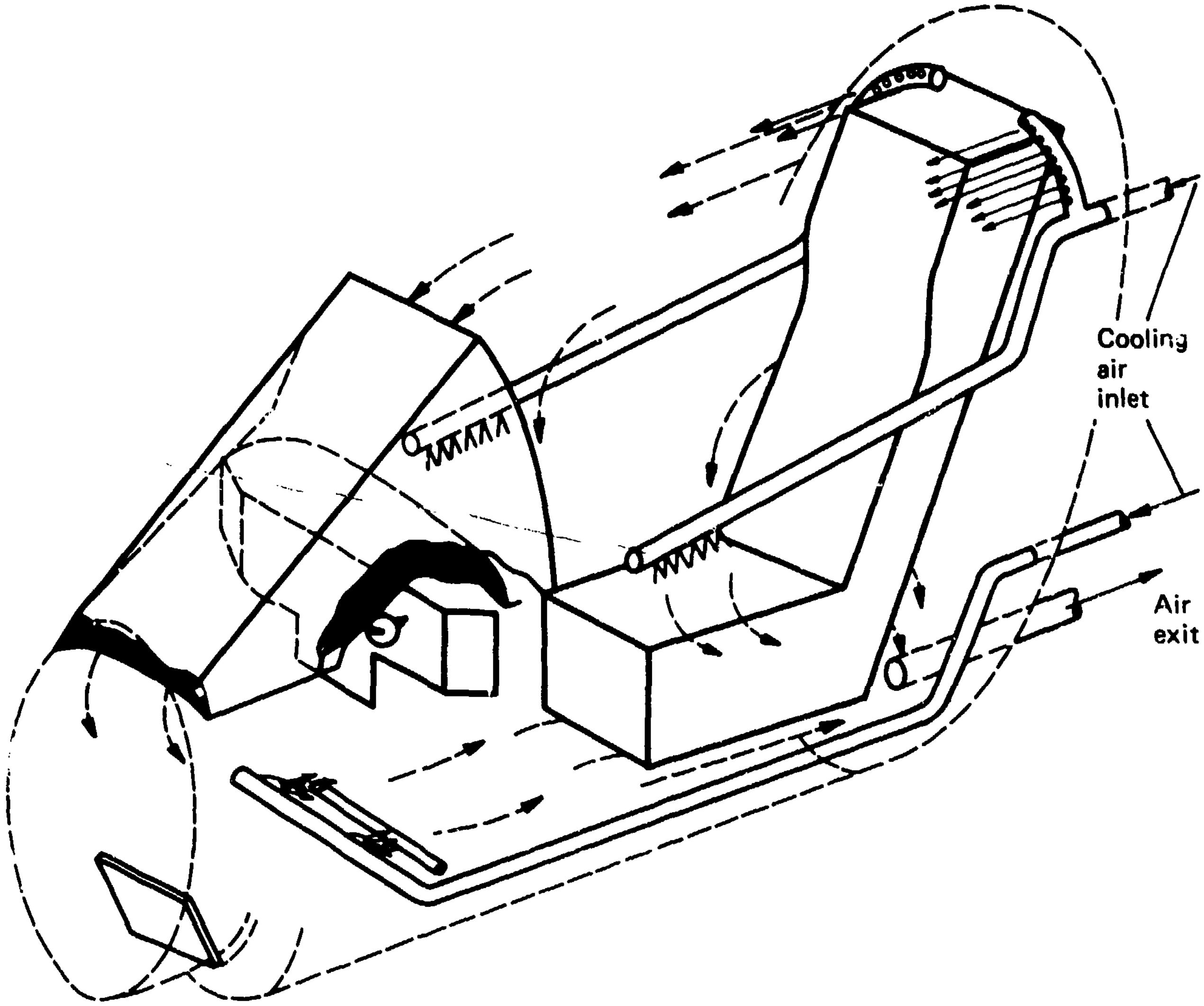


FIG. 13 FINAL VERSION OF COOLING ARRANGEMENT FOR TSR-2 COCKPIT (HUGHES 1968)

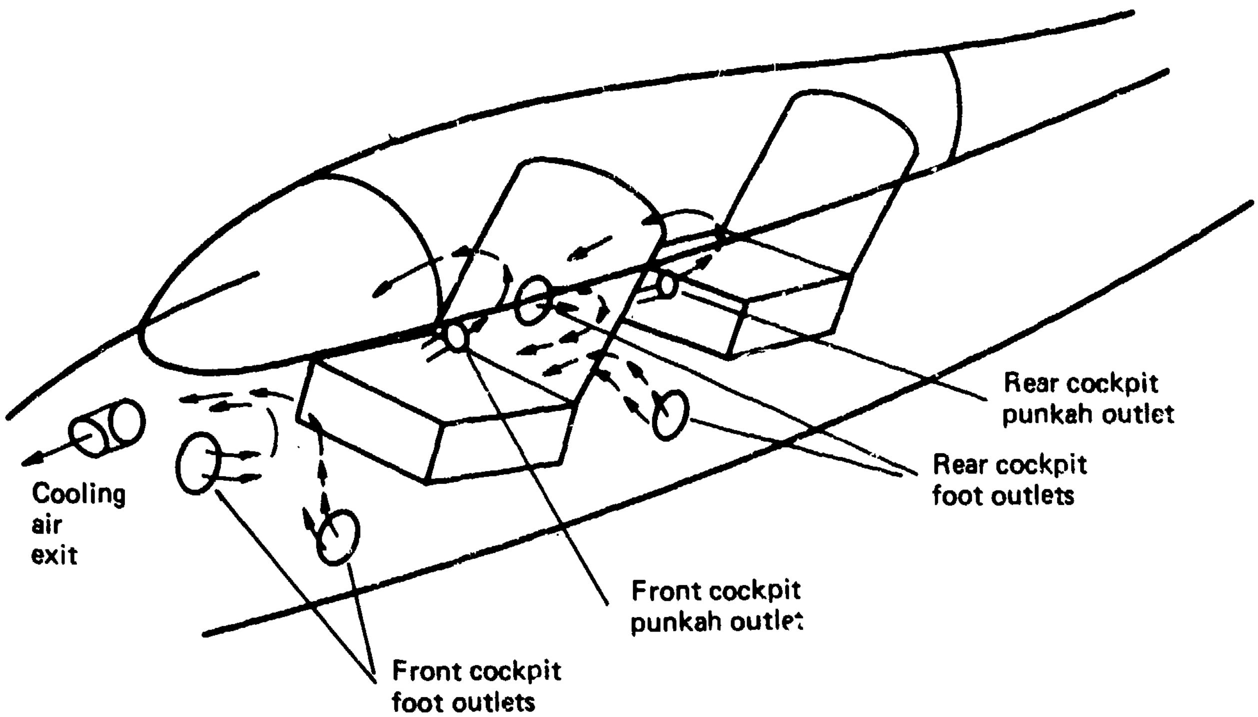


FIG. 14 MACCHI MB326H COOLING AIR DISTRIBUTION

that from a purely breathing aspect, fresh air supplies could be reduced to 2 g/s. The minimum airflow rate into the crew passenger compartments of pressurized aircraft will in any case be limited by the leakage rate, and the need to provide increasing cabin pressure when descending. These two aspects are discussed in some detail in the AFSC DH 2-3 (DN 4A.), and referred to also in MIL-E-38453A.

4.4 Aircraft Transparency Misting

Aircraft transparency misting has been a serious problem in several RAAF aircraft; it can be particularly acute when descending from high altitude (where the transparencies are cold) to the warm moist air of the tropics. Misting occurs when the inner surface of the transparency is at a temperature less than or equal to the dew point of the cabin air.

The two common methods of transparency demisting are to blow cabin conditioning air over the interior surfaces, or to heat the transparencies by means of an electrically heated conducting film or wire grid embedded in the transparency.

4.4.1 Air Demist Systems

The requirements of the air demist system are at times contradictory to the overall cockpit requirements; for the air demist system to work effectively the air must be sufficiently hot to bring the transparency inner surface above the cabin air dew point, however, this will have the effect of heating the cockpit overall at a time when the aircraft may be descending into a warm humid outside environment. This problem is described in the reports of the Aircraft Research and Development Unit (1968*a*,*b*) on modifications to the canopy demist system of the Mirage III0. Hughes (1971) has made a theoretical assessment of the effectiveness of the internal air demist system as applied to an aircraft during descent after high altitude cruise. The analysis includes the transient heat conduction through the transparencies; a computer program is described which facilitates the theoretical analysis, so that the effectiveness of various combinations of flight conditions and demist layout can be readily assessed.

A lowering of the cabin air dewpoint will of course result in a reduction in the transparency temperature at which misting occurs; recent developments of air cycle cooling units, such as that for the Westland 'Lynx' helicopter (Giles 1977) have brought about a low cooling turbine outlet temperature (−40°C). The dew point of the cabin air can then be very low, depending on the degree of mixing with air taken from upstream of the turbine (for temperature control purposes). With this cooling system much of the water extraction from the cabin air takes place at high pressure in a heat exchanger prior to turbine entry, so that a supply of dry air at moderate temperatures is available for demist purposes.

4.4.2 Electrically Heated Transparencies

Electrically heated transparencies are quite widely used in civilian aircraft. This method of heating the transparencies will add only a minimal additional heat load to the cockpit, unlike the air demist system. An electrically conductive film can be incorporated in curved canopies, suitable for use up to $M\,2{\cdot}5$ (Aircraft Engineering 1963); electrically heated windscreens for the Mirage III0 were shown in flight trials to be very successful (Aircraft Research and Development Unit 1968*a*,*b*), and very much more effective than the hot air canopy demist system simultaneously being evaluated.

It is usual for the temperature of the windscreen to be maintained at a predetermined level by a temperature controller; this control of temperature can also assist in maintaining the screen at a temperature that will ensure adequate impact strength, which is necessary to provide resistance to bird-strike damage (Pavia 1975).

5. OPEN AIR CYCLE COOLING SYSTEMS

5.1 Design

5.1.1 General Principles

The open air cycle system is at present the most commonly used cooling system type in aircraft. It is termed an open cycle, as the air is passed only once through the cooling system, through the cabin, and then exhausted to atmosphere. The two frequently used system configurations are the turbofan (as in the Sabre Mk 31, F-111C, and Macchi MB326H) and the bootstrap system (Mirage IIIO). The two systems are sketched in Figures 15 and 16. There are quite a number of variants of these two basic types in use at the present time, such as the three-wheel bootstrap system and recirculation air cycle unit (termed 'shoe-string' by the manufacturers, Normalair-Garrett Ltd.).

5.1.2 Review of Publications

General design princples for air cycle systems are described in such references as the United States Air Force Systems Command Design Handbook. The McDonnell Aircraft Company (under a USAF contract) has published an extensive study of environmental control systems for aircraft. This study includes both air and vapour cycle cooling systems; it is published in four volumes (Diekmann *et al.* 1972; Whitney *et al.* 1972*a,b*; Glover *et al.* 1972) and is essentially a computer program to predict system performance and estimate weight and reliability. An earlier report on aircraft equipment cooling systems is that of Zimmerman and Robinson (1954), which was intended to provide a guide to the analysis and selection of cooling systems for aircraft, and includes discussion of expanded ram-air systems. A theoretical analysis of various air cycle system configurations is given by Stevens (1956), and Thomas (1948); a more general discussion of aircraft air conditioning systems is provided by Still (1957) and Sanders (1970). Description of air cycle systems pertaining to particular aircraft are given by Gregory (1968), Sherbourne (1973), Trebosc (1967), Livingstone (1969) (these four describing the Concorde air-conditioning); Gregg (1957)—the Boeing B-47; Hauger *et al.* (1968)—the Douglas DC-9; Aircraft Engineering (1974)—the Lockheed S-3A Viking; Aircraft Engineering (1970)—the Hawker Siddeley Harrier.

5.2 Cooling System Performance

5.2.1 Theoretical Coefficient of Performance

5.2.1.1 Turbofan System

The temperature–entropy diagram for a turbofan system is sketched in Figure 17. The expansion work (Q_{34}) is not subtracted from the compression work (Q_{12}), but is used to drive a fan which induces a flow of air through the heat exchanger (Fig. 15). In this application only a small fraction of the energy supplied to the fan is required to provide cooling air for the heat exchanger;* the fan is mostly an energy absorber. The theoretical coefficient of performance (COP) of the turbo-fan system is:

$$\mathrm{COP} = \frac{\text{cooling effect}}{\text{nett work input}} \tag{10}$$

$$= Q_{14}/Q_{21}$$

$$= (T_1 - T_4)/(T_2 - T_1). \tag{11}$$

From equation (11) it may seem that quite high COPs could be obtained; however, practical considerations such as a lower limit to T_4 (as discussed in Section 5.2.4), a need to minimise system size and weight and therefore maximise the cooling effect per unit mass of working fluid, and the need usually to maintain T_4 above 0°C to avoid icing problems, result in a theoretical

* Messinger (1946) illustrated this when he described a system where the fan power required for cooling the heat exchanger was 3·7 kW, compared with the available turbine power of 54 kW.

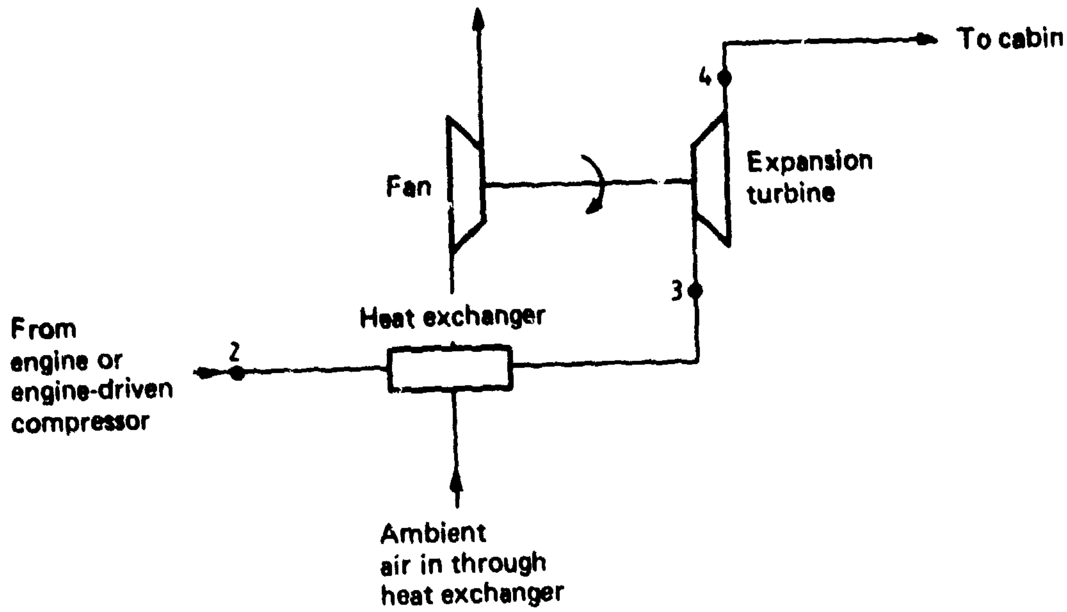


FIG. 15 TURBOFAN COOLING SYSTEM

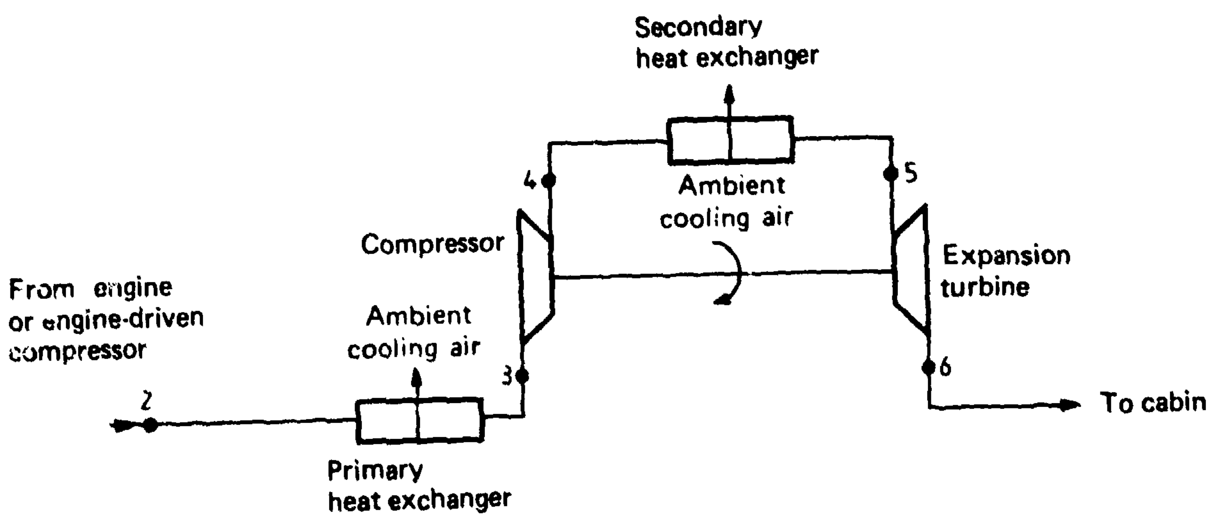


FIG. 16 BOOTSTRAP COOLING SYSTEM

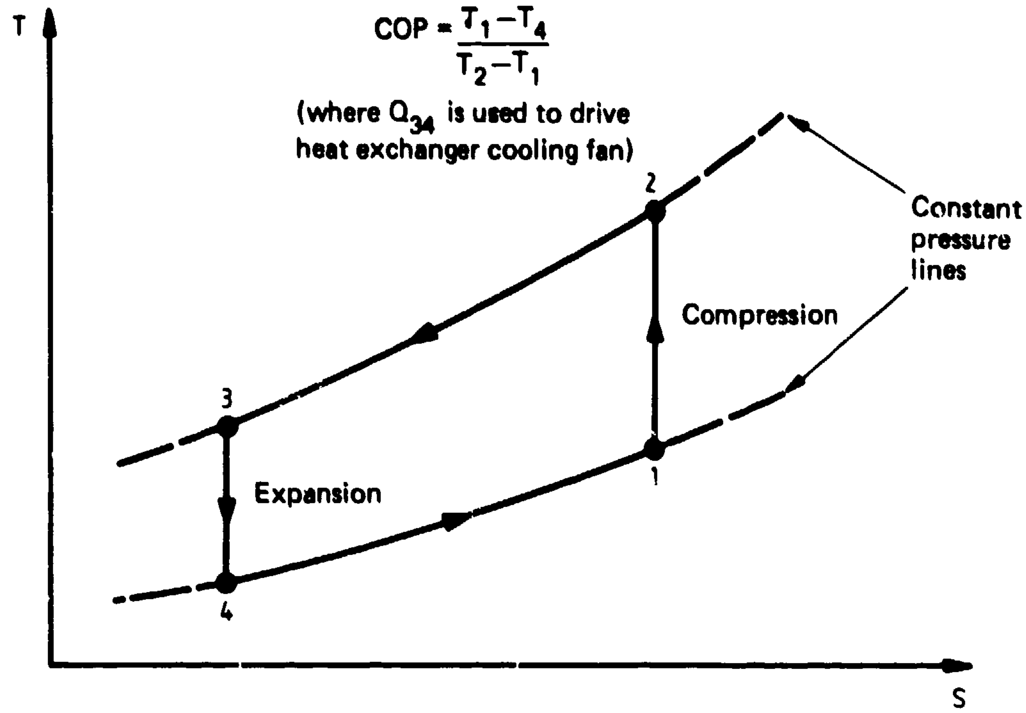


FIG. 17 TEMPERATURE-ENTROPY DIAGRAM FOR TURBOFAN SYSTEM

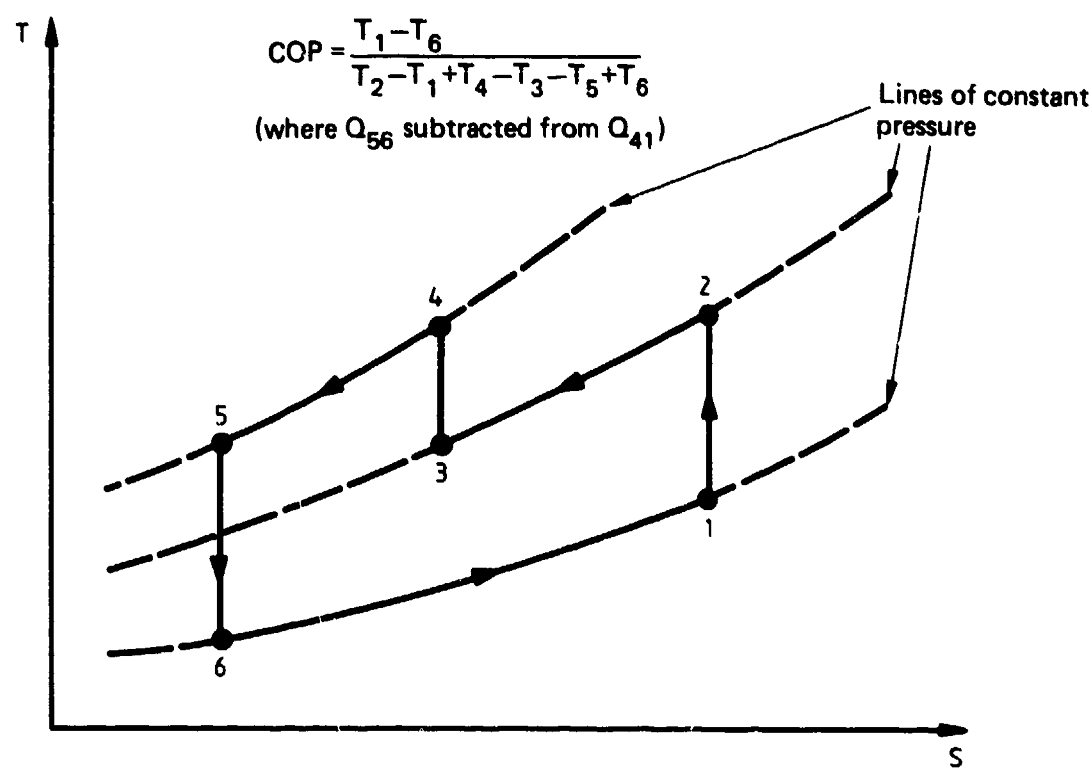


FIG. 18 TEMPERATURE-ENTROPY DIAGRAM FOR BOOTSTRAP SYSTEM

COP less than 0·35. Where T_4 can be brought below 0°C, as in a system developed by Normalair-Garrett (Giles 1977) for the 'Lynx' helicopter (where $T_4 = -40$°C), then the theoretical COP for a cycle pressure ratio of 3 may rise to 0·7.

5.2.1.2 Bootstrap System

The temperature–entropy diagram for the bootstrap system is shown in Figure 18; the only immediate apparent difference from the turbofan system is inter-cooling in the compression phase (1–4), which lowers the compression work. However, the turbine work Q_{56} is also subtracted from the compression work. The theoretical COP is:

$$\text{COP} = (T_1 - T_6)/(T_2 - T_1 + T_4 - T_3 - T_5 + T_6). \tag{12}$$

This unwieldy expression can be simplified by considering only two special cases, as shown in Figure 19. Where there is no inter-cooling (1–2–3–4–5–6) the theoretical COP is

$$\text{COP} = (T_1 - T_6)/(T_4 - T_1 - T_5 + T_6), \tag{13}$$

and for isentropic compression and expansion,

$$T_4/T_1 = T_5/T_6 \tag{14}$$

$$= r_p^{(\gamma-1)/\gamma}, \tag{15}$$

where r_p is the overall pressure ratio p_4/p_1, and γ the ratio of specific heats. Then,

$$\text{COP} = T_1/(T_4 - T_1), \text{ or alternatively} \tag{16}$$

$$\text{COP} = T_6/(T_5 - T_6) \tag{17}$$

$$= 1/(r_p^{(\gamma-1)/\gamma} - 1). \tag{18}$$

The COP is plotted versus pressure ratio, r_p, in Figure 20. It is much greater than for a turbo-fan system, which as seen from the previous section is in the region of 0·35–0·7 for a system pressure ratio of 3.

Where there is a large degree of inter-cooling (1–2–3′–4′–5–6 on Figure 19), the theoretical COP is then $T_1/(T_2 - T_1)$, alternatively $T_6/(T_{3'} - T_6)$. The COP is then a function of the pressure ratio p_2/p_1 (Fig. 19), and as the overall pressure ratio is r_p, it will appear as the dashed line in Figure 20. Inter-cooling to intermediate temperatures between T_2 and $T_{3'}$ will result in the COP lying between the full and dashed lines of Figure 20.

5.2.2 Effect of Compressor and Turbine Inefficiencies

The effect on cycle performance of inefficiencies can be seen by reference to Figure 21; the work input Q_{12} is increased, the cooling effect Q_{41} decreased, and the turbine work Q_{34} decreased. The effect of these inefficiencies is to decrease considerably the COP of both the turbofan and bootstrap systems. The extent of this decrease can be seen from Figure 22, where the COP is plotted for bootstrap and turbofan systems; a comparison is made between the theoretical case with efficiencies of 1·0, and the case with realistic turbine and compressor efficiencies. Details of the calculations leading to Figure 22 are given in Appendix 4.

From Figure 22 the suitability of a cooling system type for a particular engine can be readily seen; it is immediately apparent that the turbofan system, while adequate for the F-111C range of engine pressure ratios, is quite inadequate for the Macchi. For engines of low pressure ratio a bootstrap system is very much more effective.

5.2.3 Temperature of Heat Rejection

Heat rejection takes place between points 4 and 5 (Fig. 19). A lower limit to T_5 is imposed by the available heat sink temperature, which for ram-air heat exchangers is the stagnation temperature of the outside air stream. As this temperature can be quite high, particularly so in

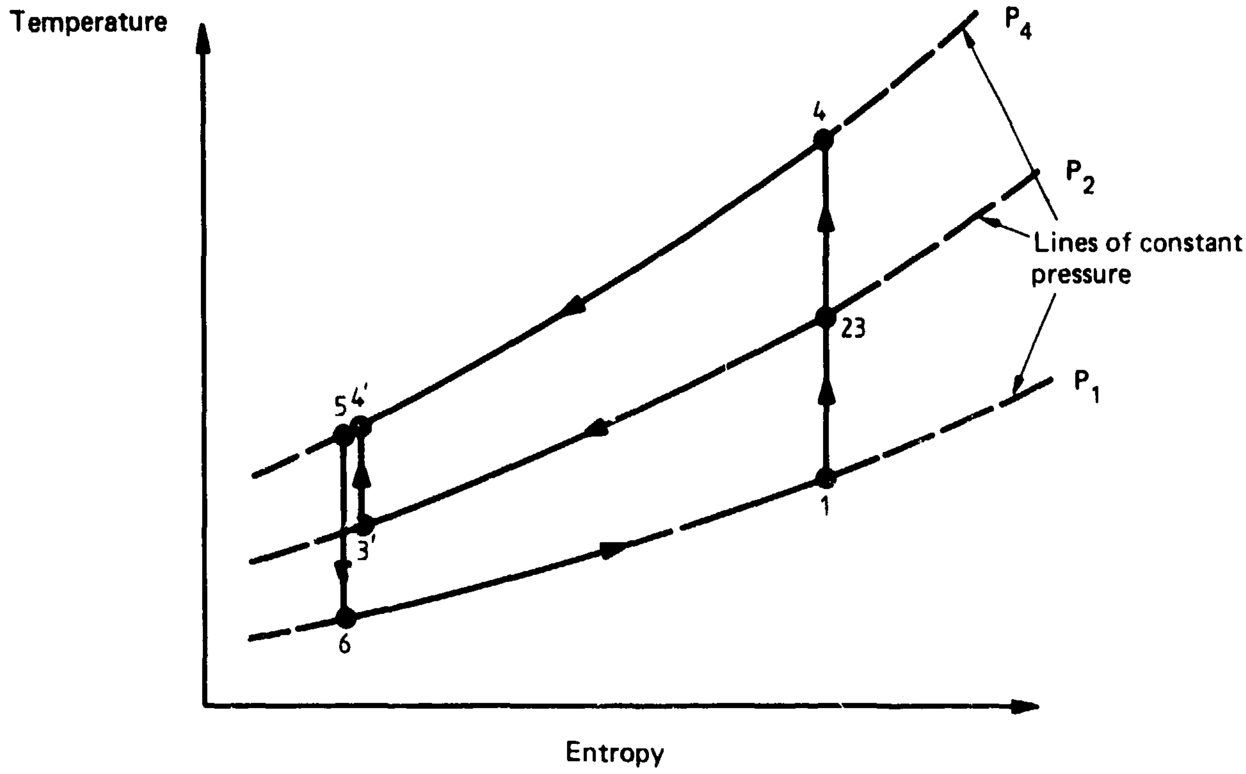


FIG. 19 TWO SPECIAL CASES OF THE BOOTSTRAP COOLING SYSTEM CYCLE

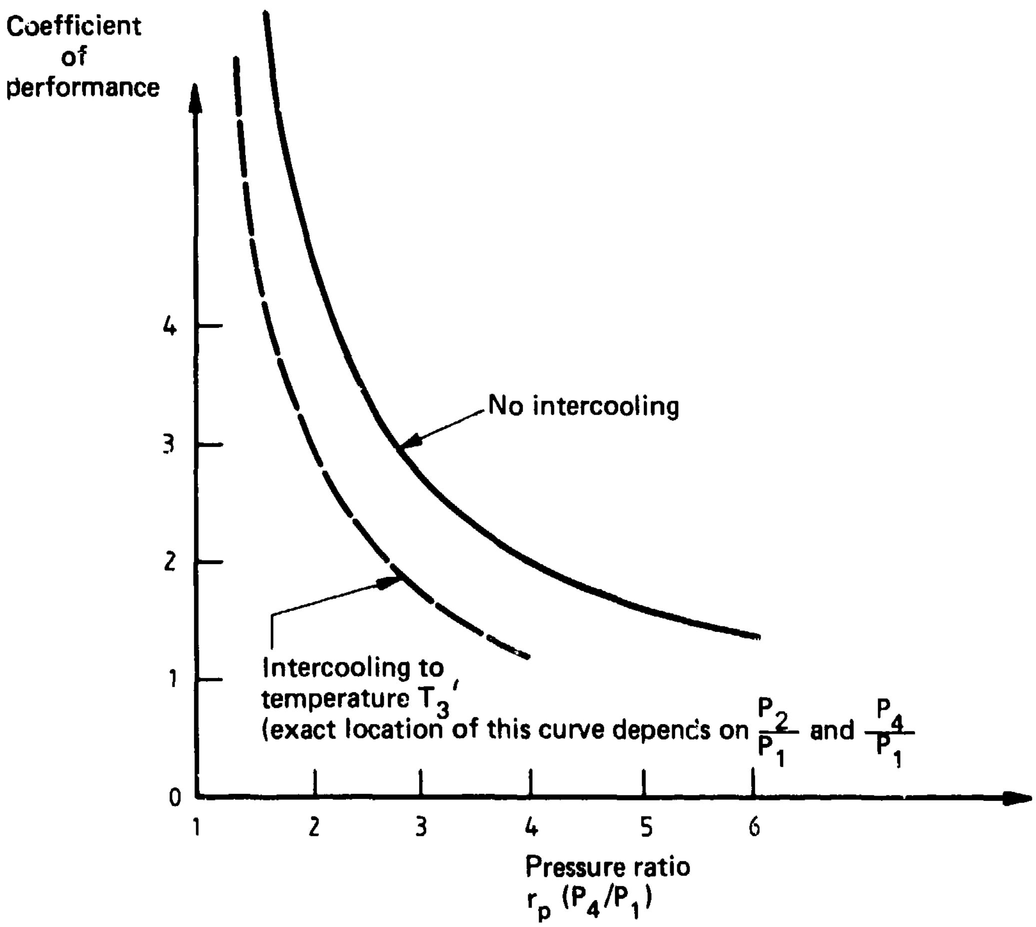


FIG. 20 COEFFICIENT OF PERFORMANCE VERSUS PRESSURE RATIO FOR A BOOTSTRAP SYSTEM

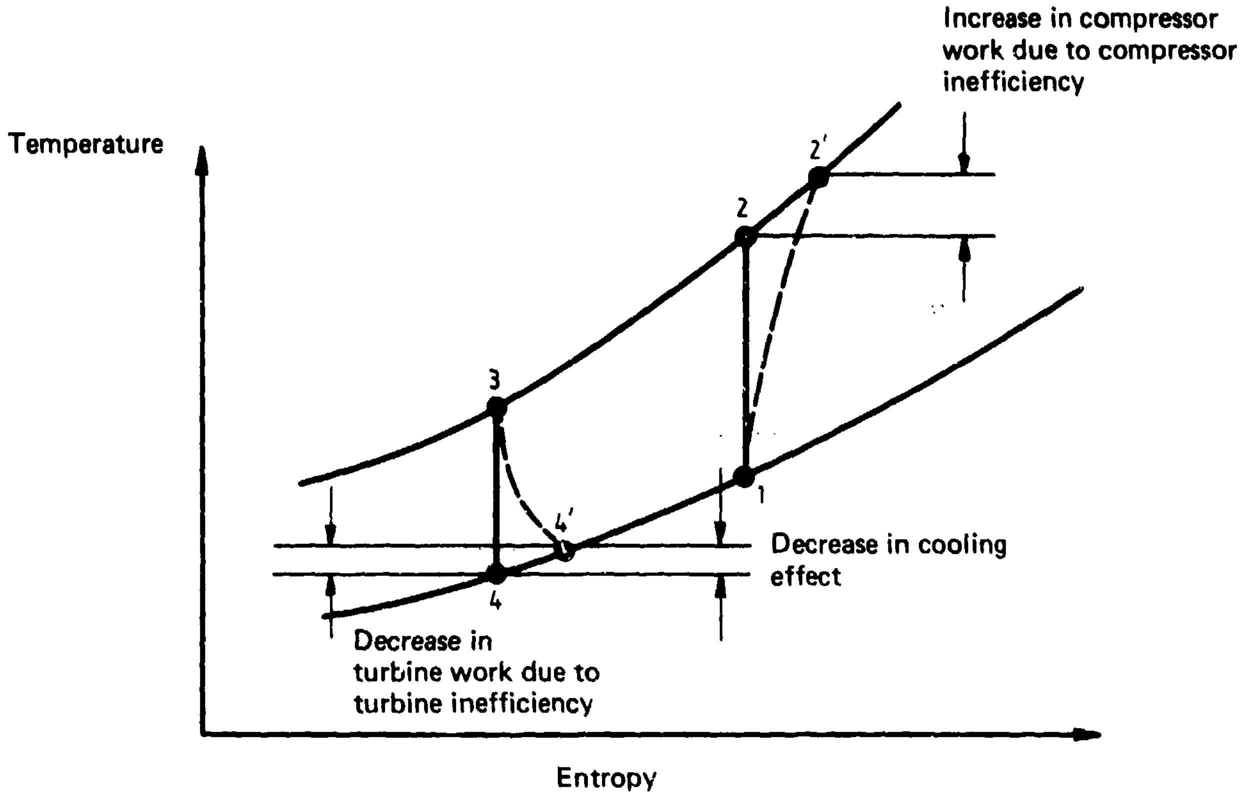


FIG. 21 EFFECT OF TURBINE AND COMPRESSOR INEFFICIENCIES

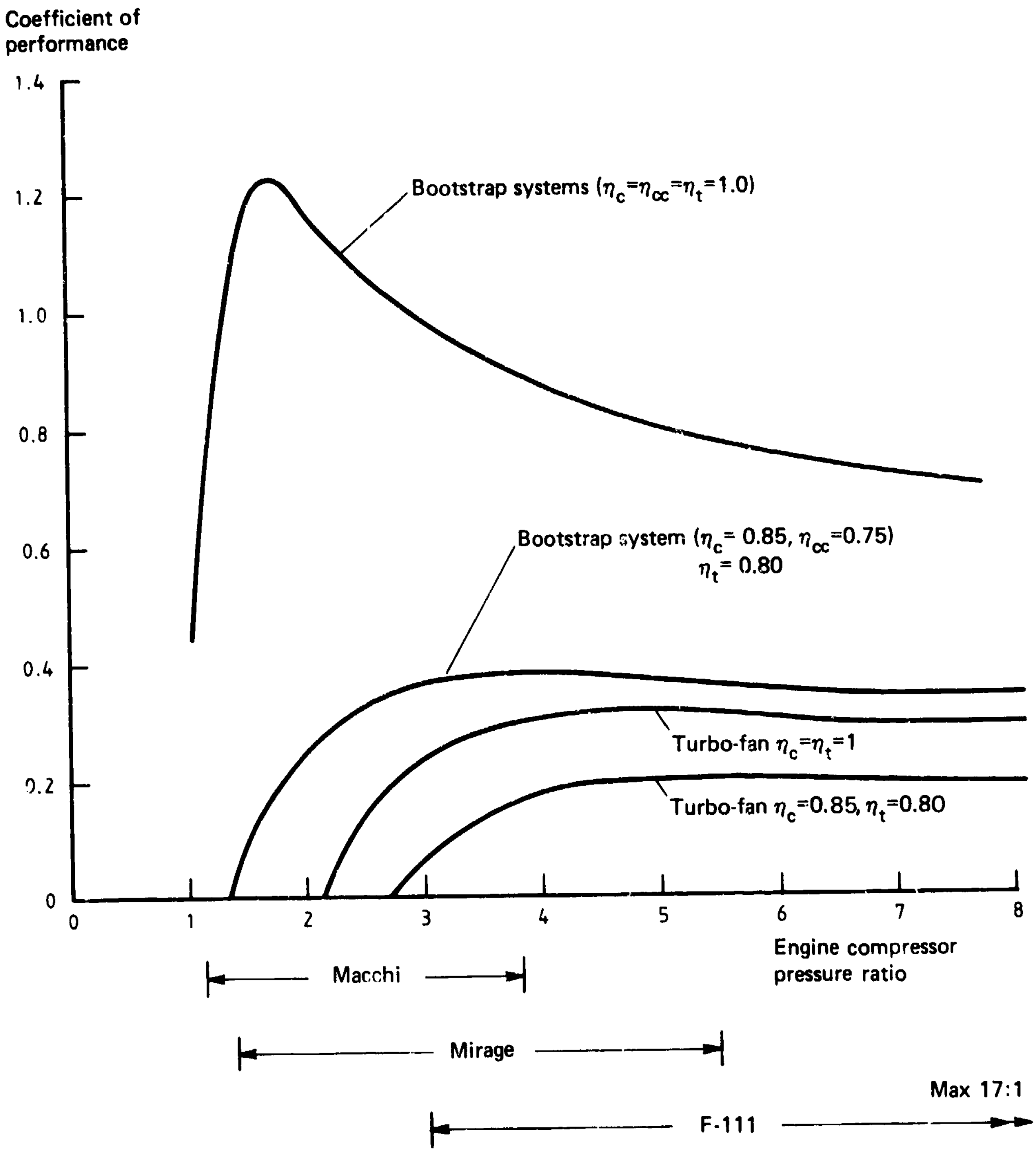


FIG. 22 COEFFICIENT OF PERFORMANCE FOR BOOTSTRAP AND TURBOFAN SYSTEMS WITH DRY AIR

supersonic flight, use is sometimes made of a water boiler, as in the F-111C and Mirage IIIO (see also Still 1957) for short high speed dashes. The fuel can also be used as a supplementary heat sink (Lowery and Howells 1974; Googan 1974; Lewis 1974; Le Clair 1974).

The condensate collected from the water separator can be sprayed into the ram-air stream used to cool the heat exchangers; some of the cooling effect lost when this moisture is condensed (see Section 5.2.5) can be then regained. However, the gain in system performance by this method is not great; a recent investigation by the author showed only 7% improvement in overall cooling for a particular system. A greater cooling effect is recovered from the condensate if it is passed through a separate heat exchanger (just before entry to the cooling turbine), where the airflow on the condensate side is small.

5.2.4 Turbine Outlet Temperature

The turbine outlet temperature (T_6 in Fig. 19) is limited in most air cycle systems to be greater than 0°C, to avoid freezing of entrained moisture on the turbine rotor and turbine outlet casing, and in the water separator. In a recent development by Normalair-Garrett Ltd., the turbine outlet temperature can be brought to -40°C (Giles 1977). This reduction in T_6 will markedly improve the COP of a turbofan system, but will not change the theoretical COP for a bootstrap system. A reduction in T_6 will, however, increase the cooling effect per unit mass of working fluid (in this case air) and so reduce the overall size of the system. Further discussion on the Normalair-Garrett system is given by Giles (1980).

5.2.5 Effect of Water Vapour

So far, in this discussion of air cycle systems, the effect of water vapour on cooling system performance has not been considered. However, in practice the moisture content of the air can be as high as 0·030 kg moisture/kg dry air (see Section 2). This moisture content can cause a large reduction in the cooling effect, in some cases reducing by one half the cooling attained with dry air. This reduction occurs because of the large amount of heat given up by condensation of moisture in the air, as it is cooled. The effect of moisture on the expansion process in a cooling unit can be readily assessed using a modified psychrometric chart as shown in Figure 23 (modified from Scofield 1949). A diagrammatic explanation of the use of the modified psychrometric chart is given in Figure 23, where the temperature rise T_{BD} of air due to heat given up by condensation of moisture is given by

$$T_{BD} = \delta_r \,.\, q_L/C_p, \tag{19}$$

where δ_r = reduction in water vapour content of air (kg/kg),

C_p = specific heat of air (kJ/kg °C), and

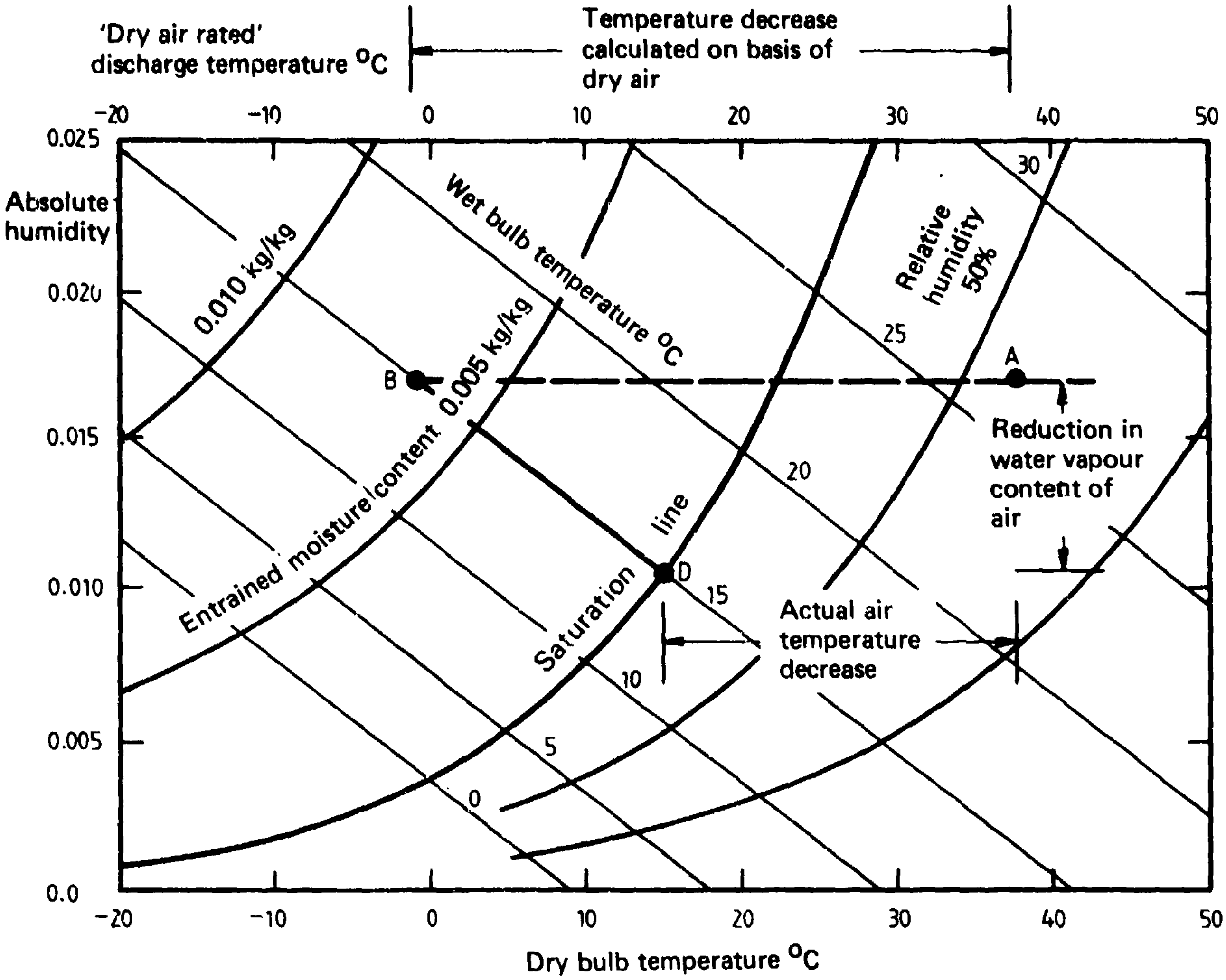
q_L = latent heat of condensation of moisture in air (kJ/kg).

5.3 Installed Performance

5.3.1 Effect on Engine Performance of Compressor Air Bleed

The air cycle system is frequently thought of as being inefficient and wasteful of engine power, presumably because the air is usually taken at a high temperature and pressure from the latter stages of a gas turbine compressor. The use of air from a compressor is, however, a fundamentally correct and necessary part of an air refrigeration cycle, as can be seen for example from Figures 17–19.

The *application* of compressor air bleed to some modern high pressure ratio engines is at times very wasteful, as can be seen from the temperature-entropy diagram of a turbo-fan system in Figure 24. In this figure the refrigeration cycle pressure ratio is p_2/p_1, and that of the engine p_3/p_1. In order that a substantially constant pressure is available at entry to the cooling unit, under engine conditions ranging from idle to maximum power, compressor air is frequently tapped from the last compressor stage; a restrictor is used to limit the pressure to a level less than 700 kPa (101 psia).



Note: Point A represents state of the ambient air. From this point the air is compressed, cooled in a heat exchanger, and delivered to the expansion turbine. Point B gives the entrained moisture content of the turbine outlet air, saturated at temperature T_D.

FIG. 23 PSYCHROMETRIC CHART FOR USE WITH AIR CYCLE COOLING SYSTEMS

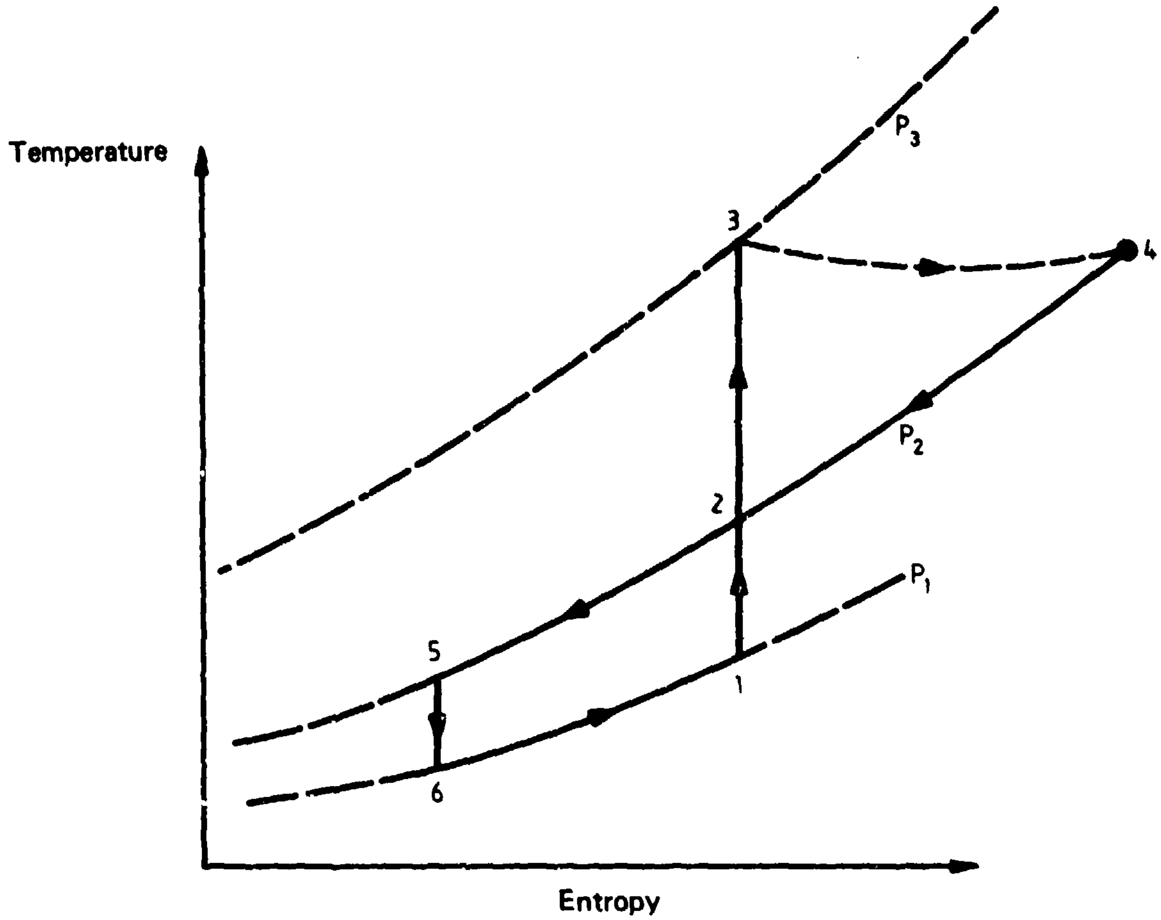


FIG. 24 THE EFFECT ON A TURBOFAN SYSTEM OF COMPRESSOR AIR BLEED FROM A HIGH PRESSURE RATIO ENGINE

At low engine power settings the cycle of Figure 24 follows the sequence 1-2-5-6. However, at high engine power settings engine compression follows the line 1-2-3, throttling then occurs through the restrictor, at constant temperature, to point 4, cooling through the heat exchangers occurs then from 4-2-5, and expansion in the turbine to point 6. Thus operation at high engine power is very inefficient; the sequence 2-3-4-2 is quite unnecessary, considerable engine power is used in the compression phase 2-3, and a very considerable drag may be generated by the heat exchangers in cooling from 4-2. Figure 24 is applicable to the F-111C (engine compressor pressure ratio 17 : 1), and later generation aircraft where the engine pressure ratio may be 25 : 1.

An obvious improvement is to bleed air from the earlier compressor stages at high engine power; this has been carried out to a limited extent with some recent engines.

The effect of compressor air bleed on engine power is greater than the power required to compress that fraction of the total flow which is bled off. This is because the compressor power requirement remains constant, but the airflow through that section of the turbine driving the compressor, and then used to drive either a "power" turbine or for propulsive thrust, is decreased by the amount of compressor bleed. The power available at entry to the first stage turbine will decrease by the same percentage as the compressor air bleed (unless the turbine inlet temperature is increased). As the compressor power requirement, which may be greater than a half of this available power, is constant, the power available for propulsive thrust (or another separate turbine stage) decreases by the order of twice the percentage air bleed. This is illustrated by data available for the RAN Mk 50 Sea King, where a 6% compressor air bleed results in a power decrease of 15%. The actual fuel consumption (kg/s) will decrease if air is bled off from the compressor, although the specific fuel consumption will probably increase. Hartley (1974) discusses the effect of compressor air bleed on the performance and fuel consumption of several Rolls-Royce aircraft engines.

5.3.2 Use of Engine Driven Compressor

The use of an engine driven compressor (EDC) is now quite rare, the only common use in Australia being in the P-3B and P-3C Orion. In the 1950s United States commercial aircraft were not permitted to use gas turbine engine bleed air for cabin conditioning; it is probable that for this reason the Orion uses an EDC. However, the use of an EDC confers a definite advantage in that when coupled to a constant-speed drive, the airflow is constant regardless of engine power setting, and thus very much more efficient than using engine compressor air bleed from a single tapping point. However, a constant-speed drive is readily available only from aircraft such as turbo-props, and helicopters, and then only when the propellers/rotor blades are rotating.

An alternative to using a constant-speed drive would be to use an auxiliary power unit (APU) to drive a compressor, or even direct air bleed from the compressor of the APU, as in the Orion P-3C (where it is used during ground operation of the aircraft).

5.3.3 Drag Penalty of Heat Exchangers

The aircraft drag penalty of a cooling system designed to cool 30 kW is typically 400 kW (Lowery and Howells 1974). A discussion on the ways by which this drag can be minimised, for example by improving thrust recovery and modulating the ram-air flow, is given by Le Claire (1974, 1980) and Simmonds (1974).

The drag penalty can be reduced by using the fuel as a heat sink, though this may be limited by the demand on fuel cooling made by other aircraft systems for example hydraulics oil, engine oil, and alternator constant-speed drive. Several references on the use of fuel cooling are Googan (1974), Singleton (1974), Lydiard (1974), Lewis (1974), Andrews (1974), Le Clair (1974, 1980).

5.3.4 Overall Effect on Aircraft Performance

This section on air cycle systems has detailed the reasons for a low overall COP; it can be seen that there is not one but a number of reasons which when combined give this result. For a modern high performance military aircraft weighing in the region of 30 000 kg, the overall aircraft penalty of a system designed to cool 30 kW heat load is estimated by Lowery and Howells (1974) to be 700 kW. This is made up of 300 kW power loss from the engines, and 400 kW to overcome drag of the heat exchangers and system mass, resulting in an overall coefficient of performance of 0·04. A discussion on the overall design of cooling systems to minimise the fuel penalty on the aircraft, is given by Le Clair (1980).

6. OTHER COOLING SYSTEM TYPES

6.1 Closed Air Cycle Systems

The basic cycle of the closed air cycle system is identical to that of the open air cycle system. A schematic of a closed bootstrap system is given in Figure 25(a), and the temperature–entropy diagram in Figure 25(b). The input shaft power required is the difference between compressor and turbine work.

A significant thermodynamic advantage when compared with the open air cycle is that the air in the system is at all times dry, and thus the system will not suffer a large decrease in cooling effect when operating in humid ambient conditions, as does the open air cycle (see Section 5.2.5).

The only present example of this system known to the author is a prototype manufactured by Normalair-Garrett Limited (Rootes, 1980).

6.2 Vapour Cycle Systems

6.2.1 Thermodynamic Analysis of Vapour Cycle Systems

The theoretical vapour cycle refrigeration cycle is sketched in Figure 26; the theoretical coefficient of performance is:

$$\mathrm{COP} = T_1/(T_2 - T_1). \qquad (20)$$

This COP is similar to that of the air cycle bootstrap system of Figure 19, where for no inter-cooling the COP from Equation (16) is $T_1/(T_4 - T_1)$. The air cycle system has the immediate disadvantage, however, that the heat rejection Q_{45} for the cycle does not take place at constant temperature, so that T_4 will be greater than the minimum hot sink temperature, thus lowering the COP.

A practical vapour refrigeration cycle is shown by the dashed line in Figure 26; the COP will then be lower than that found theoretically, which is, for typical hot and cold sink temperatures of 80°C and 10°C respectively, 4·04. However, the bench testing of a vapour cycle refrigeration system at the Aeronautical Research Laboratories has demonstrated practical COPs in the range 1·5–2·2.

A survey of possible refrigerants to be used when high heat sink temperatures are encountered in high-speed aircraft has been carried out by Van Winkle (1956) and Barger *et al.* (1956*a*). Another alternative is to use a hybrid system (incorporating an air cycle unit); this is analysed by Barger *et al.* (1956*b*). Dickmann *et al.* (1972) described the design and computer analysis of a number of vapour cycle system types for aircraft (including hybrid systems).

6.2.2 Practical Application of Vapour Cycle Systems to Aircraft

At the time of writing this report, vapour cycle systems are used in very few aircraft. The Australian application of vapour cycle systems is almost entirely limited to light aircraft. The Vickers VC-10 uses a vapour cycle system for cabin cooling (Sanders 1970). The use of vapour cycle systems overseas appears limited (apart from light aircraft) to systems manufactured by Stratos (a division of Fairchild, United States) and Sundstrand. These are installed in a number of aircraft, including helicopters. An experimental vapour cycle system was constructed by ARL and flight-tested in an RAN Sea King (Rebbechi 1979, 1980). This system performed well, with an overall COP of approximately 1·0. The results of bench testing of automotive-type components is reported by Rebbechi *et al.* (1980); the report contains a detailed analysis of the performance of individual components.

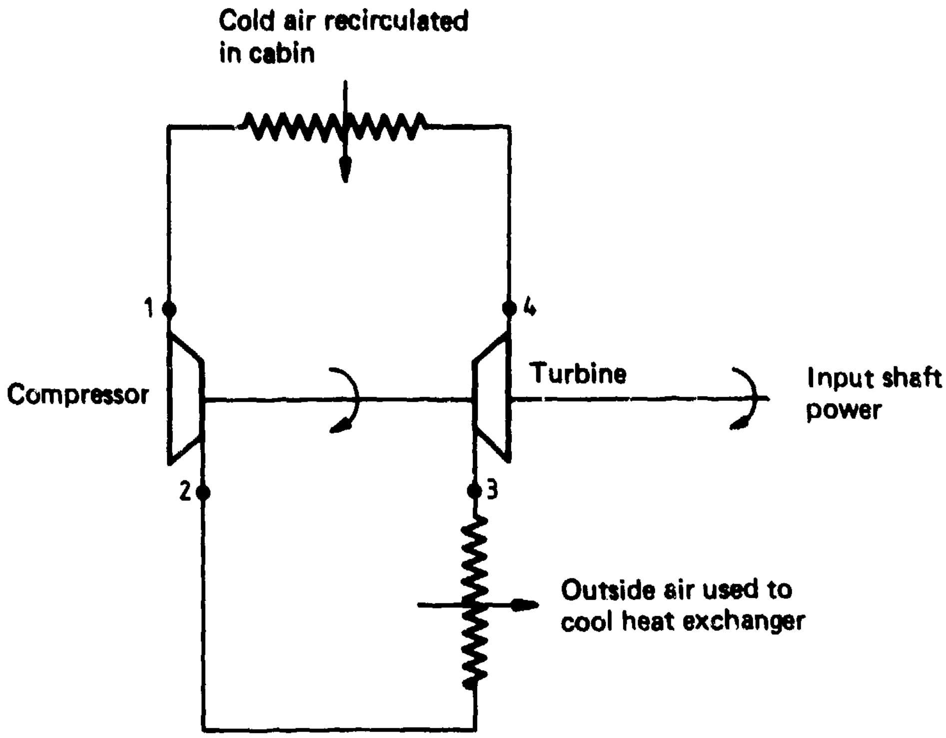


FIG. 25(a) SCHEMATIC OF CLOSED CYCLE BOOTSTRAP SYSTEM

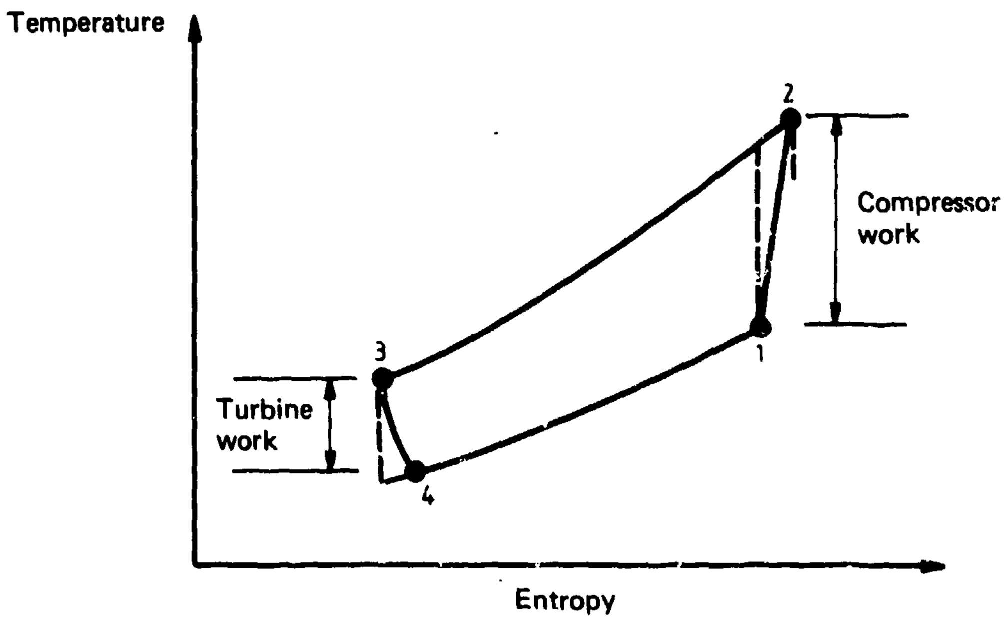
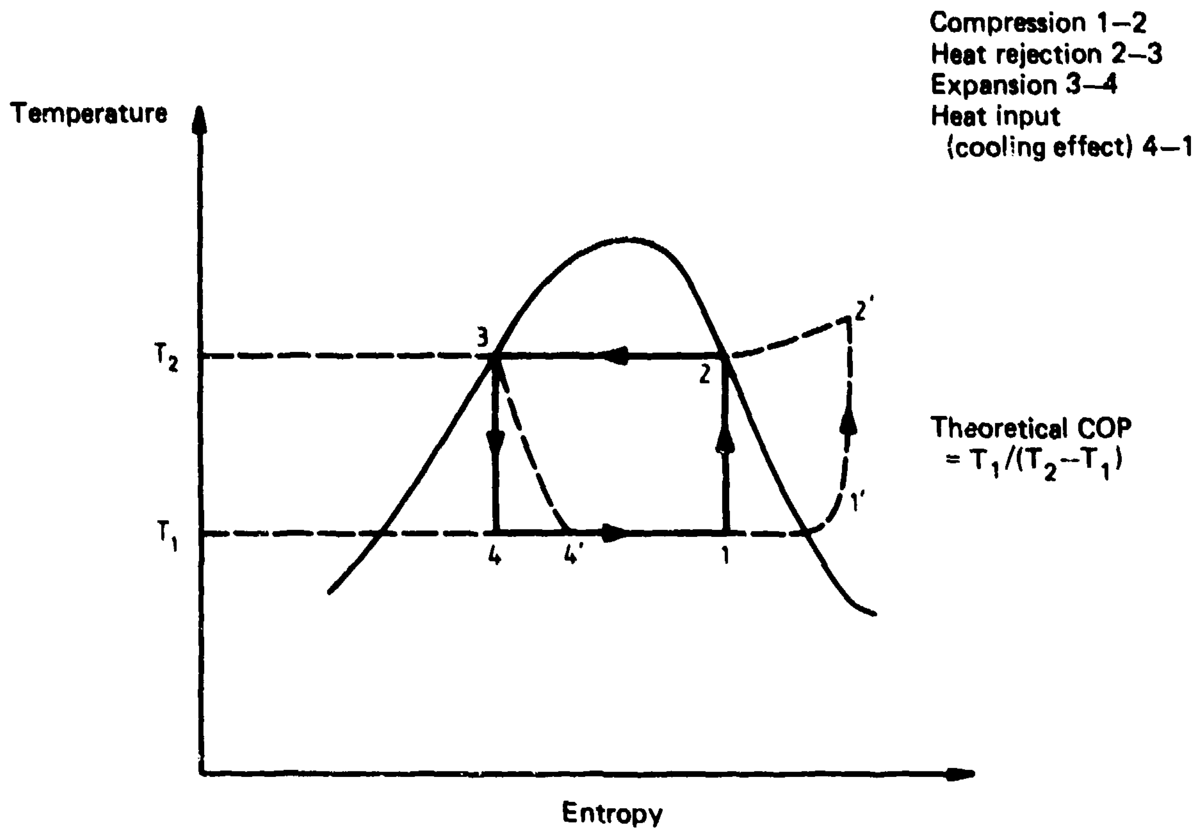


FIG. 25(b) TEMPERATURE-ENTROPY DIAGRAM FOR CLOSED CYCLE BOOTSTRAP SYSTEM



Note A practical refrigeration cycle is 4'–1'–2'–3, where 3–4' is throttling, 4'–1' evaporation, 1'–2' compression of superheated vapour, 2 –2 cooling of superheated vapour, and 2–3 condensation of vapour.

FIG. 26 TEMPERATURE – ENTROPY DIAGRAM FOR VAPOUR CYCLE SYSTEM

The three main types of aircraft vapour cycle systems are those using:

(*a*) centrifugal refrigerant compressor—as in VC-10;

(*b*) rotary compressor—as manufactured by Stratos (Lysholm-type) and Sundstrand;

(*c*) reciprocating piston refrigerant compressor—this type is applied only to light aircraft and utilises components manufactured for automotive air conditioning systems.

The centrifugal compressor is suited to systems of greater than 40 kW cooling capacity. The rotary compressor is best suited for systems of smaller capacity; it is very small and light and for electrically powered systems is built with an integral hermetically sealed 400 Hz electric motor. The reciprocating compressor, as built for automotive use, is not suitable for cooling loads greater than 7 kW.

The refrigerants used in aircraft vapour cycle systems are either R11, R114 or R12; refrigerant R12 being used in the rotary and reciprocating compressors.

Further details on the application of vapour cycle systems to aircraft can be found in the Society of Automotive Engineers Aerospace Recommended Practices 731A.

7. SPECIFICATION OF COCKPIT ENVIRONMENTAL CONTROL SYSTEMS FOR AIRCRAFT

The specification of cockpit environmental control systems (ECSs) includes the specification of both the design and installed performance of the system. The intent of this section is to describe, firstly, the overseas specifications that are written for environmental control systems, secondly, the existing RAAF specifications, and lastly to suggest a minimum content for future RAAF specifications.

7.1 United States Military Aircraft

One single document is not available to specify the ECSs in United States military aircraft; recourse is made to quite a number of documents. The principal reference is the Air Force Systems Command Design Handbook (AFSC DH). This handbook, which is in several volumes and supersedes the earlier Air Force Systems Command Manual 80-1 (AFSCM 80-1), has a similar function to the British Av. P. 970, in that it describes the design requirements for USAF aircraft. The relevant sections in the AFSC DH are listed in Appendix 5.

The other references used are Military Standards and Specifications; these are also listed in Appendix 5.

In addition to the military standards there are two Society of Automotive Engineers Aerospace Recommended Practices—SAE ARP 699, Design and Installation Practice for Engine Bleed Air Systems, and SAE ARP 731A, General Requirements for Application of Vapour Cycle Refrigeration Systems for Aircraft. Reference is also made in the AFSC DH to a four-part report on the Development of Integrated Environmental Control System Designs for Aircraft, published by the McDonnell Douglas Aircraft Company under a USAF contract. This is a computer program for conducting a trade-off study, and predicting the performance of environmental control systems. Part 1 (Diekmann *et al.* 1972), describes ECS design; Part 2 (Whitney *et al.* 1972*a*) describes the computer program; Part 3 (Whitney *et al.* 1972*b*) gives the computer program user's manual; and Part 4 (Glover *et al.* 1972) is a laboratory demonstration test, to verify the program. Both air cycle and vapour cycle systems are considered.

There is a number of military specifications relating to ground cooling support systems; these are listed in Appendix 5.

The aspect of the specifications which is discussed here relates only to the cockpit environment as it affects crew heat stress; other aspects are obviously important, but are less contentious than specification of the crew thermal environment.

A summary of the various design specifications for cabin thermal environments is given in Figure 27. It can be seen that there is some variation depending on the source of the recommendation. The design requirements range from a dry bulb temperature of 21·1°C for all conditions except transients (AFSC DH 2-3 DN4A1), to the zone 15·5–26·6°C dry bulb

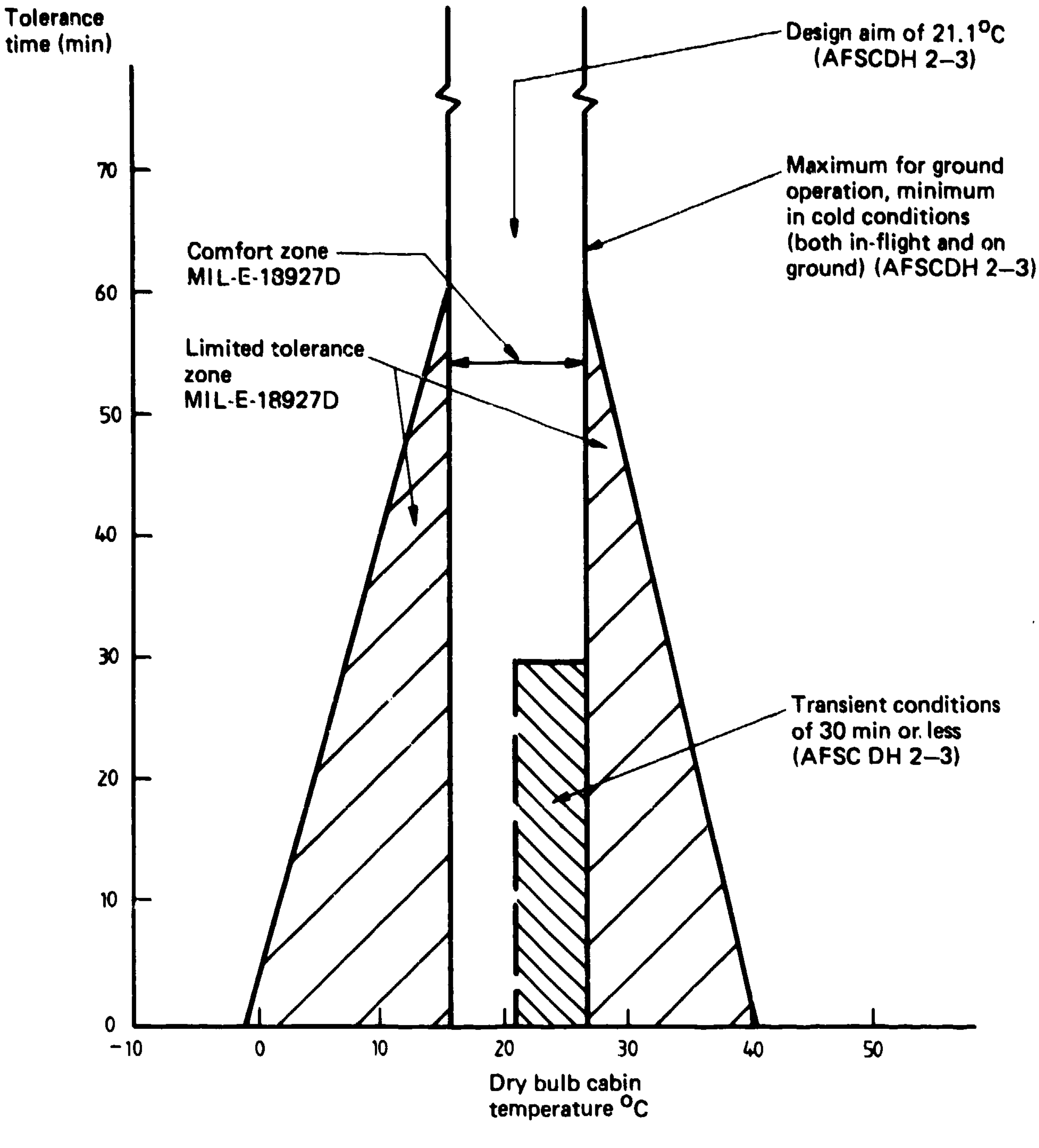


FIG. 27 SUMMARY OF VARIOUS CABIN THERMAL ENVIRONMENT DESIGN REQUIREMENTS

temperature during steady state and mild transients (MIL-E-18927D). In Figure 28 a graph is given of cabin dry bulb temperature versus cabin WBGT. This graph enables comparison between United States specifications (which are in terms of dry bulb temperature) and those of the RAAF which are in terms of cabin WBGT. It can be seen that the resulting cabin WBGT for all of the United States specifications is very low—in the region of 20°C WBGT.

The *external* climatic environment specified depends upon the particular publication; the AFSC DH specifies MIL-STD-210B (see Section 2). MIL-STD-890 refers to the ambient conditions specified in MIL-E-38453A; and MIL-E-18927D (USN ECS specification) refers to the Air Force-Navy Aeronautical Bulletin 421, a standard for aeronautical design atmospheric properties.

The cabin air velocities are not to be greater than 1 m/s (200 ft/min) at head level (MIL-E-18927D) or a maximum in the crew area of 1·5 m/s (300 ft/min) (AFSC DH), in either case not to be directed into eyes of the crew. Temperature variation is not to be greater than 5·5°C between foot and head level (MIL-E-18927D), or, as prescribed in the AFSC DH, not to deviate more than 2·8°C from the average compartment temperature.

7.2 British Military Aircraft

The principal British specification is Av. P. 970 Volume 1—Design Requirements for Aircraft for the Royal Air Force and Royal Navy. The relevant sections for cockpit environment are Chapter 105 (January 1968)—Operation in Various Climatic Conditions, and Chapter 109—View and Clear Vision (including maintenance of clear vision, rain removal, demisting, etc.).

Reference is made to a number of reports; those relating to cabin heat balance and cabin air conditioning are referred to in Section 4. There is a number of specifications relating to testing of systems and installed performance; these are listed in Table 1.

TABLE 1

Mintech Specifications

Specification	Subject
DTD1085	Climatic Testing of Airborne Instruments and Equipment
DTD1086	Climatic Testing of Ground Instruments and Equipment
DTD/RDI3716	Specification and Schedule of Tests on Cold Air Units
DTD/RDI3763	Specification and Schedule of Tests for Aircraft Filters for Cabin Air Conditioning Systems
DTD/RDI3810	Specification and Schedule of Tests for Airborne Heat Exchangers

The reference for crew thermal comfort is that of Billingham and Kerslake (1960*b*)—Specification for Thermal Comfort in Aircraft Cabins.

Environmental temperature-pressure requirements for *equipment* in aircraft are given by the British Standard 3G.100—General Requirements for Equipment in Aircraft, Part 2: Section 3: Subsection 3.2 (October 1970).

The cockpit environment requirements appear to be contradictory, in that Chapter 105 (1968) of Av. P. 970 prescribes that 'The heating and cooling systems shall be designed to ensure adequate thermal comfort for the crew, and efficient functioning of all equipment in the various climatic conditions likely to be encountered.' However, Leaflet 105/1 (1950) of Av. P. 970 has a provision that the cockpit temperature is not required to be more than 20°C below outside air temperature. The heating requirement prescribed is a mean of 25°C, in worst design conditions.

Adoption of the criterion of thermal comfort—the term 'adequate', being not defined, is meaningless—leads directly to Equation (4) of Section 3. Thus this is not a direct definition of cockpit environment, but takes into consideration clothing insulation, air velocity (as it affects heat transfer), and other factors as listed in Section 3. A sample calculation (as in Appendix 3)

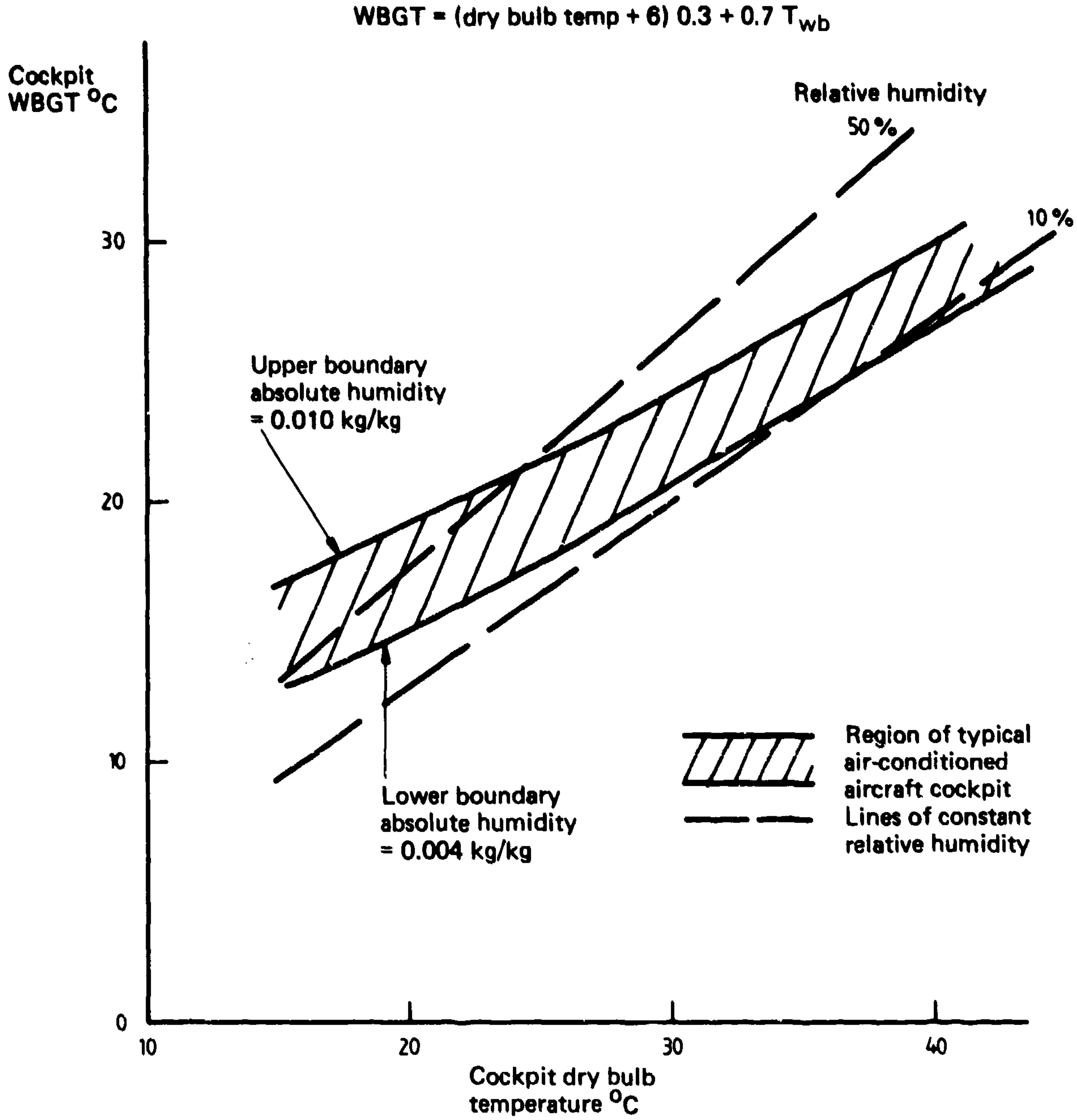


FIG. 28 GRAPH OF CABIN WBGT VERSUS CABIN DRY BULB TEMPERATURE, FOR A TYPICAL AIRCRAFT COCKPIT EXPOSED TO FULL SOLAR RADIATION

has been made for a well-insulated aircraft cockpit at high Mach number (1·0), and full solar radiation. The resulting dry bulb temperature is 18·2°C for thermal comfort. From Figure 27, it can be seen that this lies within the prescribed comfort zone of MIL-E-18927D. At higher aircraft Mach numbers, or for greater metabolic heat production by crew (working heavy controls, combat missions), the required dry bulb temperature will be below the comfort zone of Figure 27.

7.3 Existing RAAF Cockpit Environment Specifications

7.3.1 Thermal Environment

There is not at the present time a general specification for the cockpit environment in RAAF aircraft. Air Force Staff Requirements (AFSRs) prescribe conditions to be met by new aircraft; they specify the environment according to what is regarded as appropriate at that time. Recent AFSRs embody the following features:

(*a*) a cockpit environment upper limit of 27–28°C WBGT, with a lower limit of 20°C dry bulb temperature;

(*b*) consideration of heat-soak effects and pull-down times; and

(*c*) both of the requirements (*a*) and (*b*) to be met during operations in the RAAF Atmospheric Environment (see Figs. 1 and 2 and Appendix 1).

7.3.2 Canopy Misting and Fog

Minimum acceptable visibility is defined by reference to ASCC Air Standard 10/53A. Freedom from misting, or from free moisture entering the cockpit, is specified for a rapid descent to sea level, with engine at idle, after operation at high altitude. This test is to be carried out in the high temperature and humidity extremes of the RAAF Atmospheric Environment.

7.4 Summary of Requirements for Effective Specification of Cabin Conditioning Systems

7.4.1 Cabin Thermal Environment

The present RAAF specifications prescribe an upper limit to the cabin environment which is much more severe with regard to crew thermal load than either United States or British specifications. For example the upper limit prescribed by the United States MIL-E-18927D, of 26·6°C dry bulb temperature, is equivalent to approximately 21°C WBGT in a typical cockpit, compared with the RAAF limit of 27–28°C WBGT. The climatic environment (with regard to dry bulb temperatures) for the United States and British specifications is more severe than the RAAF atmospheric environment.

The minimum requirements which are necessary for specification of the cabin thermal environment are:

(*a*) Specification of outside climatic extremes.

(*b*) Definition of the aircraft operating regimes, that is:

(i) all flight conditions, including low-level flight with reduced power (circuits, for example), with some consideration of the duration of high-speed flight and whether a widening of the cockpit thermal specification is permissible for a short duration (as in Fig. 27);

(ii) ground operation—parked (where ground-based supplies may be necessary), engine idling, and taxiing.

(*c*) Cockpit temperature and humidity—use of a WBGT index or similar where the upper limit to the cockpit environment is high (28°C WBGT) and evaporation is a significant heat loss mechanism. Where the evaporation mechanism is not as significant and the WBGT index inappropriate (that is for the comfort zone and lower limit), the use of a black globe temperature only, is appropriate.

(*d*) Temperature control—to be automatic with manual override, with a small tolerance on its accuracy (typically ± 2°C). Temperature sensed at crew stations.

7.4.2 Cabin Air Distribution

It is generally accepted that cabin air velocities are not to exceed 1 m/s (200 ft/min) at head level, and 3 m/s (600 ft/min) at other places. United States standards prescribe limits to the temperature difference between foot and head level of 5·5°C; while this is desirable, a small differential such as this is unlikely to be achieved in a relatively small fighter aircraft cockpit, with a high mass flow of cooling air. However, it is a desirable aim to work towards and should be achievable in larger cockpits, particularly if there is no direct solar radiation on the crew.

7.4.3 Defogging and Demisting

Several standards prescribe limits to the amount of canopy area to be free from misting; the Australian standard is ASCC Air Standard 10/53A, The United States standard MIL-T-5842, and the British standard Av. P. 970 Chapter 109—View and Clear Vision. The flight test procedures to satisfy this requirement are important; current RAAF requirements require extended operation above 9 100 m (30 000 ft), followed by a rapid descent, with the engine at idle thrust setting, into conditions of high temperature and humidity. This requirement is similar to that of the United States MIL-T-18606 (Aer) (Test Procedures for Aircraft Cabin Pressurising and Air Conditioning Systems), Par. 4.3.

The cabin ventilation air should also be free from entrained moisture.

7.4.4 Acoustical Noise Level

Acoustical noise generated by operation of the air conditioning system has not been identified as a problem in RAAF aircraft; however, the relevant United States specification is MIL-A-8806 (General Specification for Acoustical Noise Level in Aircraft), together with two specifications—MIL-I-7171 and MIL-S-6144—relating to acoustical insulation.

7.4.5 Equipment Cooling

Compartments for air cooled equipment must necessarily be maintained within the specified maximum and minimum temperature limits in accordance with the equipment specifications. The relevant United States document is MIL-E-5400P (General Specification for Airborne Electronic Equipment) and the relevant British document is British Standard 3G.100—General Requirements for Equipment in Aircraft, Part 2: Section 3: Subsection 3.2 (October 1970).

8. CONCLUSIONS

(*a*) The Australian climate is not as severe with regard to high temperatures and humidity as that which may be encountered elsewhere. The RAAF Atmospheric Environment accords closely with recently collated data for 1% extreme temperatures in Australia and nearby islands. However, the RAAFAE is inadequate for specification of world-wide aircraft operations; in these circumstances recourse is necessary to the United States MIL-STD-210B.

The RAAFAE specifies a relationship between high temperature and humidity which is much more severe than the Australian environment. For operation of aircraft within Australia, consideration should be given to amending this part of the RAAFAE.

(*b*) The RAAF specification of a cabin environment of 27–28°C WBGT will result in a higher level of crew heat stress than either United States or British specifications. However, current knowledge in aviation ergonomics indicates that this WBGT level will probably allow safe and effective flying operations in most cases (providing crew are wearing only their current issue flying clothing), although the crew will be uncomfortable, may be perspiring freely, and will suffer dehydration unless maintaining a regular intake of fluids.

(*c*) It is shown that there is no technological difficulty in arriving at a cabin heat balance; it is desirable that cabin air distribution be studied with full-size models. Cabin cooling tests in an environmental test chamber with static air conditions are quite unrepresentative of flight conditions.

(*d*) The open air cycle system has been shown to have a high power consumption in comparison with closed vapour cycle and closed air cycle systems. At the present time, the light weight and apparent simplicity of the open system appears to be given greater consideration than the high power loss. Possible ways of bringing about large reductions in the power consumption are discussed; however, at the present time these improvements are not widely used.

(*e*) Present RAAF specifications for cabin conditioning systems would, if properly met by an aircraft manufacturer, ensure an adequate cabin environment. Detailed comments have been made here on possible additions to the present specification; these comments are intended to amplify rather than change the specification.

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APPENDIX 1

RAAF Atmospheric Environment for Humidity

The RAAFAE for humidity is given only as a function of dew point versus altitude (see Fig. A1). The graphical relationship can be expressed in tabular form, as in Table A1.

TABLE A1

Altitude (ft)	ICAN std. pressure (kPa)	Dew point temp. (Fig. A1) (°C)	Saturation pressure (Pa)	Absolute humidity (kg/kg)
0	101·3	30		0·027
2000	94·2	27	3·50	0·024
5000	84·2	22	2·55	0·019
10 000	69·6	15	1·76	0·016

The absolute humidity can be found from the expresssion (contained in standard thermodynamic texts),

$$\text{absolute humidity} = 0{\cdot}622 p_s/(p - p_s), \qquad \text{(A1)}$$

where p_s = saturation pressure of water vapour at the mixture temperature (kPa), and

p = total (barometric) pressure of the mixture (kPa).

Using ICAN standard pressures and saturated water vapour tables for the various altitudes, the absolute humidity can be calculated from the pressures listed in Table A1. The resulting absolute humidity as a function of altitude is plotted in Figure 3.

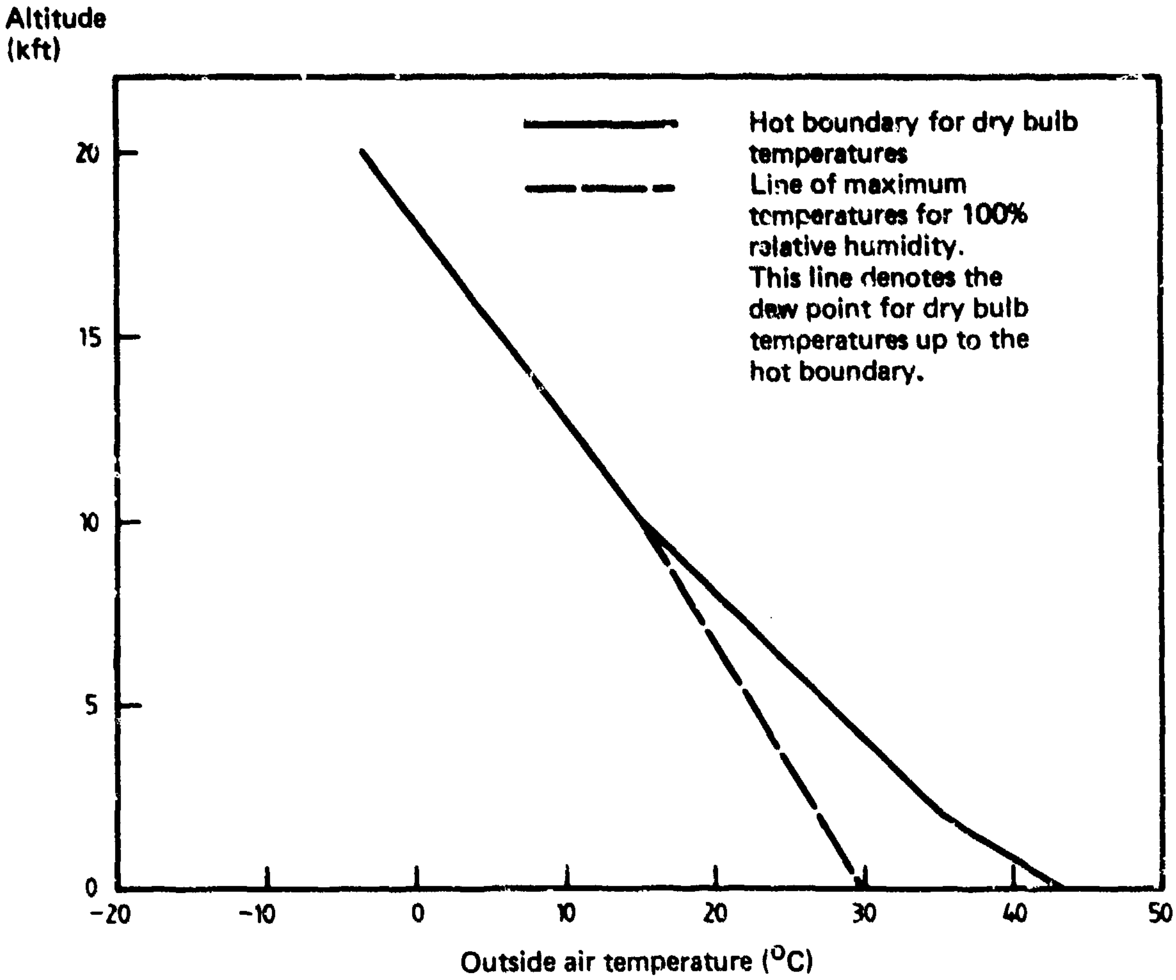

FIG. A1 RAAF ATMOSPHERIC ENVIRONMENT

APPENDIX 2

Heat Balance of a Black Globe

The heat balance of a black globe is given by

$$Q_{rg} + Q_{sg} = Q_{cg} \tag{A2}$$

where Q_{rg} = heat transfer due to radiation exchange with surroundings (W),

Q_{sg} = heating of globe by direct solar radiation (W), and

Q_{cg} = convective heat transfer between surface of globe and surrounding air (W).

Now, as the projected area of the globe is $\pi D^2/4$, and the total surface area πD^2, where D is the diameter, Equation (A2) can be written as

$$h_r(T_w - T_g) + Q_{si}/4 = h_{cg}(T_g - T_a), \tag{A3}$$

where h_r = linearised radiation exchange coefficient (W/m² °C),

T_w = radiant temperature of surroundings (°C),

T_g = temperature of the black globe (°C),

Q_{si} = incident solar radiation (W/m²),

D = globe diameter (m),

h_{cg} = convective heat transfer coefficient (W/m² °C), and

T_a = temperature of air surrounding globe (°C).

From Kerslake (1972), the linearised radiation exchange coefficient h_r can be taken as 6·6 W/m² °C for the temperature range 30–37°C; the convective heat transfer coefficient h_{cg} is given, for air at atmospheric pressure and in the region of 30°C, as

$$h_{cg} = 6{\cdot}6\ V^{0{\cdot}6}\ D^{-0{\cdot}4}, \tag{A4}$$

where V = air velocity (m/s).

Because of the current use of both 150 mm and 50 mm globes, it is of interest to compare the temperatures reached by the two globes in equal environments. The convective heat transfer coefficient of the smaller globe is greater; this results in a lower temperature of the smaller globe in equal environments.

From Equation (A4), for a 150 mm globe,

$$h_{cg(150)} = 14{\cdot}1\ V^{0{\cdot}6}, \tag{A5}$$

and for a 50 mm globe,

$$h_{cg(50)} = 21{\cdot}8\ V^{0{\cdot}6}. \tag{A6}$$

The temperature differential between a globe and the surrounding air can be found by re-casting Equation (A3) in the form

$$T_g - T_a = [h_r(T_w - T_a) + Q_{si}/4]/[h_{cg} + h_r]. \tag{A7}$$

Denoting the temperature of the 150 mm globe as $T_{g(150)}$, and the temperature of the 50 mm globe as $T_{g(50)}$, from Equation (A7), for equal environments,

$$(T_{g(150)} - T_a)/(T_{g(50)} - T_a) = (h_{cg(50)} + h_r)/(h_{cg(150)} + h_r), \tag{A8}$$

where $h_{cg(50)}$ = convective heat transfer coefficient of 50 mm globe (W/m² °C), and

$h_{cg(150)}$ = convective heat transfer coefficient of 150 mm globe (W/m² °C).

From Equation (A5) and (A6), for a velocity of 3 m/s, and noting that $h_r = 6 \cdot 6$ W/m^2 °C, then

$$(T_{g(150)} - T_a)/(T_{g(50)} - T_a) = 1 \cdot 44. \tag{A9}$$

This ratio is fairly constant over a limited range of air velocity (for $V = 2$ m/s the ratio is $1 \cdot 42$). Equation (A9) is plotted in Figure 8.

It is of particular interest to note that the ratio of globe temperature differentials in Equation (A8) is independent of the source of radiant heat loading, whether from surroundings or direct solar radiation, and is also largely independent of temperature, other than the small influence of temperature on h_{cg} and h_r.

The relationship between globe temperature, surrounding wall temperature and incident solar radiation is given by Equation (A7). For a 150 mm globe and $V = 3$ m/s, Equation (A7) can be brought to the form

$$T_g - T_a = 0 \cdot 195(T_w - T_a) + 0 \cdot 0074 Q_{si}. \tag{A10}$$

This expression is presented graphically in Figure 9. Equation (A10) is applicable also to cabin wall temperatures *less* than cabin air temperatures; these could be illustrated in Figure 9 by extending the lines of Q_{si} = constant to the left of the vertical ordinate.

APPENDIX 3

Aircraft Cabin Heat Balance

An example is given here of the cabin heat balance as applied to a Macchi MB326H jet trainer, at the following conditions:

(1) outside air temperature 43°C;

(2) cloudless sky;

(3) Mach number 0·6;

(4) low altitude; and

(5) two crew members.

The heat balance is analysed for two cabin conditions—a cabin environment of 28°C WBGT, and a cabin environment where thermal comfort of the crew is attained (utilising Equation (4) of Section 3.3).

Basic data for the Macchi is as follows:
projected area of transparency, 1·89 m^2; and
transmittance of transparency, 0·83.

Internal surface areas:
canopy, 3·32 m^2;
fuselage sides, 4·51 m^2;
floor, 2·39 m^2;
rear bulkhead, 1·2 m^2; and
front bulkhead, 0·54 m^2.

A3.1. Cabin Heat Balance for cabin WBGT of 28°C

Assuming initially that the cabin wall temperature is equal to cabin air temperature adjacent to the crew and that the cabin absolute humidity is 0·008 kg/kg, then for a cabin air temperature of 38°C, $T_{wb} = 20{\cdot}8°C$, $T_g = T_a + 6 = 44°C$ (from Fig. 9), then from Equation (2), WBGT = 28°C.

It has been shown by Hughes (1968) that an optimum cockpit air distribution scheme gave a mean temperatue (T_e) of the air in the immediate vicinity of the pilot such that

$$T_e - T_{in} = 0{\cdot}75(T_{out} - T_{in}), \tag{A11}$$

where T_{in} = cockpit inlet air temperature (°C), and

T_{out} = cockpit outlet air temperature (°C).

Then, assuming an inlet air temperature of 4°C (usually maintained just above 0°C to avoid icing problems), and noting that T_e = 38°C, then from Equation (A11),

$$T_{out} = 49°C. \tag{A12}$$

The mean cabin air temperature T_m is given by

$$T_m = 0{\cdot}5(T_{in} + T_{out}), \tag{A13}$$

and where T_{in} = 4°C, T_{out} = 49°C, from Equation (A13), T_m = 26·5°C.

The total heat input to the cabin, Q_t, is given by

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Aircraft Cabin Heat Balance

An example is given here of the cabin heat balance as applied to a Macchi MB326H jet trainer, at the following conditions:

(1) outside air temperature 43°C;

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(4) low altitude; and

(5) two crew members.

The heat balance is analysed for two cabin conditions—a cabin environment of 28°C WBGT, and a cabin environment where thermal comfort of the crew is attained (utilising Equation (4) of Section 3.3).

Basic data for the Macchi is as follows:
projected area of transparency, $1 \cdot 89\ m^2$; and
transmittance of transparency, 0·83.

Internal surface areas:
canopy, $3 \cdot 32\ m^2$;
fuselage sides, $4 \cdot 51\ m^2$;
floor, $2 \cdot 39\ m^2$;
rear bulkhead, $1 \cdot 2\ m^2$; and
front bulkhead, $0 \cdot 54\ m^2$.

A3.1. Cabin Heat Balance for cabin WBGT of 28°C

Assuming initially that the cabin wall temperature is equal to cabin air temperature adjacent to the crew and that the cabin absolute humidity is 0·008 kg/kg, then for a cabin air temperature of 38°C, $T_{wb} = 20 \cdot 8$°C, $T_g = T_a + 6 = 44$°C (from Fig. 9), then from Equation (2), WBGT = 28°C.

It has been shown by Hughes (1968) that an optimum cockpit air distribution scheme gave a mean temperatue (T_e) of the air in the immediate vicinity of the pilot such that

$$T_e - T_{in} = 0 \cdot 75(T_{out} - T_{in}), \tag{A11}$$

where T_{in} = cockpit inlet air temperature (°C), and

T_{out} = cockpit outlet air temperature (°C).

Then, assuming an inlet air temperature of 4°C (usually maintained just above 0°C to avoid icing problems), and noting that T_e = 38°C, then from Equation (A11),

$$T_{out} = 49°\text{C}. \tag{A12}$$

The mean cabin air temperature T_m is given by

$$T_m = 0 \cdot 5(T_{in} + T_{out}), \tag{A13}$$

and where T_{in} = 4°C, T_{out} = 49°C, from Equation (A13), $T_m = 26 \cdot 5$°C.

The total heat input to the cabin, Q_t, is given by

$$Q_t = Q_w + Q_m + Q_e + Q_s, \tag{A14}$$

where Q_w = heat flow through cockpit walls and transparencies (W),

Q_m = pilot's metabolic heat output (W),

Q_e = electrical heat load (W), and

Q_s = solar radiation entering the cockpit via the transparencies (W).

(*a*) ***Heat flow through cockpit walls.*** The cockpit outer skin temperature is given by Hughes (1968) as

$$(T_s + 273) = (T_a + 273)(1 + 0{\cdot}2\, r_e M^2), \tag{A15}$$

where T_a = ambient temperature (°C),

M = Mach number, and

r_e = recovery factor (usually taken as 0·89).

Then, for $M = 0{\cdot}6$, $T_a = 43$°C, from Equation (A15),

$$T_s = 63°\mathrm{C}. \tag{16}$$

The cockpit wall conductivity is assumed to be 5 W/m² °C (Hughes 1968) and that of the transparency 11·6 W/m² °C (Beenham 1969). Using then an internal convective heat transfer coefficient of 22 W/m² °C (Hughes 1968), the overall heat transfer coefficient of the cockpit wall, including the floor, is 4·07 W/m² °C, and that for the transparency 7·6 W/m² °C. Then, neglecting front and rear bulkheads, the overall heat transfer coefficient (h_w) for the cockpit walls, including the floor, is

$$h_w = 4{\cdot}07\,(4{\cdot}51 + 2{\cdot}39)$$

$$= 28{\cdot}08\ \mathrm{W/°C} \tag{A17}$$

and for the transparencies (h_t) is

$$h_t = 7{\cdot}6\,(3{\cdot}32)$$

$$= 25{\cdot}2\ \mathrm{W/°C}. \tag{A18}$$

The heat flow through the walls and transparencies is then

$$Q_w = (T_s - T_m)(h_w + h_t) \tag{A19}$$

$$= 36{\cdot}5\,(28{\cdot}1 + 25{\cdot}2)\ \mathrm{W},$$

$$= 1945\ \mathrm{W}. \tag{A20}$$

(*b*) *Pilot's metabolic heat load.* This is usually taken as 120 W per crew member for light activity.

(*c*) *Electrical heat load.* For the Macchi this is estimated at 500 W.

(*d*) *Solar radiation entering cockpit through transparencies.*

$$Q_s = A\,\tau\,Q_{si}, \tag{A21}$$

where A = projected area of transparencies (m²),

τ = transmittance, and

Q_{si} = incident solar radiation (W/m²).

Then, for $A = 1{\cdot}89$ m², $\tau = 0{\cdot}83$, $Q_{si} = 1000$ W/m²,

$$Q_s = 1570\ \mathrm{W}. \tag{A22}$$

The total heat input to the cabin, Q_t, from Equation (A14), is

$$Q_t = 4260\ \mathrm{W}. \tag{A23}$$

The cabin air mass flow requirement can now be calculated, as

$$Q_t = m_t C_p (T_{out} - T_{in}), \tag{A24}$$

where m_t = cabin air mass flow (g/s), and

C_p = specific heat of air (kJ/kg °C).

And, from equation (A12), T_{out} = 49°C, also T_{in} = 4°C, C_p = 1·01 kJ/kg °C. Hence

$$m_t = 94 \text{ g/s} \qquad (12 \cdot 4 \text{ lb/min}). \tag{A25}$$

Checking now on the actual cabin surface temperatures of inner wall and transparencies, from the values of heat flow through the walls (including floor) and transparencies from Equation (A19),

cockpit wall inner surface temperature = 33·3°C, and

transparency inner surface temperature = 39·1°C.

The mean radiant temperature is then approximately 36·2°C. As T_e is 38°C, from Figure 9 it can be seen that this result will depress the globe temperature by 0·4°C. The resulting change in WBGT will then be (from Equation (2)), 0·4/3 $\simeq$ 0·1°C.

A3.2 Cabin Heat Balance for Crew in a State of Thermal Comfort

For the crew to be in a state of thermal comfort their mean skin temperature is 33°C (Billingham and Kerslake 1960*a*,*b*). A comprehensive analysis of this heat balance is given by Hughes (1968), where he includes design curves to enable the rapid evaluation of cooling air requirements for a wide variety of aircraft and crew conditions. To evaluate crew comfort requires consideration of both the aircraft heat balance equation, (A14), and the crew heat balance equation, (3). To simplify this analysis it will be initially assumed that T_e = 18°C, the crew and cabin heat balances can then be evaluated, and if the crew skin temperature is other than 33°C, the analysis is repeated, with a revised assumption for T_e.

(*a*) *Cabin heat balance.* From Equation (A11), if T_e = 18°C, and T_{in} = 4°C, then

$$T_{out} = 22 \cdot 7°\text{C}, \tag{A26}$$

and from Equation (A13),

$$T_m = 13 \cdot 3°\text{C}.$$

Using the values of cockpit heat transfer coefficients evaluated in the previous section A3.1, from Equation (A19),

$$Q_w = (T_s - T_m)(53 \cdot 3), \tag{A27}$$

$$= (63 - 13 \cdot 3)(53) \text{ W},$$

$$= 2650 \text{ W}. \tag{A28}$$

Using now the previous values for Q_m, Q_e, Q_s, the cabin heat input Q_t is

$$Q_t = 4960 \text{ W},$$

and, from Equation (A24),

$$m_t = 263 \text{ g/s} \qquad (34 \cdot 7 \text{ lb/min}). \tag{A29}$$

(*b*) *Pilot heat balance.* The internal surface temperatures of the cockpit wall and transparency can be calculated from the heat flows and insulation values of these two regions. The resulting temperatures are

cockpit wall inner surface = 22·5°C, and

transparency inner surface = 30·5°C,

giving an approximate mean radiant temperature of 26·5°C.

The crew heat balance equation, (4), can be simplified by using a linearised radiation exchange coefficient as in Appendix 2 of 6·6 W/m² °C. Equation (4) can then be rewritten as

$$T_s = q_m/k_c + [T_e h_a + q_m + 0\cdot31\alpha_{cs}\tau Q_{si} + 6\cdot6T_w]/[6\cdot6 + h_a]. \qquad (A30)$$

Hence, from Equation (A30), and noting that $T_e = 18°C$, and using typical values referred to in Section 3.3 of $q_m = 87 W/m^2$, $h_a = 21\cdot5 W/m^2 °C$, $\alpha_{cs} = 0\cdot5$, $k_c = 11\cdot6 W/m^2 °C$, and also $Q_{si} = 1000 W/m^2$, $\tau = 0\cdot83$, then

$$T_{sk} = 35\cdot2°C. \qquad (A31)$$

As this value is greater than the comfort level of $T_{sk} = 33°C$, the calculations should be repeated with a lower value of T_e.

A3.3 Summary

The result of Section A3.1 implies that a relatively low airflow of 94 g/s (12·4 lb/min) is adequate to maintain a cockpit WBGT of 28°C. In practice, this is not so; the calculations are based on the assumption that an optimum air distribution scheme exists (equation A11), whereas a low airflow will result in a considerable departure from optimum air distribution. The air temperature in some areas of the cockpit may then be *greater* than the outlet air temperature.

APPENDIX 4

Calculation of Cooling System Coefficient of Performance versus Engine Pressure Ratio

A4.1 Turbofan Cooling System

Taking firstly compressor and turbine efficiencies of 1·0, and designating $T_3 = 373$ K, and $T_1 = 338$ K as shown on the temperature–entropy diagram of Figure A2, the COP is then

$$\text{COP} = (T_5 - T_4)/(T_2 - T_1),$$

where T_5 is the cabin outlet temperature, taken to be 303 K.
Then, as

$$T_4 = T_3/r_p^{(\gamma-1)/\gamma},$$

where $r_p = p_2/p_1$, and $T_2 = T_1\, r_p^{(\gamma-1)/\gamma}$, the COP can be evaluated. Using $\eta_c = 0{\cdot}85$; $\eta_t = 0{\cdot}8$,

$$(T_3 - T_4') = 0{\cdot}8(T_3 - T_4),$$

and

$$(T_2' - T_1) = (T_2 - T_1)/0{\cdot}85.$$

The COP can now be re-evaluated using these realistic efficiencies, thus

$$(\text{COP})_{\text{actual}} = (T_5 - T_4')/(T_2' - T_1).$$

A4.2 Bootstrap Cooling System

Referring to Figure A3, the engine compressor bleed air pressure ratio $r_p = p_2/p_1$, the cooling unit compressor pressure ratio $r_c = p_3/p_2$; hence the cooling turbine pressure ratio $r_t = r_c\, r_p$. To evaluate the COP of this system, it is necessary to equate the turbine work to the cooling unit compressor work (taking the mechanical efficiency to be 1·0).

For turbine and compressor efficiencies of 1·0, there results

$$r_p = [(2 - r_c^{0{\cdot}285})\, r_c^{0{\cdot}285}]^{-3{\cdot}5}.$$

The pressure ratio r_c can be plotted versus r_t, from the following values:

r_c	r_p
2	1·2
3	1·7
4	2·6
5	4·25
6	7·8

For the case where the turbine and compressor efficiencies are less than 1·0, we designate

turbine efficiency $\eta_t = 0{\cdot}8$,

cooling system compressor efficiency $\eta_{cc} = 0{\cdot}75$, and

engine compressor efficiency $\eta_c = 0{\cdot}85$.

Now, again equating turbine and cooling system compressor work,

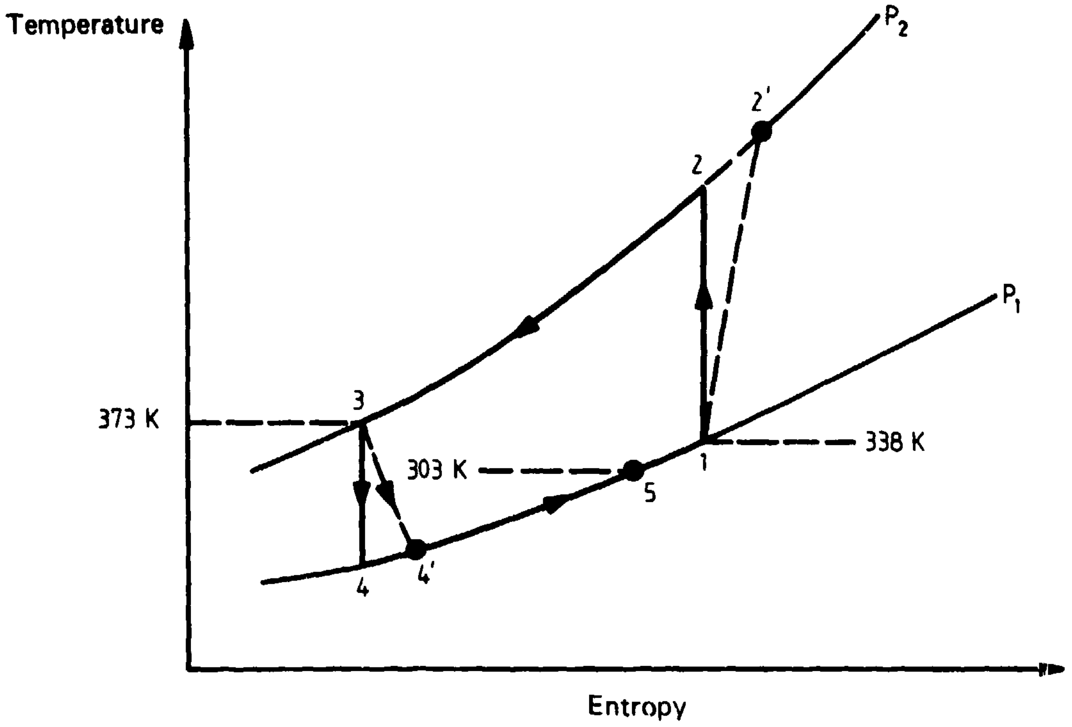

FIG. A2 TEMPERATURE-ENTROPY DIAGRAM OF TURBOFAN SYSTEM

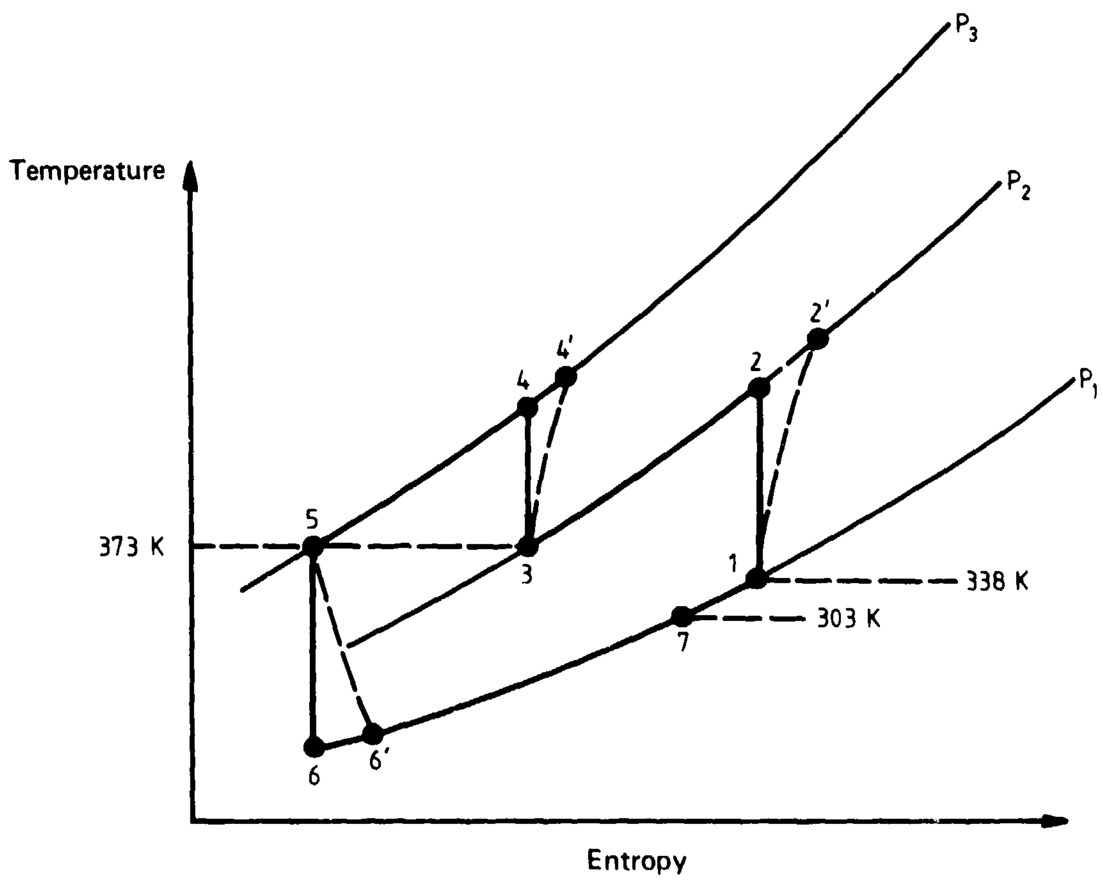

FIG. A3 TEMPERATURE-ENTROPY DIAGRAM OF BOOTSTRAP SYSTEM

$$r_p = [1{\cdot}67 r_c^{0.286} (1{\cdot}6 - r_c^{0.286})]^{-3.5}.$$

There then results:

(*a*)

All efficiencies $= 1{\cdot}0$

r_p	r_t	r_c
2	6·8	3·4
3	12·9	4·3
5	26·5	5·3
10	63·5	6·35
15	103·5	6·9

(*b*)

$\eta_t = 0{\cdot}8$, $\eta_{cc} = 0{\cdot}75$,
$\eta_c = 0{\cdot}85$

r_p	r_t	r_c
2	3·70	1·85
3	6·6	2·2
5	13·0	2·6
10	30·5	3·05
15	49·5	3·3

Thus the system COP can be calculated as previously described. In the tables above the values of r_t at the higher engine pressure ratios are of course not practicable. In practice, a reasonable turbine efficiency can only be maintained up to a pressure ratio of 10 : 1. An engine with a high pressure ratio will in any case use a restrictor on the engine compressor bleed port, limiting the bleed-air pressure.

The resulting values of COP have been plotted in Figure 22.

APPENDIX 5

Summary of United States Specifications and Standards Relating to Cabin Conditioning

A5.1 Relevant Sections from Air Force Systems Command Design Handbook

AFSC DH 2-3	Design Note 4A1—Airconditioning Systems
AFSC DH 2-3	Design Note 4A2—Pressurization Systems
AFSC DH 2-3	Design Note 4B1—Rain, Snow and Insect Removal
AFSC DH 2-3	Design Note 4B2—Anti-icing, Defrosting and Defogging Systems
AFSC DH 2-3	Design Note 4B3—Anti-g, Ventilating, and Pressure Suit Air Supply
AFSC DH 2-3	Section 4C—Engine Bleed Air Systems
AFSC DH 1-6	Design Note 3D3—Bleed Air Systems
AFSC DH 1-6	Design Note 6A3—Noise
AFSC DH 1-6	Design Note 6A5—Temperature and Ventilation

A5.2 Military Standards and Specifications for Airborne Units

MIL-E-38453 A	General Specification for Environmental Control, Environmental Protection, and Engine Bleed Air Systems
MIL-E-18927 (D)	General Requirements for Environmental Control Systems
MIL-E-5400	General Specification for Electronic Equipment
MIL-E-7080	General Specification for Installation of Electrical Equipment
MIL-STD-210 B	Military Standard for Climatic Extremes for Equipment
MIL-T-5842	General Specification for Transparent Areas, Anti-Icing, Defrosting and Defogging Systems
MIL-A-8806	General Specification for Acoustical Noise Level in Aircraft
MIL-T-18606	Test Procedures for Aircraft Cabin Pressurizing and Airconditioning Systems
MIL-A-83116	Specific Performance, Design and Test Requirements for Air Cycle Air conditioning Systems
MIL-STD-890	Environmental Control, Environmental Protection, and Engine Air Bleed Subsystem Performance and Design Requirement Analyses
MIL-H-5484	Aircraft Combustion Type Heater
MIL-F-7872	Bleed Air Leakage Detection*
MIL-E-18927/1 (2)	Environmental System, T-28 Aircraft Cockpit Airconditioning System
MIL-S-6144	General Specification for Installation of Sound-proofing for Aircraft
MIL-I-7171	Thermal Acoustical Insulation Blanket
MIL-H-8796	Aircraft Flexible Air Ducts
MIL-W-7233	Wiper Systems
MIL-R-83056	Liquid Rain Repellants
MIL-R-83055	Repellant Dispenser Systems
MIL-W-006882	Ground Applied Rain Repellants

* The detection of bleed air leakage is intended where a leakage of high temperature air could cause damage to the structure or adjacent components, and when leakage could result in fire and explosion hazards. This is relevant to recent Australian experience with the F-111C (Hutchison 1977)—further details on requirements are given in AFSC DH 2-3 DN 4C1, par. 2.5 (25 August 1975).

A5.3 Military Specifications for Ground Airconditioning Units

MIL-A-8309	Air-Conditioner A/M32C-10A for ground support
MIL-A-38446	Air-Conditioner A/S32C-3, truck mounted
MIL-A-38339	Air-Conditioners, lightweight, compact, general requirements for
MIL-A-27491	Air-Conditioner A/M32C-7, trailer mounted
MIL-A-27490	Air-Conditioner A/M32C-6, trailer mounted
MIL-A-26824	Air-Conditioner M32C-3.
MIL-A-26846	Air-Conditioner M32C-4.
MIL-A-26501	Truck mounted air conditioner A/SC32C-2
MIL-A-26107	Truck mounted air conditioner. 11½ ton. Freon
MIL-A-4914	Trailer mounted air conditioner

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