GUIDELINES FOR THE DESIGN OF AIRCRAFT WINDSHIELD/CANOPY SYSTEMS

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AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433
Chapter Seven
Environmental Design
### CHAPTER 7
**ENVIRONMENTAL DESIGN**

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7-100 INTRODUCTION

The major technical areas of environmental design for windshield and canopy transparencies that will be covered in this chapter include the topics of:

a. Environmental operational requirements.

b. Thermal design considerations.

c. Rain removal systems.

d. Anti-icing systems.

e. Defog and defrost systems.

f. Atmospheric electricity.

g. Electrical system development.
SECTION 2
ENVIRONMENTAL OPERATIONAL REQUIREMENTS

7-200 INTRODUCTION

Environmental operational requirements may vary for each type of aircraft and are dependent on the aircraft's function, mission profile, design usage, and expected service life. The requirements to be established for a transparency must be based on environmental factors that will ensure adequate vision and material survivability under all conditions. These criteria should be applied to the transparency design as early in the aircraft's definition and design phases as possible within the scope of the aircraft's mission profile and flight envelope.

The items addressed in this section include:

a. Climatic temperature criteria.

b. Aerodynamic heating.


d. Pressurization.

e. Hail.

f. Material selection.

7-201 TEMPERATURE CRITERIA

As noted in MIL-A-008860 (Reference 7.1), aircraft transparencies shall be designed to withstand:

b. The effects of aerodynamic heating.

c. The effects of all other heat inputs having an effect on structural integrity.

d. Cumulative effects of the time-temperature-load history of the airplane for its planned service life.

The use of data noted in MIL-STD-210 could be adjusted, depending on the aircraft usage, to fit the conditions noted in Figure 7.1. In some cases for transport type aircraft, it might be advisable to revise the extreme cold temperature from -65°F to -90°F since most transports operate at approximately 40,000 feet altitude.

Figure 7.1. Worldwide Temperature Extremes
Tables 7.1 and 7.2 provide the actual data for worldwide temperature extremes for those noted in MIL-STD-210. Table 7.3 presents the U.S. Standard Atmosphere (Reference 7.3) which may also be required for use along with the high and low data.

These criteria represent "free air" conditions and not aerodynamically induced conditions; e.g., aerodynamic heating. Linear interpolation between adjacent levels is acceptable to obtain extremes for heights not specified. Extremes for altitudes up through 98,000 ft. (30 km) are given for both actual (geometric) and pressure altitude.

These design criteria do not represent internally consistent profiles of the atmosphere typical of the extreme climatic areas, but are rather envelopes of extreme conditions.

The assumed sea level conditions for these U.S. Standard Atmosphere data are: pressure, \( p_0 = 29.91 \text{ in.} \ (760 \text{ mm}) \text{ Hg} = 2,116.22 \text{ lb./ft.}^2 \); mass density, \( p_0 = 0.002378 \text{ slugs/ft.}^3 \ (0.001225 \text{ g/cm}^3) \); \( T_0 = 59^\circ \text{F} \ (15^\circ \text{C}) \).

Additional combat temperature conditions may be derived from the ultimate conditions implied by Chapter 8.

7.007 AERODYNAMIC HEATING

Aerodynamic heating, the result of air friction at supersonic speeds, causes high transparency surface temperatures. Temperature increases with velocity, air density and time until steady state conditions are reached. Surface temperature also varies with the slope of the windshield, the cabin temperature, and the edge structure. The time at velocity must be considered carefully in defining operational requirements, since a short term spike at Mach 3.0, for example, might not be as detrimental to material survival as a long term cruise at Mach 2.5.
### Table 7.1. High Air Temperatures

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<th>Temp (°F)</th>
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(1) Reference 7.3

(2) 1 ft = 0.3048 meters

(3) °C = °F - 32/1.8
Technical reports AFFDL-TR-77-92 and AFFDL-TR-79-3058 (References 7.4 and 7.5) present the results of wind tunnel tests conducted at velocities of Mach 1.6 to Mach 3.0 for a variety of windshield configurations at specific altitudes and exposure times. The surface temperatures varied from approximately 200°F to 500°F, and are an indication of the thermal environment that might be experienced by a supersonic aircraft.

The determination of surface temperatures from an aircraft's flight characteristics may be based on wind tunnel tests or by the use of a heat transfer analysis as described in Section 3. A total temperature gradient through a transparency may be determined with this type of analysis.

7-203 SOLAR RADIATION

Military aircraft may frequently be exposed and affected by solar radiation while parked on ramps at various military installations. Commercial aircraft will also be exposed to solar radiation while parked on a ramp. The temperature affects of this solar radiation should be taken into account when providing a total thermal analysis for aircraft transparencies.

Table 7.4 shows the average solar radiation per day for various parts of the world. This data was excerpted from References 7.6, 7.7 and 7.8.

7-204 PRESSURIZATION

The internal and external pressurization and flight loads must be established in accordance with criteria established by MIL-STD-1530 and FAR 25 as described in Chapter 4. The frequency of these loads is very important and greatly influences the selection of material, material thicknesses, and transparency service life.
### TABLE 7.4. SOLAR RADIATION ANNUAL AVERAGE FOR DAILY TOTAL

<table>
<thead>
<tr>
<th>LOCATION</th>
<th>LANGLEYS/DAY</th>
</tr>
</thead>
<tbody>
<tr>
<td>Worldwide Standard</td>
<td>420</td>
</tr>
<tr>
<td>Philippines</td>
<td>383</td>
</tr>
<tr>
<td>Germany</td>
<td>254</td>
</tr>
<tr>
<td>Inland Plains, USA (Montana)</td>
<td>342</td>
</tr>
<tr>
<td>U.S. Desert (California)</td>
<td>496</td>
</tr>
<tr>
<td>U.S. Coastal (Florida)</td>
<td>400</td>
</tr>
<tr>
<td>Arctic (Alaska)</td>
<td>208.3</td>
</tr>
<tr>
<td>Equatorial (Panama)</td>
<td>426</td>
</tr>
<tr>
<td>Middle East</td>
<td>510</td>
</tr>
</tbody>
</table>

7-205 HAIL IMPACT

The mission profile and flight envelope of an aircraft must be studied to determine whether or not a hail encounter is imminent during the life of the airplane.

An expression to determine the risk of a hail encounter has been published in Reference 7.9. This expression is based on the assumptions that no avoiding action was taken, that the diameter of a hail cell or shower was 1 nautical mile, and that the duration of a hail shower at a single location was 0.1 hour. The expression derived was that \( N_0 \), the number of encounters per flight hour with hail which is \( x \) inches or greater in diameter, will be:

\[
N_0 = 7.26 \times (10^{-5})NVP_x \quad (7.1)
\]
where:

\[ N = \text{Number of thunderstorms per year on the ground at the geographical area considered.} \]

\[ V = \text{Aircraft speed in knots at the altitude considered.} \]

\[ P_x = \text{Probability of occurrence of hail diameter } x \text{ inches or greater during the storm at the altitude and geographical area considered. } P_x \text{ is assumed to be constant up to mid tropopause and to decrease by one order of magnitude very } 10,000 \text{ feet above that level.} \]

Then:

\[ P_x = P_o 10^{-X} \quad (7.2) \]

The suggested values for \( N \), mid-tropopause height \( H \), and \( P_1 \) (the probability of occurrence of hail of 1 inch diameter or greater) are tabulated below:

<table>
<thead>
<tr>
<th>Region</th>
<th>( N )</th>
<th>( H )</th>
<th>( P_1 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>UK</td>
<td>15</td>
<td>20,000 ft.</td>
<td></td>
</tr>
<tr>
<td>Europe</td>
<td>25</td>
<td>20,000 ft.</td>
<td>( 1.25 \times 10^{-3} )</td>
</tr>
<tr>
<td>Denver</td>
<td>80</td>
<td>25,000 ft.</td>
<td></td>
</tr>
<tr>
<td>Singapore</td>
<td>100</td>
<td>30,000 ft.</td>
<td>( 2.5 \times 10^{-4} )</td>
</tr>
</tbody>
</table>

Criteria for hail size are presented in Reference 7.2. A summary of this data is presented in Table 7.5 for a worldwide environment.
### TABLE 7.5. HAIL SIZE(1)

<table>
<thead>
<tr>
<th>RISK</th>
<th>ALTITUDE</th>
<th>HAILSTONE SIZE</th>
<th>WEIGHT(2)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1%</td>
<td>Ground</td>
<td>2 cm (0.8&quot;)</td>
<td>.009 Lb</td>
</tr>
<tr>
<td>.1%</td>
<td>Ground</td>
<td>5 (2.0&quot;)</td>
<td>.139 Lb</td>
</tr>
<tr>
<td>.1%</td>
<td>Ground (1)</td>
<td>14.2 cm (5.6&quot;)</td>
<td>3.05 Lb</td>
</tr>
<tr>
<td>1%</td>
<td>S.L.</td>
<td>1.1 cm (0.42&quot;)</td>
<td>.0013 Lb</td>
</tr>
<tr>
<td>1%</td>
<td>4 km (13,100')</td>
<td>2.1 cm (0.81&quot;)</td>
<td>.0092 Lb</td>
</tr>
<tr>
<td>1%</td>
<td>6 km (19,700')</td>
<td>2.1 cm (0.81&quot;)</td>
<td>.0092 Lb</td>
</tr>
<tr>
<td>.1%</td>
<td>S.L.</td>
<td>3 cm (1.2&quot;)</td>
<td>.031 Lb</td>
</tr>
<tr>
<td>.1%</td>
<td>4 km (13,100')</td>
<td>6.1 cm (2.4&quot;)</td>
<td>.24 Lb</td>
</tr>
<tr>
<td>.1%</td>
<td>6 km (19,700')</td>
<td>6.1 cm (2.4&quot;)</td>
<td>.24 Lb</td>
</tr>
</tbody>
</table>

NOTES:  
(1) Data is from Reference 7.2.  
(2) Hail/ice density = 57.3 Lb/ft².

### 7-206 MATERIAL CONSIDERATIONS

The scope of operational requirements as applied to high and low temperature exposure is the delineation of the aircraft’s total flight envelope—velocity, altitude and time; maximum and minimum ground temperatures; and all other temperature related environments that a transparency would be exposed to during its expected life.

Since the adverse effects of high temperatures are cumulative for certain transparency materials, a temperature-time summation must be completed for the aircraft’s total life. This analysis should include the effects of aerodynamic heating, solar radiation and hot air removal systems (see Section 1) in addition to the pressurization and other flight loading conditions.
The allowable stresses used for design of the windshield system shall include the effects of material strength reduction due to expected long-term, short-term, and repeated exposure to elevated temperatures in combination with applicable loads consistent with the planned employment and shall include the effects of creep, thermal expansion, joint fastener relaxation and elevated temperature fatigue.

As noted in Technical Report AFFDL-TR-77-96 (Reference 7.10), polycarbonate material is affected by in-service aging and tends to become less ductile when exposed to temperatures above 176°F (80°C). This was evidenced by accumulative decreases in impact strength and fracture energy, and increases in tensile yield strength. Other evidence indicated a degrading of mechanical properties due to weathering and storage.

The use of stretched acrylic must be carefully considered. If it is heated above 220°F it will begin to revert to its as-cast dimensions prior to stretching. The deleterious effects of a high temperature environment on stretched acrylic material were observed during wind tunnel tests reported in References 7.4 and 7.5. Cyclic and thermal loading conditions are cumulative and greatly affect the service life of transparencies designed with stretched acrylic materials as noted in Chapter 6.

To meet hail impact requirements, it has been shown by experience and testing that 0.100 glass will withstand the impact. The probable affect of hail on as-cast acrylic face plies and coatings may require testing to satisfy design conditions of a specific aircraft.
SECTION 3
THERMAL DESIGN CONSIDERATIONS

7-300 INTRODUCTION

During aircraft operations, the transparencies can be exposed to extreme hot and extreme cold temperatures. The windshield or canopy must survive all these normal thermal conditions to which the aircraft is exposed and suffer no deleterious effects as a consequence. The thermal conditions, including maximum, minimum and transient temperature conditions are discussed in Section 2. The windshield or canopy must also withstand the thermal effects of solar heating and those due to combat conditions (if the aircraft is to operate in a combat environment).

Early definition of a temperature-time profile is essential to the completion of the thermal studies used to select materials that will survive the aircraft's intended flight regime, while fulfilling its optical specifications without need for costly revisions in the later stages of development.

Maximum and minimum flight temperatures are necessary to determine transparency material allowables, especially those that would apply during a transparency birdstrike.

Temperature gradients are necessary for the design of laminated configurations where thermal expansion and contraction may lead to delamination of the plies, especially at the transparency edges.

The survivability limits for current transparency materials, polycarbonate and acrylic, are being approached as aircraft speeds increase. The effect of temperature on these materials is time related — hence, the need for an accurate mission profile. The interior surface temperature of aircraft transparencies must not exceed a temperature of 160°F to prevent crew member injury, but lower temperatures are preferred for crew comfort.
Due to the low thermal conductivity of the various transparency materials in combination with the high thermal mass of a transparency, high thermal stresses may be induced within the transparency by the large temperature differentials that occur during transient states of the aircraft; such as velocity during descent and climb and the minimum time of descent and climb, deceleration and acceleration rates between minimum velocity and maximum velocity, plus initial warm-up of the aircraft. The heating due to the anti-icing system is an important factor to be considered. The combat exposure design requirements are also needed as noted in Chapter 8.

At elevated temperatures which occur at supersonic flight, materials may change mechanical, or to a lesser extent, optical characteristics. Minimum temperatures may also present design problems. The materials selected must withstand the temperature range and thermal shocks experienced throughout the operational envelope of the aircraft. To do this, the aircraft operational envelope as a function of altitude, velocity and ambient temperature must be known. The actual temperatures must be derived from conditions noted in Section 2.

The following 15 conditions, including exposure times for the various conditions, should be analyzed for material temperatures to determine thermal compatibility of the windshield or canopy. These conditions may be analyzed by one-dimensional heat transfer analysis.

1. Ground soak at minimum temperatures.

2. Cruise or a hold condition at an altitude where maximum subsonic velocity and minimum ambient temperature produces a minimum total air temperature.
3. Continuous high velocity at an altitude where the high ambient temperature produces a maximum total air temperature.

4. Ground soak at maximum temperature, including solar heating effects.

5. Maximum velocity at an altitude that produced maximum total air temperature (supersonic dash). NOTE: This should follow a steady state high temperature condition and return to it.

6. Ground soak at minimum temperature, followed by warm-up of the aircraft, followed by takeoff and climb as quickly as possible. This should include warm-up of anti-icing (refer to Section 5) and defog system (refer to Section 6 if applicable).

7. Cruise or hold at an altitude that produces a minimum total air temperatures, followed by a maximum acceleration rate and descent or climb to a maximum velocity at an altitude that produces a maximum total air temperature (Condition #2 followed by #5). This includes going from a MIL-STD-210B cold atmosphere to the hot atmosphere. This may include selection ON of the anti-ice and defog systems to an appropriate level for mechanical characteristics and may be required for possible changes in optical characteristics.

8. The reverse of Condition #7 using a maximum deceleration rate. Anti-ice and defog systems should be off.

9. Condition #4 followed by start-up, takeoff and climb as quickly as possible.

10. Condition #2 with anti-ice and defog system OFF and then selecting ON.

11. Condition #2 with anti-ice and defog ON and then selecting OFF.
12. Condition #2 followed by a maximum rate descent to 8000 feet high velocity and minimum ambient temperature (for birdstrike testing).

13. Condition #5 followed by a maximum rate descent to 8000 feet high velocity and maximum ambient temperature (for birdstrike testing).

14. Effects of thermal energy from combat (nuclear, laser, etc.).

15. Normal mission utilization rates and expected life of the aircraft - to be used in cyclic testing.

More detailed thermal analysis (3-dimensional heat transfer) is usually required in the area of the edge attachments. This information is used for stress analysis and for developing thermal conditions for edge attachment testing. Conditions #7, #8 and #15 should be analyzed in this manner.

To analyze for the conditions of #15 above, the normal mission profiles of the aircraft must be specified. This analysis is used in developing the conditions for cyclic testing of the windshield or canopy with regards to temperature and pressure.

7-302 THERMAL ANALYSIS DISCUSSION

To analyze conditions 1 through 4 above, the following parameters must be known: (an alternate approach to that outlined below can be obtained from Reference 7.11):

a. Velocity of the aircraft (from flight profile).

b. Ambient temperature encountered by the aircraft (Section 2).
d. The temperature in the cockpit (Section 2 or Procuring Agency).

e. The thermal conductivity and thickness of the individual materials in the transparency (Chapter 6).

f. Location of the transparency with respect to the nose of the aircraft (from aircraft conceptual design).

g. Spectral transmissivity, absorptivity and surface emittance of transparent materials (Chapter 8).

These four conditions have steady state solutions: the external heat transfer coefficient; the thermal conductance of the transparency; the internal heat transfer coefficient; and the temperatures on either side of the transparency are then used to perform the heat balance. NOTE: For simplification without a large error, this equation does not include transmission and radiation to space and solar heat flux. These variables become of greater importance when evaluating transparency and crew compartment temperatures when on the ground.

\[
\frac{Q}{A} = \frac{1}{\frac{1}{h_0} + \sum_{j=1}^{N} \frac{X_j}{K_j} + \frac{1}{h_i}} (T_c - T_o) = h_o (T_{so} - T_o)
\]

\[
= \frac{1}{\frac{1}{N} \sum_{j=1}^{X_i} \frac{1}{K_j} j} (T_{si} - T_{so}) = h_i (T_c - T_{si})
\]  

\[ (7.3) \]
Where:

\[ Q = \text{the heat flow through the transparency BTU/HR.} \]
\[ A = \text{the surface area of the transparency - ft}^2. \]
\[ h_0 = \text{the external heat transfer coefficient - BTU/HR-FT}^2 - ^\circ\text{F (from Equation 7.4).} \]
\[ j = \text{denotes individual plies in the transparency.} \]
\[ N = \text{the number of plies in the transparency.} \]
\[ X = \text{the thickness of each ply - inches.} \]
\[ K = \text{the thermal conductivity of each ply - BTU-IN/HR-FT}^2 - ^\circ\text{F.} \]

**NOTE:** Thermal conductivities for various transparencies are shown in Chapter 6.

\[ h_i = \text{the internal surface heat transfer coefficient (can be assumed at 2 BTU/HR-FT.}^2 - ^\circ\text{F for static air).} \]
\[ T_c = \text{the cockpit temperature - } ^\circ\text{F.} \]
\[ T_0 = \text{the external recovery temperature - } ^\circ\text{F (from Equation 7.5).} \]
\[ T_{so} = \text{the external surface temperature of the transparency - } ^\circ\text{F.} \]
\[ T_{si} = \text{The interior surface temperature of the transparency - } ^\circ\text{F.} \]
The value of 2 noted as the value for $h_i$ is a nominal that includes the free convective heat transfer coefficients and the effect of radiation.

The external heat transfer coefficient is defined by the equation (Reference 7.11, Equation 3F-8):

$$h_o = 0.51 \times T_o^{0.3} \times (V_o \rho)^{0.8} \times 1.15 / S^{0.2}$$

(7.4)

where:

- $V_o$ = free-stream velocity - FEET/SEC.
- $\rho$ = ambient air density - LBS/FT.$^3$
- $S$ = distance from stagnation point on the nose of the aircraft to the transparency - FEET.

The factor 1.15 is the factor which transforms the basic equation for a flat plate to that for a cone. If the nose of the airplane does not resemble the shape of a cone, delete the 1.15 from the equation.

To compute $T_o$, the following equation may be used:

$$T_o = T_{\infty} \left[ 1 + rf \left( \frac{\gamma - 1}{2} \right) M^2 \right]$$

(7.5)

where:

- $\gamma$ = 1.4 for air.
- $rf$ = recovery factor = 0.90 for the nose of most aircraft, while the total range might be = 0.87 to 0.95.
- $T_{\infty}$ = ambient air temperature - OF.
- $M$ = Mach number.
For conditions #5 through #15 a transient heat transfer analysis must be performed. This should be done with a transient heat transfer computer program. In addition to the parameters needed for steady state analysis the following are also needed.

a. Specific heat and density of the transparency material (refer to Chapter 6).

b. Flight profile, including velocity and times.

NOTE: Conditions 5 through 15 may be done with one dimensional analysis. The edge attachment analysis must be done three-dimensionally to study edge effects as well as heat transfer through the transparency.

For aircraft velocities greater than Mach 2.0, radiation starts to be an important factor. This means an additional heat loss must be considered (References 7.11, Equation 3A-44):

\[
\frac{Q_R}{A} = 0.178 \varepsilon \left(\frac{T_{SO} + 460}{100}\right)^4
\]

(7.6)

where:

\(Q_R\) = radiation heat loss - BTU/HR.

\(\varepsilon\) = emissivity of the transparency.
To compute the effect of a nuclear blast on the temperature of a transparency, the thermal energy as a function of time must be specified. This is usually specified in heat flow per unit area (Q/A) and can be used as a heat gain to the outer surface. Experience has shown that a nuclear blast of a magnitude that would overheat the transparency would most likely destroy the aircraft. However, this should be considered for aircraft that are designed to resist a certain level of nuclear blast. Additional information may be found in Chapter 8.
All aircraft encounter rain. Provisions must be made to provide adequate visibility to maneuver the aircraft during rain encounters. FAR 25 (Reference 7.12) translates this need to a rain removal system which "...must be designed to function, without continuous attention... in heavy rain at speeds up to 1.6 \( V_s \), where \( V_s \) is the stalling speed with flaps retracted." In addition to a "sufficiently extensive view along the flight path in normal flight attitudes..." MIL-E-38453A (Reference 7.13) further delineates this requirement to specify "...the area of the pilot and co-pilot's windshield required for an adequate field of vision in heavy rain (0.59 inch per hour, 1,500-micron-median-droplet diameter)...for several specified flight attitudes." These attitudes include:

a. Ground taxi
b. Takeoff
c. Landing approach
d. Landing
e. In-flight refueling on aircraft where this will be accomplished at altitudes below 20,000 feet.
f. Level flight at 1.6 times the stall speed at maximum weight with flaps and gear retracted for fixed wing aircraft.
g. Maximum cruise speed for rotary wing aircraft.

In addition to the above, the rain removal system must be adequate to provide sufficient visibility for landing during an excessive rain (defined as 1.6 inches per hour, 2,300-micron-median-droplet diameter, in MIL-E-38453A).
The area requiring clearance and the degree of clearance necessary for each of these ground and flight conditions must be established early in the aircraft development program through the use of a cockpit mockup and rain tunnel testing. In addition to the required clearance capability, the rain removal provisions must not be damaged by flight velocities within the anticipated flight envelope.

Rain removal systems which have successfully been utilized on aircraft include wipers, jet blast systems, inflight applied rain repellant, and ground applied rain repellant.

7-402 WIPER DESIGN DISCUSSION

A wiper system, with or without an inflight applied rain repellant system, has been used on many aircraft. Commercial jets and military transports are the main aircraft which use a wiper system. General design requirements for electrical windshield wiper systems are contained in MIL-W-7233 (Reference 7.14). The advantage of the wiper is that it provides good visibility during rain. A disadvantage is that it is compatible only with glass face plies, because softer materials, such as acrylics or polycarbonates, are susceptible to abrasion. In addition, a wiper system is not compatible with highly curved surfaces as would be encountered with a canopy. Stowing the wiper blades on high velocity aircraft becomes a complicated design problem.

7-402.1 Windshield Wiper Clearance Envelope

Aerospace Standard 580A (Reference 7.15) defines the minimum area to be cleared by a windshield wiper as a section of the forward windshield panel 10 degrees up to 15 degrees down and between 15 degrees left and 15 degrees right, or an equivalent area. Angles are taken from the pilot's design eye position with respect to a horizontal plane and a vertical plane parallel to the aircraft's longitudinal axis.
7-402.2 Windshield Wiper System Design

Electrically operated windshield wiper installations must conform to the requirements of MIL-W-7233 (Reference 7.14) and shall be adapted for proper fit and operation to the particular application. Hydraulically operated windshield wipers shall not be used if the installation will involve routing high pressure hydraulic supply lines through the cockpit, unless adequate shrouding is provided with overboard drainage, the lines are well anchored, and the flow control valve is located outside of the crew compartment. A completely separate wiper system for each windshield shall be provided in side by side cockpit aircraft.

An electrical windshield wiper system must be operated by the available aircraft accessories. Per MIL-STD-704 (Reference 7.16) the power source may be either a 28 V dc or a 400 Hz, three phase 115/200 V ac as the supply source.

The windshield system must be designed and constructed so that no parts will work loose in service. It is to be designed and built to withstand the strains, jars, vibrations and other conditions incident to shipping, storage, installation and service. The mean-time-between-failures (MTBF) for the windshield wiper subsystem and components thereof should be equal to or greater than the value necessary for achievement of the required reliability of the overall system. The maintainability characteristics of the windshield wiper subsystem should be such that the required aircraft missions may be accomplished with a minimum of maintenance manhours and personnel skills, ground equipment and technical data.

7-402.3 Jet Blast System Design

A jet blast system, with or without an inflight applied rain repellent system, has been used on many aircraft where wipers cannot be used and/or high pressure air was readily available. The system consists of a high pressure air source which supplies air to a nozzle
at the forward base of the transparency. The nozzle may consist of a line of orifices, a tapered or untapered converging slot, or a line of supersonic nozzles. Tests have shown that the row of orifices is the least effective and the tapered converging slot is the most effective. The advantages of this system are that it is compatible with all shapes and sizes of windshields or canopies and, in conjunction with a chemical rain repellant system, it provides good visibility. The disadvantages are that the system design usually requires development work and testing, and that the design is critical for assuring that the transparency is not overheated by the hot, high pressure impinging air or operation during dry weather.

Adequate flow and pressure must be available for this method during all ground and flight conditions including the minimum engine power settings normally associated with descent, flareout, touchdown and taxi. It is recommended that a flow rate of at least 7 pounds per minute per inch of nozzle length and sonic flow through nozzles should be maintained at all conditions where rain removal is required. Excessive temperatures must be sensed and indicated to warn the crewmembers of impending windshield overheat. A normally closed shutoff valve that will fail closed shall be provided in the jet blast air supply line to each cleared windshield. Airflows to both windshields in side by side cockpit aircraft must not be lost due to failure of a flow control or pressure regulating device.

7-402.4 Inflight Applied Rain Repellant System Design

The inflight rain repellant is applied through a nozzle located at the forward lower edge of the windshield or canopy. The rain repellant is forced through the nozzle by high pressure air. A wiper or jet blast is required to assist in distributing the repellant across the transparency at static and slow speed operations. Noted in MIL-R-83056 (Reference 7.17) is a description of the fluids that may be used for application to military aircraft. The design of the
dispensing system for military aircraft must be in accordance with MIL-R-83055 (Reference 7.18) such that it will "...deposit the required amount of fluid at the correct windshield location to give the pilot of an aircraft improved forward vision through the windshield during flight in rain."

Some repellants currently in use when applied must be done during actual rain conditions and the wipers operated to make the solution operate efficiently. This particular repellant, if applied to the surface of the windshield when dry, will cause the repellant to stick to the windshield and the view through the windshield will be somewhat translucent and distorted. If at all possible, the repellant material selected should be capable of being applied in a dry, wet, hot, or cold environment without causing any reduction of visibility through the windshield.

7-402.5 Ground Applied Rain Repellant System

The final method of rain removal is a ground applied rain repellent fluid. This system provides adequate visibility and adds no weight or complexity to the aircraft. The drawbacks are that it is not permanent and requires frequent reapplication to maintain its rain repelling qualities; therefore, the transparencies to be protected must be readily accessible.
SECTION 5
ANTI-ICING SYSTEMS

7-500 INTRODUCTION

During operation, an aircraft can penetrate icing clouds which contain supercooled water droplets. These droplets can impinge on and freeze to the transparencies and other surfaces and reduce visibility. An anti-icing system is needed to maintain visibility during and following an icing encounter.

Anti-icing refers to the prevention of ice buildup on the aircraft's windshield or canopy and all transparent areas essential to the mission of the aircraft, by the process of either evaporating the impinging water or allowing it to run back and freeze on a noncritical area. The aircraft's mission requirements must be defined and studied to determine anti-icing requirements.

Tests have shown that to de-ice a transparency during subsonic flight while in icing conditions is an impossibility. This condition is particularly true on a small transparency when ice has formed, attached itself to adjacent structure, and bridges the transparency.

A windshield and/or canopy must be designed to meet the intent of MIL-T-5842, (Reference 7.19) if the aircraft is to be designed as an all-weather aircraft. Therefore, before designing an anti-icing system, the requirements for the aircraft must be assessed. If an anti-icing system is to be designed, it must perform satisfactorily in the environments defined in MIL-E-38453A (Reference 7.13). If the aircraft is to be designed under civil regulations, it must meet the requirements of Federal Aviation Regulations Part 25 (Reference 7.12).

Subsonic cargo/transport aircraft commonly use electrically heated laminated windshields. Typically, these systems consist of
vacuum deposited metallic coatings applied to a glass surface equipped with contacts and sensing devices. The coated glass plate is then laminated to other plates of glass or plastic depending on the specific windshield design.

Many current fighter type aircraft use a monolithic transparency. These aircraft either rely on a hot air blast anti-icing system or operate without an anti-icing capability. The decision to fly with or without anti-icing capability should be made early in the development of the aircraft.

The operating requirements for an anti-icing system require that it be capable of operation during engine warm-up, taxiing, takeoff, touchdown as well as during normal flight.

7-501 ANTI-ICING DESIGN DISCUSSION

There are four types of anti-icing systems that have been proposed and/or used in the past for anti-icing of aircraft transparencies. One system which has been found to be generally inadequate by itself, but can be used as a back-up, is the use of an external jet blast. This system could be feasible when it is used on small windshields. Attempting to anti-ice the windshield without overheating it is a design problem. This system may be considered as marginal and only used when a jet blast rain removal system is installed. In many cases, air flow and temperature requirements for anti-icing are less severe than for rain clearing.

Another system, which was used on early aircraft, is a double paned windshield with hot air flowing between the panes. This system was adequate in performing as intended, but, due to the many surfaces, produced multiple images. Also, the air passing between the panes required special attention to prevent discoloration of the panes, humidity formation and the depositing of dust in the gap.
A system which has been proposed for windshields and canopies, but not used often, if at all, is a fluid de-icing system. This system uses a freezing point depressant (such as an ethylene glycol/water mixture) sprayed onto the windshield. This produces a slush which is then blown off the windshield by aerodynamic forces. The principle disadvantage is the residue left by the fluid and the large quantities of fluids needed to be carried on board.

The final system, the one that is used most frequently, is an electrical anti-ice system. The system consists of a conductive coating on the inboard surface of the external face ply, sandwiched between the interlayer and the face ply. Power is supplied to the conductive coating through two bus bars at either edge of the coating. The system requires a controller to regulate the power applied to the windshield, which ensures adequate anti-icing performance and prevents overheat of the transparency. The disadvantages are tied to the initial design compatibility of the temperature, electrical system and structural integrity. Refer to Section 8 for the design of an electrical system that will meet the requirements.

7-502 ANTI-ICING ANALYSIS DISCUSSION

For the jet blast system and the fluid system, development test must be run. Adequate analytical methods have not been developed. For the electrical and hot air systems, the basic analysis is the same. A heat balance must be performed on the external surface.

The heat loss per unit area from the outer surface is (Reference 7.11):

\[ \frac{Q}{A} = h_0 (T_{so} - T_0) + 2.9 h_0 L_e \left( \frac{e_{sw} - e_{sw0}}{P_m} \right) + W_p C_p (T_{so} - T_0) \] (7.7)
where:

\[ Q = \text{heat flow rate from windshield - BTU/HR.} \]

\[ A = \text{area of windshield - FT}^2 \]

\[ T_{so} = \text{outboard surface temperature of windshield - OF (must be greater than 320F to remain ice free).} \]

\[ T_0 = \text{ram air temperature - OF.} \]

\[ L_e = \text{heat of vaporization of water - BTU/LB.} \]

\[ e_{sw} = \text{vapor pressure of water at } T_{so} - \text{PSI.} \]

\[ e_{aw} = \text{vapor pressure of water at ambient temperature - PSI.} \]

\[ P_\infty = \text{ambient pressure - PSI.} \]

\[ W_B = \text{rate of water impingement - LB/HR-FT}^2 \text{ (from Equation 7.10).} \]

\[ C_{pw} = \text{specific heat of water - BTU/LB - OF.} \]

\[ T_\infty = \text{ambient static air temperature.} \]

The external heat transfer coefficient (h₀), based on turbulent flow over a cone, is given in equation (7.4).

During the instance of low water impingement rate on the windshield, the heat-flow rate may be sufficient to evaporate all the water. In this case, the second term of equation (7.7) will change to \( W_B L_e \), which is the amount of heat required to evaporate all the impinging water.
The water collection rate \((W_B)\) on the windshield during a particular icing flight condition is determined by the method given in Reference 7.20. This method uses the droplet Reynolds number \((R_{ed})\) and the modified droplet inertia parameter \((K_o)\).

\[
R_{ed} = \frac{2a \rho V_o}{\mu}
\]

(7.8)

where:

- \(a\) = droplet radius - FEET (nominally 3 to 50 microns).
- \(\mu\) = air viscosity - LB/FT-SEC.

\[
K_o = \frac{2a \rho V_o \lambda}{9 \mu \ell \lambda_s}
\]

(7.9)

where:

- \(\rho_w\) = water density - LB/FT³.
- \(\ell\) = height of windshield - FEET.
- \(\frac{\lambda}{\lambda_s}\) = droplet range ratio - dimensionless (Reference 7.20).

With \(K_o\) established, the collection efficiency \((E_m)\) is found from Figure 5 in Reference 7.20.

The following equation is then used to determine \(W_B\):

\[
W_B = 0.33 E_m \frac{A}{P} LWC U_o
\]

(7.10)
where:

\[ A_p \] = forward projected windshield area - \( \text{FT}^2 \).

\[ \text{LWC} \] = liquid water content of the icing cloud - GN/\( \text{M}^3 \).

\[ U_o \] = free stream velocity - MPH.

The above equations define the heat loss from the external surface of the transparency. For both the hot air and the electrical anti-ice systems, an amount of heat must be applied to the internal surface of the external ply to provide no less than a 320°F temperature on the outer surface of the transparency or to evaporate all the impinging water.

**7-502.1 Hot Air Gap System**

Addressing first the hot air gap system, the equation for heat transfer from the air to the external ply is:

\[ \frac{Q}{A} = h_a (T_a - T_{go}) \] (7.11)

where:

\[ h_a \] = average heat transfer coefficient from hot air to external ply - BTU/HR-\( \text{FT}^2 \) - °F (use turbulent flow in a flat duct).

\[ T_a \] = average air temperature - °F (from Equation 7.14).

\[ T_{go} \] = temperature of outer surface of air gap - °F.
The heat transfer to the external ply equals the heat transferred through the ply:

\[ \frac{Q}{A} = h_a (T_a - T_{go}) = \frac{K}{x} (T_{go} - T_{so}) \quad (7.12) \]

where:

- \( K \) = thermal conductivity of external ply - BTU-IN/HR-FT²-OF.
- \( x \) = thickness of external ply - INCHES
- \( T_{so} \) = external surface temperature - OF (should be above 320F).

By combining the two right hand portions of Equation 7.12:

\[ \frac{Q}{A} = \frac{1}{h_a + K} \left( T_a - T_{so} \right) \quad (7.13) \]

Equation 7.13 is set equal to Equation 7.7 and the external surface temperature \( T_{so} \) is set equal to 320F to determine the heat transfer coefficient \( h_a \) and average air temperature \( T_a \) needed to anti-ice the transparency.

The average hot air temperature is equal to the average of the entering air temperature and the exit air temperature.

\[ T_a = \frac{T_{in} + T_{out}}{2} \quad (7.14) \]
where:

\( T_{\text{in}} = \) entering air temperature - °F.

\( T_{\text{out}} = \) exit air temperature - °F.

The air flow is determined by the following equation which shows the heat loss from the air:

\[
W C_p (T_{\text{in}} - T_{\text{out}}) = h_a A (T_a - T_{\text{go}}) + h_a A (T_a - T_{\text{gi}})
\] (7.15)

where:

\( W = \) hot air flow rate LB/HR.

\( C_p = \) specific heat of air BTU/LB - °F.

\( T_{\text{gi}} = \) temperature of inner surface of air gap - °F.

The heat loss inboard is given as:

\[
h_a (T_a - T_{\text{gi}}) = \frac{1}{\frac{1}{h_a} + \frac{N}{j=1} \frac{X_j}{K_j} + \frac{1}{h_i}} (T_{\text{gi}} - T_c)
\] (7.16)

where:

\( N = \) number of plies inboard of the air gap.
Enough air must be introduced into the air gap to maintain the entering and exit air temperature close to the required average air temperature. This prevents overheating near the inlet and insufficient anti-ice capability near the exit.

If the hot air is left on continuously, the heat flow inboard will maintain the flight compartment surface of the transparency fog free.

7-502.2 Transparency Electrical System

The temperature of the electrically conductive film required to keep the windshield or canopy ice free can be determined by setting the following equation equal to Equation 7.7 and $T_{so}$ equal to 32°F.

$$Q/A = \frac{K}{X} (T_f - T_{so}) \quad (7.17)$$

where:

$K$ = thermal conductivity of external ply - BTU-IN/HR-FT$^2$. -°F.

$X$ = thickness of external ply - INCHES.

$T_f$ = average conductive film temperature - °F.

The heat required to be applied to the conductive film is shown in this equation:

$$\frac{Q}{A} = \frac{K}{X} (T_f - T_{so}) + \frac{1}{N} \sum_{i=1}^{N} \frac{X_i}{h_i} (T_f - T_c) \quad (7.18)$$

7.041
The film temperature ($T_f$) is the average temperature of the conductive film. The film has a nonuniformity which is defined by the heat transfer quality control constants ($K_a$, $K_h$, $K_c$ and $K_m$). $K_m$ is the ratio of average power density to hot spot power density, $K_a$ is the ratio of the average power density to the power density at the control point, $K_h$ is the ratio of the power density at the hot spot to the power density at the control point, and $K_c$ is the ratio of the power density at the cold spot to the power density at the control point. The control point is defined as the point on the film where the control sensor is located. $K_m$ is dependent only on the uniformity of the film, while $K_a$, $K_h$ and $K_c$ are dependent upon the location selected for the control point as well as the uniformity. The control point temperature ($T_{cp}$) is determined by solving the following equations:

\[
(Q)_{cp} = (Q)_{ave} \times \frac{1}{K_a}
\]  

(7.19)

where:

\[
(Q)_{ave}
\] is defined by Equation (7.12):

\[
(Q)_{cp} = \frac{K}{X} (T_{cp} - T_{so}) + \frac{1}{h_i + \sum_{j=1}^{N} \frac{X_i}{K_j}} (T_{cp} - T_c)
\]  

(7.20)

and:

\[
\frac{K}{X} (T_{cp} - T_{so}) = h_o (T_{so} - T_o) + 2.9 h_o L_e \left( \frac{e_{SW} - e_{MW}}{P_{in}} \right) + W_{p} C_{pw} (T_{x_o} - T_w)
\]  

(7.21)
This control point temperature is the value that should be used to set the sensor temperature. The same type of analysis should be used to determine the hot spot temperature for the possibility of overheating the transparency and the cold spot temperature to check the anti-icing performance.

This analysis should be performed to predict what values of heat transfer quality control constants should be imposed on a transparency specification to provide an acceptable design.

An electronic controller must be designed to control the heat to the windshield and to detect failures in the windshield. See Section 8 for a discussion of electrical components.

7-503 SUMMARY

The above described analysis must be performed at several points in the continuous and intermittent icing envelopes to determine the design conditions. In continuous icing, the windshield should be maintained ice free because of the long duration in an icing cloud. In intermittent icing, below −220°F, it may be permissible to allow some ice accretion. During these conditions, the liquid water content is low and the time in the cloud is short. Therefore, the ice accretion will be very low, will not completely obscure vision and will sublime off the transparency in a short time (usually less than 5 minutes). Also, the areas around the control point and the hot spot will normally be ice free.
7.600 INTRODUCTION

Fog or frost will form on the inside surface of an aircraft windshield or canopy when the inside surface temperature is below the cockpit air dew point. To maintain adequate visibility for the flight crew during all phases of flight, a defog or defrost system for the internal surface of the windshield must be provided. The defog system will be designed to adequately maintain good visibility during all phases of flight, but especially during taxiing, takeoff, approach and landing. Also vision areas for combat actions such as evasion and formation flights, and special purposes such as reconnaissance and spatial checks must be continuously cleared by monitoring the surface temperature above the surrounding air dew point temperatures.

7-601 DEFOG DESIGN DISCUSSION

There are several methods which may be used to maintain the surface fog free. If the hot air double-paned anti-ice system or electrical anti-ice system is installed and operates continuously, the inboard surface will remain fog free during most conditions. A separate defog system should be designed for the conditions where the anti-ice system is not installed, does not operate continuously, or the anti-ice system is not adequate for defogging. Two defog systems that are frequently used are the hot air and electrical systems.

The hot air system consists of a slot fed by a plenum, or a row of piccolo holes in a tube blowing warm air at or parallel to the inboard surface of the transparency. The slot performs better because there is less entrainment of flight compartment air. Entrainment of flight compartment air lowers the heat transfer.
coefficient and lowers the air velocity across the conditioned air to the flight compartment. If needed, an auxiliary heater may be added to heat the air. Care must be exercised in designing the system to prevent overheating the transparency near the nozzles and maintaining sufficient defogging performance away from the nozzles.

The electrical system is similar to the electrical anti-ice system. It will consist of an electrically conductive film on the outboard surface of the inboard ply, a sensor to sense the control temperature and a controller to control the heat applied to the windshield and to protect the windshield from overheat. This system is most widely used on windshields that already have an electrical anti-ice system.

The following factors must be considered when choosing a defog system:

a. Area and configuration (shape) to be defogged or defrosted.

b. Availability of supply air for the defog system.

c. Accessibility to the edge of the windshield or canopy for the nozzles.

d. The size of the windshield with respect to the distance the air can be effective.

e. Capability of the transparency to accept a conductive film, bus bars and electrical connection.

f. Compatibility of transmission requirements and the transmissivity of the conductive film.
g. Differences in reliability between the hot air system and the electrical system.

h. Differences in windshield replacement cost for the same transparency using each system.

7-602 DEFOG ANALYSIS DISCUSSION

The two defogging system analyses that will be discussed are the hot air defogging system fed from a continuous slot and the electrical conductive film system.

The hot air system works best with the plenum and slot at the lower edge of the windshield or canopy. This allows the natural convective currents to assist the hot air flow upwards along the inner surface.

The principle of operation is to provide a jet of warm air across the surface at a certain temperature and heat transfer coefficient so as to raise the inner surface temperature of the transparency above the dew point.

For slot widths of 0.1 inch to 0.5 inch, the following equation is used to compute the heat transfer coefficient versus distance from the slots:

\[
h_x = 0.1016 K_s M^{-0.4} X^{-0.6} \left( \frac{W_s}{\mu_s L} \right)^{0.8}
\]

where:

\( h_x \) = the heat transfer coefficient at point \( X \)-BTU/HR-FT\(^2\)-OF.

\( K_s \) = thermal conductivity of the air at slot conditions - BTU/HR-FT-°F.

\[7.047\]
\[ T_x = T_s (1-n) + nT_c \]  
(7.23)

where:

\[ T_x \] = temperature of the air at point X - °F.

\[ T_s \] = temperature of the air at the slot - °F.

\[ T_c \] = temperature of cockpit air - °F.

\[ n \] = air jet decay factor (Figure III - 10, part IV, Reference 7.21).

For slot widths of 0.15, 0.25 and 0.50; Equations 7.24, 7.25 and 7.26 respectively, give relationships for \( n \), which are valid from 7 to 32 inches from the slot

\[ n = 0.82 - (0.58) (2.9)^{-0.01 X/M} \]  
(7.24)

\[ n = 0.77 - (0.75) (7.33)^{-0.01 X/M} \]  
(7.25)

\[ n = 0.68 - (0.88) (26.27)^{-0.01 X/M} \]  
(7.26)
By using the temperature and heat transfer coefficient at point \( X \), the heat balance can be made to compute the inner surface temperature of the transparency. The balance is as follows:

\[
h_x (T_x - T_{si}) = \frac{1}{\sum_{j=1}^{N} \frac{1}{K_j} + \frac{1}{h_o}} (T_{si} - T_o)
\] (7.27)

where:

- \( T_{si} \) = inner surface temperature of the transparency - °F (should be greater than flight compartment dew point temperature).
- \( j \) = individual plies of the transparency.
- \( N \) = number of plies.
- \( X_j \) = thickness of each individual ply - FEET.
- \( K_j \) = thermal conductivity of each individual ply - BTR/HR-FT\(^2\) - °F.
- \( h_o \) = external heat transfer coefficient - BTU/HR-FT\(^2\) - °F (see Equation 7.3).
- \( t_o \) = external recovery temperature - °F (see Equation 7.5)
The internal surface temperature is then compared to the dew point temperature. If the surface temperature is above the expected dew point, no fogging will occur. If the surface temperature is below the expected dew point, the supply air flow and/or temperature would need to be increased to provide adequate defogging.

The electrical defogging system is similar to the electrical anti-icing system described previously. The main difference is that the electrically conductive film is deposited on the outside surface of the inner ply of the transparency. The temperature of the conductive film must be raised high enough to raise the inner surface temperature above the expected dew point.

The heat balance for electrical defog to determine the temperature required on the film is:

\[
h_i (T_{si} - T_c) = \frac{K_i}{X_i} (T_f - T_{si})
\]  

(7.28)

where:

- \( h_i \) = internal heat transfer coefficients - BTU/HR-FT\(^2\) - °F.
- \( K_i \) = thermal conductivity of the inner ply - BTU/HR-FT - °F.
- \( X_i \) = thickness of inner ply - FEET.
- \( T_f \) = conductive film temperature - °F.

To determine the heat required, a heat balance on the entire windshield or canopy is used.

\[
\frac{Q}{A} = \frac{1}{\frac{1}{h_i} + \frac{X_i}{K_i}} (T_f - T_{si}) + \frac{1}{\sum_{j=1}^{N} \frac{X_j}{k_j} + \frac{1}{h_o}} (T_f - T_o)
\]  

(7.29)

7.050
Refer to the discussion on heat transfer quality control constants for the electrical anti-icing system. They can be applied for the defog system in the same manner as for the anti-icing system.

7-603 EXTERNAL DEFOGGING (DEMISTING)

A benefit of the electrical defogging system is that it can be sized to provide enough heat to the outer surface of the transparency to keep the outer surface from fogging after the aircraft has landed. The fog occurs when an aircraft makes a high speed descent and landing from a cold cruise condition to a hot humid condition on the ground. The design condition for demisting is to maintain the outer surface temperature above the ambient temperature. A transient heat transfer analysis is required to determine the adequacy of the heat applied.
7-700 INTRODUCTION

Lightning is a common but frequently misunderstood natural environment to which most aircraft can be exposed sometime during their operational life. The results of a lightning strike to an aircraft can be catastrophic, but need not be if the proper design precautions are employed.

7-701 LIGHTNING ENVIRONMENT

Lightning is the sudden discharge of large electrical charges which accumulate within portions of a cloud mass. The discharge may take place between dissimilar charge centers within the cloud mass or between the cloud and the earth. More is known about the latter (cloud-to-ground) type of lightning because ground measurements are more easily made, although intra-cloud lightning can be very important to aircraft operations.

All of the details of the charging and discharging process are not required to safely design an aircraft windshield system, but some elementary understanding of the discharge process will be beneficial. Excellent background information on the whole subject of aircraft lightning protection has been accumulated and presented by Fisher and Plumer in Reference 7.22. Reference 7.23, by Cianos and Pierce, presents probably the best available definitions of the lightning environment as it applies to engineering usage. The AFSC Design Handbook DHl-4, Reference 7.24, also contains useful background information.

Lightning does not strike an aircraft in flight as it does a tree, a building, or a person standing in an open field. The electrical capacitance of even the largest aircraft when in flight is too small to store or receive an electrical charge of sufficient
amplitude to be the object of a lightning strike. When a flying aircraft is struck by lightning, the aircraft is usually an incidental electrical conductor that just happens to be in the discharge path. The presence of the aircraft may only slightly alter the inevitable path, but in several instances the aircraft may have been instrumental in triggering a flash since the aircraft presents a perturbation to the ambient electromagnetic field.

Because of the incidental involvement, lightning strikes to a specific aircraft are not a common occurrence. However, when even small numbers of a given type of aircraft are in use, it is quite probable that the aircraft type will be involved in a lightning strike. Because the largest lightning discharges take place between the cloud cover and the earth, lightning involving aircraft is most frequent during takeoff and climb and during landing procedures when weather is poor and windshield visibility is most needed. Therefore, lightning should be an important consideration in the basic design of windshield/canopy systems for most types of aircraft.

7-701.1 Lightning Attachment to the Aircraft

Just prior to the highly visible portion of a lightning strike, there is a flash path preparation phase which involves local ionization of path segments called stepped leaders. When these probing leaders approach an aircraft, they induce localized ionization of the air on portions of the aircraft which provide a high electrical gradient. The local ionizing on the aircraft grows to streamers of ionization which reach out in an attempt to join the leader or leaders advancing between the major opposing charge centers of the cloud or cloud-earth combination. When leaders join to provide an ionized path between the charge centers, a large current flows. This is the initiation of the brilliant lightning event, which is termed the return stroke.
If the streamers from the aircraft bridge the main leaders from the charge centers, the aircraft becomes a part of the return strike path, and large current can flow through the aircraft. Many path combinations are possible. There may be simultaneous parallel paths through the aircraft and the surrounding air. Multiple paths through the aircraft are also a frequent occurrence. There must always be at least one entry and one exit point or area on the aircraft. Typical lightning attach points (high gradient areas) for a conventional aircraft configuration are the nose, wing tips, vertical stabilizer tip, horizontal stabilizer tip, and extended landing gear. Rotor tips are frequent attach points for helicopters.

The nose area of fighter, bomber and transport configurations is a high probability attach point. Protruding bubble canopies of fighter aircraft configurations are also possible attach points.

Most lightning flashes consist of more than one stroke. A lightning flash may last for several hundred milliseconds and contain as many as ten strokes. The initial stroke may pass a peak current as high as 200,000 amperes or more, which will reach a peak amplitude in approximately 2 microseconds. The initial stroke may be essentially complete in 3 milliseconds, but the flash path through the air remains sufficiently ionized after each successive stroke to assure a similar path for succeeding strokes of the total flash.

Because of the duration of the total flash path, ionization is sufficient to allow the aircraft to move a substantial distance during the total flash; a phenomena known as swept stroke occurs. This is important to windshield design because it is the swept stroke that is the most probable windshield encounter. The swept stroke is most easily visualized as an initial attachment to the nose of the aircraft with a second attach point somewhere in the tail area. The flash path is considered stationary in space. The forward motion of the aircraft causes the initial nose attach point to move over the aircraft (and windshield) as time passes.

7.055
If the nose area were of clean, unpainted, highly conductive metal, the flash path would slide rather smoothly along the metal surface, with subsequent restrikes of a multi-stroke flash passing pulses of high current through the conducting skin. In practice, however, the aircraft nose area is not a continuous highly conductive surface. A plastic radome may be a large part of the nose path, followed by a painted metal surface, followed by a glass or plastic windshield or canopy.

Starting from an initial attachment to a pitot tube on the nose radome or to a radome lightning diverter strip, the ionized lightning path will dance aft in a hop-skip-and jump motion. The initial stroke will tend to stay attached to its first point of contact, with the ionized path being blown over the down-stream surface until the "stretched" path finds a more conductive surface, where it reattaches to the surface. When the path is covered with an electrical insulator, such as a plastic radome or painted metal surface, the interval between reattaching points may extend to several inches or even feet, if the surface insulating qualities are high. At the time of each restrike, a new source of charge is tapped which provides the impetus for the lightning current to puncture localized surface insulation to any electrical conductor that lies beneath. It is this phenomenon which is important to windshield design. Figure 7.2 illustrates the swept stroke action.

When the aircraft configuration places the windshield or canopy in a high electrical gradient location, the transparent region may be a potential direct strike region, thus subjecting the transparency to the higher intensity lightning possibilities of the initial stroke, rather than to the usual reduced intensity, subsequent stroke environment of a swept stroke.
Figure 7.2. Swept Stroke Phenomenon.

7-701.2 Effects of Lightning

As a result of lightning attachment to an aircraft, three areas of concern are generated.

7-701.2.1 Physiological Effects

Lightning flash blindness can result from the brilliant discharge of lightning on or in the immediate vicinity of the windshield or canopy. Little or nothing can be done by the aircraft designer to eliminate the flash. A study of current literature showed that the flash resulting from a cloud-to-ground stroke can cause temporary blindness of over 3 minutes duration. This recovery time is almost entirely dependent upon the relative brightness of the flash over the background illumination.
Electric shock to the flight compartment crew can result if a lightning flash path is permitted to produce significant voltage drops in metal objects attached to or accessible by the crew. Currents of less than .06 Amps will cause shock with ventricle fibrillation at levels of . To date pilots have reported low level annoying shocks but nothing that could be termed hazardous. Direct lightning penetration through a canopy to a crew member is possible, but is normally prevented by other required design considerations. Electrostatic induction from a non-penetrating external flash to ungrounded electrical conductors within the cockpit must be prevented by proper insulation or grounding of the objects with which the crew may come in contact.

Magnetic field induction of electric current directly into the crew member's body by the external high current lightning flash is possible when the body is exposed to these fields. Such exposure could exist where the aircraft design surrounds the crew with large areas of nonmetallic material, such as a bubble canopy of plastic or glass. This phenomenon has not been definitively reported from flight experience but has been experienced in the laboratory.

7-701.2.2 Mechanical Effects

Either a direct or swept stroke lightning attachment to a windshield/canopy or edge area can produce strong, local pressures due to the rapidly expanding heated air. The magnetic forces of the high current can also distort metal. Fortunately, the direct mechanical effects of lightning are usually accommodated satisfactorily as a result of meeting other mechanical design requirements. Electrically induced mechanical considerations are less obvious and must be identified for a specific design. An example would be the mechanical stress induced in anti-icing electrical conductors when these conductors are transferring lightning induced electric currents.
7-701.2.3 Electrical Effects

Direct penetration of a windshield or canopy can cause high voltage and current to appear on electrical conductors associated with anti-icing or defogging circuits. The high amplitude of these currents and voltages can cause serious permanent damage to many systems beyond those which are directly connected to the windshield/canopy.

Non-penetration lightning can also induce potentially destructive or disruptive voltages or currents in aircraft circuits. The most directly affected are the anti-icing and defogging circuitry, although other systems which have wiring in the area of the windshield can be adversely affected.

When electro-explosives are used for canopy ejection, special precautions are in order to assure that a lightning discharge to the canopy region does not cause inadvertent firing of the explosive train.

Lightning need not attach directly to the windshield/canopy for unwanted electrical signals to be induced into sensitive circuits. Flash attachments to other regions of the aircraft or nearby flashes that do not even involve the aircraft can induce signals in aircraft conductors. These Lightning Electromagnetic Pulse (LEMP) signals may have different characteristics than those that result from direct attachment to the windshield/canopy area, and their amplitude will usually be lower - at least in the lower frequency portion of the spectrum.

From the standpoint of windshield/canopy system design, LEMP signals will be far more frequent than electrical transients caused by direct attachment of initial attach point lightning or swept stroke lightning. However, the direct attachment signals will be of
such higher amplitude as to be the predominant threat. Therefore, LEMP effects on windshield systems can usually be ignored if the system is adequately protected from the direct effects. The main exception to this general statement could be a system that has high immunity to low frequency signals and a high level of susceptibility to higher frequencies.

7-701.3 Designing for Lightning Flash Attachment

The design of an aircraft for which lightning will be a design consideration will usually undergo a lightning attach point study. This will involve the entire aircraft and will identify the various areas or zones which should have a high probability of receiving a direct lightning attachment. See Reference 7.25, Section 2.4, for a more detailed description of attachment zones.

The attach point study may be conducted by high voltage testing a model of the proposed aircraft, or by analysis of the configuration with respect to past lightning attach experience with aircraft having similar overall geometry. The windshield designer would normally consult other specialists for this information.

When the airframe configuration is of a conventional design, it is reasonable to expect that a canopy might be a direct attach point (or area prone to lightning attachment, and that windshields of transport type configurations and canopies will both be candidates for swept stroke lightning. For the latter, the point of initial attachment will usually be more forward on the nose, or radome, if it is so equipped. Rotary wing aircraft may present a somewhat different attach point geometry.

Unless lightning is a very high priority environment for the aircraft design, the windshield/canopy designer will have little choice but to accept the lightning attach point category that results
from the geometry of the total aircraft. When direct lightning attachment or swept strokes appear inevitable, their effect on the windshield/canopy system can still be minimized if other design flexibility exists. Direct lightning attachment to a point on the aircraft is preceded by local corona streamering activity which is instigated by the approaching lightning stepped-leader. Unless the transparency open area is very large and there is an unusually high gradient electrical conductor near the inner central surface of the transparency, the stepped-leader will not attach to and puncture the transparency. If the transparency is in the general region of the impending direct lightning attachment, the attachment probably will be to some external nearby metal structure. Even if local streamers are being emitted from some conductor under the transparency, the dielectric strength of the transparency can be made sufficient to force a discharge to flash across the outer surface of the transparency to the metal structure of the aircraft.

While the above scenario implies a low probability of direct lightning puncture of a windshield or canopy, puncture has happened. The possibility must be thoroughly studied. Puncture and attachment to a crewmember could be fatal. Where a study of the electro-mechanical geometry and comparison with previous high voltage testing on similar configurations still leads to doubts, high voltage testing of the proposed new configurations is strongly recommended.

7-701.4 Lightning Puncture Prevention

Physical puncture of the windshield or canopy can be prevented. If the lightning attach study and candidate window design indicate potential puncture problems, positive steps must be taken to prevent the puncture possibility. One of the most common and successful approaches is to combine electric field control with the dielectric strength of the transparency material.
Electric field control involves reducing the electrical voltage gradient through the transparency when it is subjected to the external lightning environment. As a first crude approximation, grounded electrical conductors in the cockpit, including the crew, should not come closer to the inside contour or surface of the transparency than the locus of points traced by a 120 degree cone with its sides touching the metal peripheral supports of the transparency. This locus is pictured in Figure 7.3. This "120-degree guide" is adapted from military specification MIL-B-5087, Figures 7 and 8. This is similar to the "cone of protection" discussed in many documents on lightning protection. The cone of protection concept is not precise and should be used only as a preliminary guide.

![Diagram showing the 120° Exclusion Zone]

Locus of points of "120°" angle forms surface within which conductors should not be located.

Grounded conductors within shaded volume may lead to puncture of low dielectric strength windshield material.

Figure 7.3. "120" Exclusion Zone.

The cone angle is not a go, no-go number. Much larger angles, up to 180 degrees may be acceptable where the transparency has sufficient dielectric strength to prevent puncture. An efficient
cockpit canopy layout may not permit the spacing required to meet the "120-degree guide," while many transport crew compartment configurations present no significant problems.

Possible high electrical gradient problems caused by items within the cockpit can be circumvented by coating the inner surface of the transparency with a transparent electrical conductor. The coating must be electrically grounded. The action of the coating is to form an electrostatic shield for the items within the cockpit and provide a uniform potential surface to the outside electric field. The average electrical stress on the transparency is increased due to the close proximity of the coating to the transparency, but high level non-uniform stress are prevented. When this stress-averaging technique is coupled with good insulating qualities in the transparency material, the likelihood of lightning puncture is greatly reduced. It will be noticed that the use of a grounded electrical coating clearly violates the 120 degree cone concept. This is an acceptable "violation" because for this case, the transparency provides the necessary insulation.

Lightning testing of windshield candidate configurations (Reference 7.25) has shown that the insulating qualities of glass and acrylic outer ply materials of 0.085 inch thickness or greater are sufficient to prevent puncture when sharp electrical conductors which cause higher electrical gradients are not in immediate contact with the inner surface of the transparency.

Tests and flight experience have also shown that lightning and static electric charge can puncture thinner materials with integral interior electrical coatings or conductors. This has been a particular problem where thin electrical heating wires have been imbedded within the plies of a thin transparency. In this case, the local electrical gradient around the thin wire is very high, which results in a dielectric stress sufficient for puncture.
Direct, initial stroke lightning attachment to a windshield or canopy will occur only very rarely. If the flash attaches in the windshield/canopy area it will usually go to the metal surrounding the transparency.

Swept stroke restrikes can occur during the sweep period across the surface of the transparency. If there are no conductive coatings or significant electrical conductors under the outer surface of the transparency, the flash path will probably not even touch the surface of the transparency. When grounded conductive coatings are present, the flash path during the high current restrike can hug the transparency surface and, depending on the material, the arc may leave an etched imprint on the surface of the transparency.

The tests of Reference 7.26 employed a laboratory swept stroke with controlled restrike on large 48-inch by 62-inch specimens of candidate outer windshield ply materials. Most of the test specimens had an electrically conductive coating on the specimen side opposite the flash path. Figures 7.4 and 7.5 obtained from Reference 7.26 show the simulated swept stroke lightning flash across the surface. Note in Figure 7.4 that the arc did not touch the surface of the specimen until the time of restrike, when the brilliant hook-shaped flash touched down on the specimen surface. The flash path was then along the surface to the metal perimeter of the specimen. Figure 7.5 shows the resulting etch marks which remained on the surface of the soda lime glass specimen. Similar markings were noted for Herculite II glass and Chemcor glass. No markings could be detected on an acrylic specimen.

Lightning flash etching of the type shown is not believed to be frequent occurrence. The etching is not deep and would most probably appear near the upper or aft edge of a windshield.
Figure 7.4. Re-strike Attachment to 48 Inch by 62 Inch Soda Lime Glass Specimen.

Figure 7.5. Re-strike Markings on Soda Lime Glass Specimen of Figure 7.4.
The same series of lightning tests reported in Reference 7.26 showed that the presence of a conductive anti-static coating on the outer surface of the glass specimens did not increase the likelihood of swept stroke lightning attaching to the windshield. When attachment did occur, the anti-static coating was permanently vaporized in the vicinity of the flash path. A dirty brown permanent stain remained which might be objectionable from the standpoint of vision.

Lightning strike testing of polycarbonate canopies for high performance military aircraft (Reference 7.27) has shown that outer surface flashover rather than puncture is to be expected when the material thickness is sufficient to meet the structural requirements.

If the detail design and testing of a canopy installation shows a high enough probability and tendency for puncture, protective techniques are available. These are similar to those used to protect radomes, and consist of a flash diverter strip. The strips are usually located on the outer surface and conduct or divert the flash safely to nearby metal structure. Reference 7.25, Chapter 7.3 has a good discussion of this technique together with extensive references.

7-701.5 Electric Shock Control During Lightning Strike

The high intensity electric and magnetic fields and very high discharge currents associated with a lightning strike present electric shock possibilities unless special precautions are taken. Windshields and canopies which do not use conductive coatings for electrical anti-icing, RCS control, etc., will not provide electrostatic shielding of the cockpit interior. Streamering may occur on interior electrically conductive items, including the crew. Annoying electric shocks have been received by crew members, but the streamering current, when the canopy is not punctured, is not high enough or of long enough duration to be a serious threat (Reference 7.27).
Very strong magnetic fields surround the lightning flash. When the flash is on or in close proximity to the windshield or canopy, these fields will penetrate to the cockpit interior. The thin conductive coatings used for anti-icing or RCS control are essentially ineffective magnetic field shields. The penetrating field will induce current in closed electrical paths or voltage in open circuit electrical paths. Potentially distracting or harmful voltages can also be developed when the induced current flows in resistive paths. The human body is also a conductor in which the fields can induce current.

Crew exposure to the lightning magnetic fields is probably quite low for transport type crew compartments, due to the distance between the crew and the windshield. Canopy, crew enclosures, that place the body high in the canopy and close to the surface would significantly increase the exposure. Specific problems from magnetic field exposure have not been routinely reported. However, when the cockpit/crew geometry for evolving designs provides magnetic field exposure, the possibility of directly induced electric shock should not be overlooked. Corrective action would be to lower the crew members more into the shielding environment of a metal fuselage and away from the exposure within the open volume of the canopy as well as the previously mentioned diverter strips.

Hinged canopy configurations can also present possible shock hazards due to inadequate electrical bonding between the metal frame of the canopy and the metal fuselage. Weather and pressure sealing requirements frequently may result in long non-conducting gaps between the canopy frame and the fuselage. Latch points are the only areas of contact, and these do not always provide a low impedance path. When lightning flashes over the canopy, it will generally attach to the metal framework of the canopy. The path from there to the fuselage may present sufficient impedance to allow the current to produce a high voltage between the canopy metal structure and the grounded metal items within the cockpit which may be touched by the crew.
Corrective measures are most economically taken during the design phase and consist of providing good electrical conductivity through the latch mechanisms and more hard conductive contact points, as required. An additional worthwhile approach is to keep the physical spacing between external conductive metal parts of the canopy and the metal fuselage as close together as practical. This design approach tends to direct the lightning flash across the canopy directly to the fuselage and to significantly reduce the current flowing in the canopy structure. If the cockpit layout places canopy metal parts, especially those around the canopy/fuselage interface, where the crew might reasonably touch them during flight, lightning electrical shock tests of the proposed configurations are recommended. This advice is especially appropriate if the fuselage around the cockpit is made with composite materials since these materials are poor electrical conductors.

7-701.6 Lightning Caused Mechanical and Acoustic Effects

The quickly changing, very high peak currents of a lightning flash can induce high mechanical stress in electrical conductors. Measures to prevent this mechanical stress from damaging critical mechanical parts include additional mechanical rigidity, alternate current paths to reduce the current density in a given part, and the choice of thicker conductors. Reference 7.25, Figures 4.4 and 7.6 show examples of the magnetic pinch effect.

A high velocity, high pressure shock wave will accompany a flash in the immediate vicinity of the flash path. This pressure can produce mechanical loads that must be accounted for in the design of transparency systems. No recorded history of overpressure caused problems to the transparency is available, although lightning overpressure has been known to cause inadvertent inflight activation of latch mechanisms. Overpressure investigations must include the peripherals of transparency design in addition to the transparent areas. M. A. Uman (Reference 7.22) presents some of the most readily available information on the close-in overpressure characteristics of lightning.
Some induced effects were discussed previously under Electrical Shock. This subsection addresses transients induced in electrical circuitry.

Windshields and canopies which contain electrically heated wires or coatings and electrical temperature sensors are potentially vulnerable to lightning induced electrical transients. A strike which penetrates the transparency to the electrical conductor is the most severe, but is a very rare occurrence. The more probable, yet infrequent, occurrence is caused by magnetic flux coupling from a swept stroke to the conductor. Puncture need not take place to induce destructive currents in the thin conductive films or damage to the connected electrical system. A third case of lightning induced electrical transients is caused by electro-magnetic radiation from a nearby lightning flash. This very frequent case does not involve the aircraft in the lightning flash path and is referred to as LEMP (Lightning Electromagnetic Pulse). When one designs for the higher induced transients of a swept stroke attachment, the much lower amplitudes of LEMP are usually easily accounted for.

Electrical transient suppression measures are not a final cure for the existence of lightning induced electrical transients in the electrical circuitry. In fact, some of the most effective transient suppression techniques can increase the damage to the heated film by reducing the transient electrical impedance between the film and the airframe structure. The most effective area for corrective action lies in very careful design and manufacturing control of the conductive film/bus bar. This control must also strongly address the prevention of moisture ingress from the periphery of the transparency to the bus bar/conductive film interface. External control of the flash path may help to reduce the amplitude of the induced transients.
The electrical transient that is induced in the conductive films in the transparency will also be delivered to the anti-icing or defogging electrical controller. Control circuitry of modern design usually employs solid state devices in both the temperature sensing and power control circuits. These solid state devices can be vulnerable to permanent damage from the lightning induced transients. Technique and hardware have been developed which can protect the solid state devices. Because the methods used for lightning transient control can be applicable to P-static and LEMP transient control they will all be covered in Section 7-702.

Lightning flash activity on and around a canopy may also induce unwanted current or voltage in electrically activated electro-explosives used for canopy emergency ejection or in aberrations in fly-by-wire digital flight control systems. Transient signal entry to the electro-explosive system wiring can be via magnetic flux leakage directly through the transparency or via the structural gaps between the canopy frame and the fuselage. Electrically fired ejection systems should not be used unless there are no other options.

The same design and construction techniques which can reduce the electric shock hazards around the canopy are also applicable to reducing the hazards to electro-explosives. In addition, many of the design techniques covered in MIL-STD-1512 (Reference 7.28) should be applied.

7-702 P-STATIC

This section has been titled P-static since the phenomenon that creates the static is best known by this name. Actually, P-static (from precipitation static) is one of the results of a very common natural environmental phenomenon called triboelectric charging.
The Charging Process

When an aircraft in flight impacts with quantities of dust particles or dry ice crystals, a process known as triboelectric charging occurs. The impact of the particles on the frontal areas of the aircraft creates an electrical charging mechanism which can cause the aircraft to reach a very high potential. The in-flight process occurs to a lesser degree when the aircraft is in motion on the ground. The same charging mechanism also tends to charge automobiles, trucks, trains, etc. However, vehicles in contact with the ground quickly transfer their charge to the earth through the finite electrical resistance of the tires or wheels, and no significant charge is retained by the vehicle.

When the aircraft is in flight there is no relatively low resistance discharge path. The result is a continuing build-up of the charge, assuming the particle impact process continues, until the aircraft outer surface reaches a self-discharge potential threshold. This may be tens or hundreds of thousands of volts, and depends on the external configuration of the aircraft. The charge that is created on the frontal surfaces of the aircraft is free to move over the metal surface of the aircraft. The potential gradient will be the greatest at sharp contours, such as found on control surface outer trailing edges. The self-discharge threshold is attained when the air about the high gradient locations ionizes, thus becoming an electrical conductor. The discharge can produce interference in the radio receivers on-board the aircraft. Therefore, the term P-static came into use in the early days of aircraft radio usage.

Corona discharging from the aircraft reaches equilibrium with the charging rate to prevent an ever increasing voltage from appearing on the aircraft.

Unlike the metal frontal areas of the aircraft, windshields and canopies are usually good electrical insulators. The charge on the
The insulating outer surface of the windshield/canopy is bound to the surface where the charge will build an intensity until it can no longer be supported by the electrical insulating qualities of the transparency outer surface and the air.

The amount of the charge and the rate at which it is accumulated are related to several factors; the more significant of which are the outer surface resistance of the transparency, the size of the frontal area, the impact angle of the oncoming ice crystals and dust particles, and the velocity of the aircraft. The interrelationship of these variables is shown below.

Christie, Reference 7.29, page 725 has summarized the interrelationships of the latter three in Figure 7.6 to show trends. The velocity function decreases around Mach 2.2 due to instant melting of the ice crystals on impact; i.e., the crystal is captured on impact, thus negating the triboelectric action.

![Figure 7.6. Triboelectric Charging Factors.](image)
The Consequences of Windshield/Canopy Charging

When the transparency has received the highest charge it can hold, either or both of two things will happen: surface flashover or puncture.

When the high voltage charge can no longer be supported by the surface-air interface insulating characteristics surface flashover may take place. The surface air will ionize and a flash will take place from the charged surface to the metal airframe which surround the transparency. The flash pattern resembles the branching outline of a tree. The discharge can be a brilliant pattern that covers most of the transparency, or it may be an almost unnoticeable, dim discharge on only a portion of the surface. Many factors influence the flash size and intensity.

During nighttime flight, a large flash can produce temporary blindness. During the daytime, the same flash may be only a slight distraction to the flight crew.

The surface flash may produce a momentary radiated electrical transient in addition to the visual flash. This transient would be heard as a crackle or click in the audio output of radio receivers or a momentary false reading in the radio navigation instruments. In an extreme case of rapid recharging and discharging, the radiated electrical transients might become a significant radio noise problem. Also, other electrical/electronic systems with wiring running adjacent to the windshield/canopy may receive unacceptable interfering signals.

As long as the transparency does not contain electrical conductors and there are no inadvertent conductors inside the cockpit that come within a few inches of the surface there is little likelihood that the transparency would be punctured by the discharge.
If a crew member were to come in contact with, or very near to a charged windshield/canopy that had no electrical conductors within it, the crew member might receive a stinging electric shock. This shock possibility exists even for low charge levels on the outer surface—long before the discharge threshold is reached. A shock hazard may also exist for ground crews, even for electrically heated transparencies.

When the windshield/canopy contains electrical conductors, more serious problems can result from triboelectric charging of the outer surface. The presence of a conducting surface on the inner side of an insulating transparency and the deposited electric charge on the outside form an electrical capacitor and causes more charge to be stored. The ultimate discharge may be more violent. The charge may dissipate in an outer surface flash, or the transparency may be punctured. A puncture can destroy the visual qualities of some material, and the physical strength of some windshield/canopy designs may be seriously weakened.

Either surface flash-over or surface puncture of an electrically heated windshield/canopy can introduce large electrical transients into the windshield/canopy electrical control circuitry and other circuits which may share the same wire routing as the control circuitry. These transients are potential destroyers of solid-state circuit components.

All aircraft which employ transparent windshields or canopies will be subject to frequent triboelectric charging to some low degree. Many of these aircraft may receive potentially troublesome charges on numerous flights.

Problems associated with triboelectric charging can be largely controlled and accommodated when the proper design is employed. Effective protective measures may require that special electrical
Transient protection hardware be placed right at the electrical terminals of the anti-icing or defogging elements of the windshield or canopy. Space requirements and maintenance access for these devices must be planned for during the flight compartment layout phase. Thorough coordination of the lightning and NEMP (if applicable) protection plans with the P-static protection plan will simplify and optimize the hardware that must be located in these space-scarse windshield/canopy areas.

R. L. Tanner and J. E. Nanevicz have written several excellent reports on the subject of precipitation charging of aircraft. Their works are recommended reading for those who desire further information on the subject (see Reference 7.30).

7-702.3 Charge Prevention or Reduction

P-static problems for the windshield system designer are eliminated or largely disappear if the charge of each impact can be allowed to migrate about the surface of the windshield to the conductive structure of the airframe. Such a condition is different from the usual situation where the charge is bound to the windshield at the point where the impacting ice crystal particle, with the equal and opposite charge to that on the windshield, bounces off of the surface, never to return. Charge mobility is achieved by making the impact surface conductive, as is the case for the metal portions of the aircraft that also receive the ice crystal bombardment. The degree of conductivity need not be high, as is the case for metal. Surface resistivities of several megohms per square can provide an adequate drain path for the charging rates normally encountered on even the largest area windshield.

If one were to select materials for the outer surface of a windshield/canopy that had poor electrical insulating qualities, it might be possible to substantially reduce or eliminate the bound
charge problems that can lead to surface flashover or puncture. Unfortunately, current aircraft glazing materials have too much surface resistivity and too little volume conductivity to prevent charge buildup, for most configurations. Charge retention is usually high for most plastic materials. Glass windshields do not retain the charge as do plastics, but they can accumulate a sufficiently large static charge to cause problems. Reference 7.26, discusses the investigations of the static retention characteristics of four common outer surface glazing materials. Their abilities to hold a static electric charge are ranked in the following order.

1. Acrylic sheet (most retention)
2. Chemcor glass
3. Herculite II glass
4. Soda lime glass (least retention)

The charging plots of Figure 7.6 show that the charging current will increase directly with the impacted area. However, the maximum charge which can be retained is not a direct function of the area. Rather, it is related to the distance from the charged area to the grounded metal periphery of the windshield, and to the dielectric properties of the windshield outer surface material and the air in immediate contact with the outer surface. A large, square windshield may experience more static electric discharge problems than another windshield of the same total area, but widely differing length-to-width ratios, with all other characteristics remaining the same.

Hand-applied transparent, electrically conductive coatings applied over the outer surface of conventional glazing materials provide temporary protection as they successfully allow the surface charge to dissipate rather than build up. Unfortunately, these coatings are not very practicable for high speed aircraft. For low speed which includes all general aviation and utility, this method is appropriate if the aircraft is to be flown in a P-static environment.
Charge prevention by the application of a permanent conductive coating to the outer surface cannot be satisfactorily applied to all windshield surfaces. At this time, there is no durable conductive coating that can be applied to plastic materials which is abrasion resistant. Stannous oxide, which is a much more durable coating than the coatings currently available for plastics, is an example of one conductive coating that is available for glass.

The service life of stannous oxide anti-static coatings is not well documented. Attempts to determine the life expectancy led to the conclusion that once placed in service, no surface resistance measurements are ever made. This might be interpreted to indicate that these coatings give sufficient performance.

Because a stannous oxide coating must be applied while the glass surface is quite hot, the deposition process is not applicable after laminating a multi-ply transparency which uses low melting temperature interlayers. Therefore, post laminating optical corrections by localized surface polishing or grinding are ruled out when a stannous oxide coating is used, since the coating would be removed by the operation.

The surface resistivity of any anti-static coatings may vary widely, because the coating application process is not easily controlled. The problems are in the inability to control the uniformity of the coating thickness over the transparencies' surface. Low resistivity coatings which provide low light transmission are more easily attained than high resistivity coatings.

7-702.4 Design Recommendations

Anti-static coatings applied to the outer surface of windshields should be specified as follows:
a. Maximum resistivity should be a value of approximately two megohms per square and a minimum resistance controlled by the maximum light transmission degradation that can be tolerated. The light transmission degradation to be allocated to anti-static coating must be determined from a detailed trade study of the entire windshield design (2-3% transmission loss is realistic). Voids in coverage should not be permitted at or near the periphery, although voids of six to eight inches in diameter in the central region of the windshield are allowable.

b. No conductive paths should exist to the back or interior side of the transparency material to which the anti-static coating is applied.

c. Electrically isolate the static drain bus and anti-icing power feed bus to prevent electrical feedthrough around the edge of the face ply glass. To this end, the dc resistance between the static drain bus and any segment of the anti-icing bus should have a minimum resistance of 75 megohms.

The design requirements must be verified by tests conducted prior to laminating the outer ply to the remainder of the windshield to allow corrective action if necessary. The tests should be conducted at an applied potential of at least 500 volts.

Anti-static coatings should be multi-point grounded to the airframe structure surrounding the windshield. Continuous peripheral contact is desirable. Figure 7.7 shows a method of grounding. The method to be used on a specific aircraft must be tailored to the requirements and limitations of that aircraft.
When anti-static coatings are unavailable or undesirable, there is another possible alternative. This involves a reduction in the open area which is subjected to the ice crystal or dust particle impact.

A large area can be artificially subdivided to present the static charge characteristics of a small open area. This is accomplished by placing narrow (1/16 to 1/8 inch) bands of conductive material on the outer surface to break up the open area (Figure 7.8). These strips should be electrically bonded to the aircraft structure to ensure electrical continuity. If foil is used, the thermal expansion properties of the foil, adhesive and glazing material must be sufficiently compatible to assure retention of the foil. If the foil is allowed to crack, the resulting intermittent electrical contact and discharge sparking can lead to the foil becoming an efficient radiator of electromagnetic interference to radio communication or navigation systems.
Figure 7.8 depicts two conductor patterns which would be helpful in reducing the charge amplitude on large windshields. The optical distraction caused by the conductive strips and the pattern generated by the strip placement must be optically acceptable. This has been accomplished in some instance by using fine wires rather than foil although the foil is preferred both electrically and physically.

Figure 7.8. Methods for Reducing Total Electrostatic Charge.
When glazing material restrictions or windshield system design constraints prohibit use of the foregoing charge accumulation prevention techniques, the charging must be accommodated by other design features. When surface charging does take place, it must be forced to discharge as an external surface flashover rather than as a surface puncture while being confined to the external surface rather than being allowed to flash around the edge of an outer surface ply to a conductive coating on an inner surface.

The outer surface edge grounding method depicted in Figure 7.7 is recommended even when no anti-static coating is practicable. A peripheral external bus applied to the surface of the outside material (referred to in Figure 7.7) as the "anti-static bus" will serve as the edge terminal for surface flashes. This bus must be peripherally connected to the metal of the airframe by the static drain strap. This approach helps to assure that the surface flash will not wrap around the edge of the outer ply and deliver the flash energy to an anti-icing coating. Removal of all conductive coating from the edges of the outer ply will eliminate potential parallel paths for the surface flash one path to the external structure and one path to the anti-icing coating. The anti-static bus feature is also helpful in preventing flash wrap around when the windshield/canopy encounters a lightning flash.

Outer surface puncture can be prevented by the same design features which prevent puncture by lightning. The key feature is to select material of adequate thickness. Extensive static electric charge testing (References 7.23 and 7.6) has identified the materials and thicknesses summarized in Table 7.6 as being puncture resistant.
TABLE 7.6. TRANSPARENT MATERIALS PUNCTURE RESISTANT TO STATIC DISCHARGE.

<table>
<thead>
<tr>
<th>MATERIAL</th>
<th>SIZE - INCHES</th>
<th>THICKNESS INCHES</th>
</tr>
</thead>
<tbody>
<tr>
<td>Chemcor glass</td>
<td>37 x 53</td>
<td>0.105</td>
</tr>
<tr>
<td>Chemcor glass</td>
<td>37 x 53</td>
<td>0.0085</td>
</tr>
<tr>
<td>Herculite II glass</td>
<td>48 x 62</td>
<td>0.110</td>
</tr>
<tr>
<td>Soda Lime glass</td>
<td>48 x 62</td>
<td>0.187</td>
</tr>
<tr>
<td>Acrylic Sheet</td>
<td>48 x 62</td>
<td>0.125</td>
</tr>
<tr>
<td>Acrylic Sheet</td>
<td>34 x 46</td>
<td>0.080</td>
</tr>
</tbody>
</table>

All the specimens of Table 7.6 had a grounded conductive coating on the side opposite the static charged side. The conductive coating covered all but a small peripheral border.

Static electric discharge punctures have occurred in production aircraft. The frequency of puncture is low, but the cost of replacing the windshield is evidently high enough that puncture is considered economically unacceptable. Puncture of a thin, highly tempered glass outer ply could result in an opaque windshield due to instantaneous fracture propagation over the whole surface, with obvious, severe flight safety impacts.

Figure 7.9 is a photograph of severe surface discharging created in the laboratory on a 34-inch by 46-inch electrically anti-iced windshield specimen using an outer face ply of 0.080 acrylic plastic.

Since puncture is not commonly encountered, a test specimen was turned around to deliberately expose a non-typical surface to severe charging.
Figure 7.9. Severe Surface Discharge on 34 x 46 inch Acrylic Specimen.

Figure 7.10. Discharge Pattern During Puncture of 0.005 Inch Thick Mylar Sheet.
Figure 7.10 shows the discharge pattern during puncture of a 0.005 inch thick mylar sheet separated from a transparent conductive coating by an additional interlayer of 0.025 polyvinyl butyral (PVB). The bright spot in the center of the surface discharge flash fingers is the area of puncture.

All of the information indicates that the thicker the material the more improbable is the chance of electrical puncture of the outer surface. There are practical limits to the thickness; however, the principal limitations are weight, cost and thermal conductivity. Structural considerations, such as bird strike protection may also be an important factor for some designs.

7-702.6 Effects of Subsurface Coatings on Charge Retention

Most windshield or canopy configurations which do not contain electrical conductors, for such purposes as anti-icing, defogging or radar cross section (RCS), control will not be troubled with static electric puncture. Conversely, the presence of a subsurface conductive coating in close proximity to the outer surface charge tends to allow the accumulations of a larger bound charge on the outer surface and to increase the dielectric stress and threat of puncture.

Some anti-iced windshields only heat a small portion of the total frontal area of the windshield. This is usually done to achieve a uniformly heated electrical geometry and to conserve electric power. The remaining unheated area is occasionally coated with the same density conductive coating to retain a uniform light transmission. The heated area and the unheated area are electrically isolated by a thin line (deletion line), approximately 1/32 inch wide, where no conductor is present.

When unheated conductive areas are used and effective anti-static coatings are not used, the unheated conductive coated areas must be
grounded to the aircraft structure. If they are not grounded, large voltages will be developed on the coating due to the external triboelectric charging. Internal arcing may easily occur which might ultimately destroy the usefulness of the windshield or canopy. A grounding bus around the periphery of the unheated area, excluding the deletion line, is strongly recommended. This bus should be constructed similar to the current feed bus used on the heated portion of the window. The grounding bus should be connected to aircraft structure with at least one short lead. The grounding lead(s) should not be bundled with other wiring, especially temperature sensing wires, as the grounding lead will carry high current electrical transients during periods of external surface flashover.

An alternate approach to maintaining optical uniformity would be to substitute a non-conductive material having the same optical characteristics as the conductive anti-icing coating in the non-anti-iced area. A non-conductive coating would not require grounding, it would not tend to hold the outside charge as strongly, and the overall charge stress on the windshield would be reduced. This is not an acceptable alternative if an RCS coating is required, or if electrostatic shock protection of the cockpit occupants is desired. NOTE: A deletion line in highly conductive coatings has proven to be very distracting to the pilot.

7-702.7 Electric Shock

A windshield that has neither an external anti-static coating nor the majority of an internal surface coated with a grounded transparent electrical coating can be an electric shock hazard to the flight crew. In this case, a static charge on the outside surface will create an opposite charge on a part of the body that comes in contact with or is very close to the inner surface. A shock will result that will be more of an annoyance than a hazard, unless it causes a distraction during critical flight phase.
Certain glazing materials, notably plastics and some tempered glass, will retain a static electric charge for minutes or hours after the aircraft has landed. The retained charge level may be more extreme if the inner surface has a conductive coating. This charge can present a shock hazard to the ground crew when the outer canopy surface is touched. The shock itself is probably not dangerous to a healthy person, but the resulting muscular reactions may cause hazards such as a fall from a workstand. It is necessary, therefore, for ground crews to perform canopy discharge procedures before assisting the crew or servicing the aircraft, especially on aircraft with plastic transparencies.

Another form of electric shock has been experienced by ground crews working on the outer surface of alternating current electrically heated windshields. This is particularly applicable to windshields with a high resistance external anti-static coating. When anti-icing power is on, the heated coating and the anti-static coating form the two electrodes of a capacitor. The electrical reactance of this capacitor at the electric power frequency causes a reactive current to flow in the anti-static coating. When the surface is touched by someone who is also touching the fuselage some of the current is diverted through the body and is sensed as a tingle or electric shock. The corrective action is to turn off the anti-icing power to the windshield, when this is permissible. Otherwise, insulating gloves might be used. Figure 7.11 shows the equivalent electrical circuit of this shock path.
A. SCHEMATIC OF CAPACITOR FORMED BY OUTER PLY AND CONDUCTIVE FILM.

B. SIMPLIFIED SCHEMATIC

Figure 7.11. Schematic Diagram of Potential Electric Shock Circuit.
7-702.8 **Induced Electrical Transients Due to Static Electric Discharges**

Electrical anti-icing or defogging systems are subject to static electric discharge-induced electrical transients. The windshield electrical geometry and the triboelectric static charge create a charged capacitor with the anti-icing coating/defogging conductive coating or temperature sensors as one electrode; the glass or plastic as the dielectric; and the ionized surface air during the flash as the second electrode. When surface flashover takes place, the static status changes to a dynamic condition and a large current pulse flows in the conductive coating. This transient current pulse will also flow in the circuits connected to the electrical coating. Any resistance or impedance in the circuit will result in transient voltages. The transient current or voltage can disrupt or burn out connected electronic equipment. Windshield heater control and temperature sensing electronics are prime targets for this trouble.

The electrical characteristics of the transient will vary with each surface discharge, the stored energy in the discharge, and the windshield/canopy material. Figures 7.12 and 7.13 are waveform examples of the induced current in a transparent conductive coating on a 34 inch by 46 inch by 0.060 inch thick polyester face ply test specimen. The peak current is approximately 1000 amperes occurring in about one microsecond. This current flows in the conductive coating as well as in the external circuit, which was a shorted loop of wire about 8 feet in length.

![Waveform Example](image-url)
Another series of tests (Reference 7.23), with 0.08 C thick acrylic face ply, produced induced current of approximately 100 amperes peak in about one microsecond. Over 150 large flashovers failed to show any signs of degradation in the transparent conductive coating or in the bus/coating interface. These tests indicate that static electric discharge induced currents will probably not be destructive to a well designed and manufactured anti-icing electrical coating. However, when thinner materials are used and long life is important, these conclusions should be re-evaluated for the specific configuration.

7-089 Electric Transient Control Techniques

The most thorough transient control technique is one that prevents a surface discharge in the first place. The anti-static outer surface treatment, where achievable, eliminates the static electric discharge transients in the conductive coating as well as in the connected electrical circuits. Techniques which suppress or
control the transient after it is generated by the surface discharge cannot prevent the transient from appearing in the conductive coating.

7-702.9.1 Candidate Suppression Approaches

When electrical transients, due to lightning, static electric discharges or EMP may all be present, electrical supression methods are the only effective approach since an anti-static coating is ineffective for lightning and EMP.

The details of the transient suppression approach are affected by the rise time and amplitude of the transient source. Nuclear EMP may present the steepest transient wavefront due to the nature of the signal. However, the large amplitude component of the NEMP signal will be due to the fuselage resonance currents. For large aircraft this can be of a low enough frequency, approximately 2 megahertz, transient current rise times approximating those of swept stroke lightning and static discharge transients. For small aircraft the NEMP transient rise time will be less than for larger aircraft.

The transient wave shapes and even the amplitudes of the various transient sources for a given airframe may be effectively accommodated by one transient control approach. That one approach may be reached by two basic methods or a combination of the two methods.

The first method controls the transient at the electrical terminals of the windshield. Any filters, transient clamps, etc., to be used are grouped right at the windshield. This method prevents the transient from being conducted on any part of the aircraft wiring and prevents cross coupling contamination of other systems. The major disadvantage of this path is the need for significant space for hardware at the windshield, where space is usually at a premium.
The second method uses transient containment within shielded wiring from the windshield terminals to a remote location, perhaps the temperature controller, where the necessary transient control hardware is installed. Much of the valuable windshield space is saved, but the interconnecting wires must be highly shielded and, in many instances, must be capable of withstanding high voltages.

Both methods may be combined to gain some of the advantages of each without incurring too many disadvantages. The combination method first reduces the amplitude of the transient voltage and current to a level where the remaining transient can be routed around the aircraft with conventional aircraft shielded wiring techniques. The residual transient is next controlled to an acceptable level by circuit board-size hardware within the temperature controller.

The optimum suppression method must be determined by the specific aircraft design. As a guide, previous trade studies have indicated the combination approach as optimum for the high current heater circuits.

7-702.9.2 Candidate Suppression Hardware

The long thermal constants of heated windshields and the low frequency of the heater current (usually 400 Hz or dc) compared with the rise time of the transient to be controlled permits the use of low-pass wave filters or transient clamps. Each wire must have transient control. This may even apply to return leads if local (multi-point) grounding of the return leads is not acceptable or adequate because of EMC constraints.

Low-pass filters can consist of a series connected inductor or a shunt connected capacitor, or both. The transient is a common mode signal appearing between the conductive coating or temperature sensor element and the fuselage of the aircraft. Therefore, one terminal of
a shunt capacitor must be connected directly to the fuselage with a low impedance connection, if the capacitor is to be effective.

Series inductors, either alone or in combination with capacitors, must carry the full anti-icing current with little voltage drop. The attenuation characteristics and current limitations of filters generally require hardware that is too bulky and too heavy for use when placed at the windshield, and when relied upon for attenuation of the full level transient. Also, full level attenuation via a filter placed in a remote location is usually an inefficient transient control method.

A more weight and space efficient transient control approach for the high energy, higher voltage transients found on windshield/canopy heater wires would consist of transient clamps at the windshield terminals, and small filters located at the controller.

Three types of transient clamps are generally available. These are spark gaps, metal oxide varistors and zener diodes.

The peak current, rise time to peak, and total energy in the lightning static discharge, and NEMP transients are the important factors in choosing a transient clamp. The transient characteristics for each source will depend on the transparency design, the aircraft layout, the size of the aircraft and the environment.

The transient energy dissipation in the suppressor is directly related to the clamping voltage; i.e., the voltage level to which the transient is held during the suppression period. The candidate clamping devices must be selected so that they draw essentially no steady-state current when placed across (line-to-ground) the circuitry they are to protect. For heater circuits this could be 117 volts or higher at 400 Hz. This voltage is not a problem for spark gaps and varistors, which are non-polarized and are available in 7.092
these voltage ratings. Special series-connected, back-to-back zener diode combinations can also be assembled which will operate at this power phase voltage.

It is possible that the transient control hardware could be concentrated in the electronic package that contains the sensor bridge completion circuitry. This approach would probably require that the wiring from the sensor terminals to the electronic circuitry be routed as a twisted shielded pair for each sensor, with the shield grounded at both ends of the wire run. The optimum transient control must result from a thorough integration of the temperature controller and windshield/canopy designs.
7-800 INTRODUCTION

As noted in MIL-E-38453A, (Reference 7.13) when electrically conductive coated windshields or canopies that are to be protected against icing or fogging; the separate control sensing elements, overheat sensing elements, temperature controllers, and power sources shall be provided for each windshield or canopy.

Considering the requirements specified in MIL-E-38453A and MIL-T-5842 (Reference 7.19) and the potential problems associated with swept-stroke lightning and precipitation static noted in the subparagraphs of Section 7-702, it becomes evident that the design and development of the electrical requirements for a reliable transparency are very complex. From the initial design concept to production design for aircraft transparencies, trade studies must be accomplished taking into account the shape, surface configurations for adequate anti-icing, space allocations for suppression systems, and electrical components; proximity of sensing elements to potential lightning attachment, static discharge problems and electrical transients.

7-801 TRANSPARENCY DETAIL ELECTRICAL COMPONENTS

Each forward transparency, windshield or canopy configuration, that must be electrically anti-iced or defogged must have certain electrical features designed into the specific configuration. The electrical features that the design specialists must be familiar with include:

- Materials suitable for conductive coating applications
- Coatings
- Bus bar applications
- Internal wiring
- Sensing elements
- Terminal connections
7-801.1 Conductive Coated Materials

The effectiveness of any electrical conductive coating is directly related to how well the forward fuselage of the aircraft has been shaped. The resultant effect of the shape may mean the windshield could be flat, simple curved or compound curved. Because the surface shape of the area to be heated ultimately becomes a manufacturing problem, the vendors capable of providing the finished transparencies should be consulted to ascertain their limitations for developing final configurations.

7-801.1.1 Conductive Coated Flat Windshields

Generally, flat windshields are laminated glass, in accordance with MIL-G-25871. The outer pane is usually 3/16 inch thick thermally tempered glass, in accordance with MIL-G-25667, that has a coating on the laminated side of stannous oxide (tin) or indium oxide that is utilized for anti-icing. It is possible also to have a static discharge coating on the exterior surface, but such an application must be carefully evaluated because of light transmission loss, aerodynamic erosion, susceptibility to attack by aircraft cleaning compounds, and scratching when windshield wipers are employed. Any of these defects would greatly reduce the reliability of the windshield.

Chemically strengthened glass, per MIL-G-25667, could be substituted for the thermally tempered glass, but an indium oxide or a modified gold coating would be required; otherwise, if stannous oxide were used the tempering would be greatly reduced in the chemically strengthened glass.

Ideally, the flat glass should be rectangular shaped to maximize the amount of area that can be anti-iced. Since rectangular shapes are virtually impossible to obtain, an anti-icing analysis should be
performed, as noted in Section 5, to determine the coating resistivity and bus-to-bus resistance required. Subsequently, discussions should take place with potential vendors regarding manufacturing coating processes and to establish the electrical power requirements. It must be noted that the anti-icing coatings must terminate at the lower edge of the windshield so that the coating, bus bars, or extreme icing conditions do not obstruct the pilot's down-vision, nor should there be obvious deletion lines in the coating. In the event that the coating cannot terminate at the lower edge of the windshield, or deletion lines are required, then the transparencies should be reshaped so that they can be manufactured without deletion lines, yet provide maximum down-vision under all cases.

In the event that chemically strengthened glass is selected, the minimum thickness should not be less than .085 inches, and preferably 0.100 to withstand hail impact.

7-801.1.2 Conductive Coated Simple Curved Windshields

Simple curved windshields may be constructed of laminated glass, in accordance with MIL-G-25871, or may be of composite construction consisting of an outer glass face ply and several plies of plastic material laminated with silicone or urethane interlayers in accordance with vendor proprietary processes. Either thermally tempered or chemically strengthened glass, per MIL-G-25667, may be used as face plies. The depth of crown for thermally tempered glass may limit the applications and the limitation must be determined during initial design by the potential vendor.

The thermally tempered glass may have stannous oxide or indium oxide coating applied for anti-icing or RCS control. Stannous oxide only should be used for static discharge since it is more durable than indium oxide.
If chemically strengthened glass is used as an outer face ply, then indium oxide or a modified gold coating must be used for the anti-ice or RCS applications, otherwise stannous oxide coating applications would greatly reduce the strength of the glass material.

Exterior surface coatings are not recommended because of light transmission loss, aerodynamic erosion, susceptibility to attack by aircraft cleaning compounds, and scratching when windshield wipers are employed. Any of these defects would greatly reduce the reliability of the part.

As-cast or stretched acrylic may be used as face plies, but at the present time, any coating applications will have to be limited to the use of a modified gold coating for anti-icing or RCS applications. Because of the inherently soft gold coating and its low adhesion to plastic substrate, its application for use as a static discharge coating on the exterior surface is not recommended.

Generally, it has been impossible to develop a single contoured shape that will lead to parallel edges for anti-icing coatings. Consequently, the coating has to be graded in some manner based upon the vendors coating capabilities so that a uniform resistance level is seen between each segment of the bus bars. This usually results in dividing the surface area into zones separated by 0.030 to 0.060 inch wide deletion lines. Figure 7.14 is an illustrative example of the type of zoning that could be required. As noted, it would be impossible to anti-ice the entire surface because of coating density limitations. As a result, there will be areas forward and aft that will not anti-ice. These areas should be coated so that there is continuity in the pilot's vision, yet meet the RCS requirements. It is not readily apparent, but at least 5 degrees of forward vision has been lost in this particular application because the forward section will not anti-ice.
There shall be two bus bars located on the conductive coated surface for anti-icing. The two bus bars shall be as close a practicable to being parallel and of the same length. When the coating is zoned, as shown in Figure 7.14, a neutral bus may be used as shown and locally grounded. Often time, however, the ground is accomplished to a common ground internal to the aircraft electronics compartment or through the controller.

The bus bar material normally is proprietary to the transparency manufacturers and is usually 5/16 - 3/8 inch wide, located approximately 1/8 inch from the edge of the transparency material.

When the bus bar is fired onto glass and lead wires are soldered to the bus bar, there must always be a minimum of two lead wires per bus bar with the solder joints at the third points on the bus bar. The wires must be routed through the laminate with sufficient length to compensate for the expansion/contraction rates caused by extreme thermal conditions. The two lead wires could terminate at one terminal, but two terminals are preferable. The aircraft wiring would then require a yoke (or split) so that each terminal would receive only one-half the current.

There may be a need for continuous bus bars around the entire periphery of the outer surface, whether a static coating is used or not. This type of bus bar should have grounding wires attached to the bus bar and structure to provide a discharge or drain path for static buildup. In the event that a bus bar is placed on the outer surface, there may be a need to install a protective covering in the event the material is soft or if wipers are employed.
Figure 7.14. Windshield Zoned with Anti-Ice Coating.
Temperature Sensing Element Design Considerations

The measurement of the windshield temperature is achieved by sensing the resistance variation of a temperature sensitive resistor laminated in the windshield. This resistance is an electrical analogue for the variation in temperature of the windshield.

There are two basic types of temperature sensors that can be used. One type has a positive temperature coefficient and the other a negative temperature coefficient. Both of these types are currently in use in operating aircraft windshield temperature control systems.

Experience gained from the operation of the windshield temperature control system indicates that there have been no reported sensor (sensing element) failures, per se. However, there have been cases of temperature sensing loop failures involving the opening of either the lead wire and/or the solder joints. Since the open sensor is the predominant temperature sensing circuit failure mode, the sensor type having a positive temperature coefficient is the best suited for this application.

Since the sensor resistance is proportional to the sensed windshield temperature, the controller will in the event of an open sensor failure interpret the condition as a high windshield temperature and its normal protection function will cause the controller output power to be inhibited, thereby, preventing overheating the windshield. On the other hand, a system with a negative temperature coefficient sensor would interpret an open sensor condition as a low windshield temperature and, consequently, would increase the heating power. Therefore, in order to allow the system to discriminate between an open sensor or a low windshield temperature condition, additional circuitry would be required.

It is evident that a positive temperature coefficient sensor has a clear advantage over the alternative type. In order to assure that the minimum control point temperature required for anti-icing is maintained, both the sensor manufacturing and controller operation
tolerances must be taken into account. The controller "OFF" point, which typically varies within plus or minus 1.6 ohms of sensor resistance, is tabulated in Table 7.7. Refer to the thermal analysis discussion in Section 5 for the basis of selection of control point temperature.

Table 7.7. Power-Off Control Point Tolerances Versus Temperature

<table>
<thead>
<tr>
<th>Power-Off Control Point Tolerance in OHMS (Sensor Resistance)</th>
<th>Control Point Temperature</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Max.</td>
</tr>
<tr>
<td>339.2 (Max.)</td>
<td>114</td>
</tr>
<tr>
<td>337.6 (Nom.)</td>
<td>112</td>
</tr>
<tr>
<td>336  (Min.)</td>
<td>109</td>
</tr>
</tbody>
</table>

7-801.3.1 Sensing Element Locations

In order to assure accurate windshield temperature sensing, the sensing element must be laminated into the windshield at a controlled location. Four requirements generally control the specific location of each sensing element. The first requirement is that sensing elements cannot be located at the lower or forward edge of the windshield where they would obstruct the pilot's down vision, nor should they be placed in a critical forward vision area. The second requirement is that the sensing elements must be located at least 0.040 inches below the conductive coating for insulation purposes. Since it is virtually impossible for the transparency manufacturers to apply a constant density coating on transparent materials, the resultant effects are hot spots that require the sensing elements to be placed in the close proximity of these hot spots as described in 7.102.
 Thus, the third requirement is the placement of the sensing elements to control "K" factors. Ideally, these "K" factors should be all equal to one, but since that is impossible, the following "K" factors are recommended:

\[
K_H = 1.40 \text{ or less } = \frac{\text{power density at hot spot}}{\text{power density at control point}}
\]

\[
K_A = 0.70 \text{ or greater } = \frac{\text{average power density}}{\text{power density at control point}}
\]

\[
K_M = 0.50 \text{ or greater } = \frac{\text{average power density}}{\text{power density at hot spot}}
\]

\[
K_C = 0.55 \text{ or greater } = \frac{\text{power density at cold spot}}{\text{power density at control point}}
\]

If the third requirement is violated, the resultant effect for a glass construction is large temperature differentials across the glass pane that can easily cause the glass to crack. This thermal shock can be minimized if the sensing elements are correctly located and if the electrical power is applied gradually to the coating in order to allow its temperature to rise slowly to the working point level. Generally, the optimum time for the temperature ramp to rise to the working level is approximately 3 minutes.

The fourth requirement is that the sensing elements should not be placed in areas of the transparency where there is a high probability of a swept-stroke lightning path.

In order to satisfy these requirements, trade studies must be accomplished early in the design phases. The transparency manufacturer should evaluate the design and establish the approximate locations of the sensing elements so that the radome and other lightning attach points can be established.
Since the sensing elements must be laminated into the windshield at a controlled location, such designs do not permit repair or replacement of the sensing elements. Therefore, a redundant spare should be incorporated in the design for each heating area, and in the event of failure of the sensor, the spare can readily be connected to serve as the active sensor. Frequently, it is desirable to have visual aids for the pilot to know whether the windshield is over temperature or below temperature to adequately anti-ice and maintain desired reliability of the windshield. Consequently, an over-temp/under-temp light is installed. The function of this light is controlled by an additional sensing element embedded in the windshield. Therefore, three sensing elements could be installed for each heated area defined.

Within the transparency panel, each pair of sensing element wires from an element should be insulated, twisted and shielded together because of lightning, static discharge and NEMP effects, as noted in paragraph 7-701,5. They should also be as short as possible, yet must have sufficient length to preclude failure due to thermal differences. Previously, the common ground between two terminals' sensing elements have terminated at a common terminal. On occasion, solder joints at such a terminal have failed, thus, defeating the purpose of a redundant sensing element. Each sensing element wire should terminate at an individual terminal block.

7-801.3.2 Sensing Element Selection

A commonly used sensing element is the Westinghouse AVK 1160. When laminated in a windshield, this device must have resistance values that fall within the Westinghouse curve 342764A, as shown in Figure 7.15.
The temperature of each conductive coating must be controlled within a relatively narrow range or control band. In addition, when the windshield conductive coating has to be divided into zones, each zone must have three sensing elements, and the temperatures in adjacent zones must be very close, to avoid undesirable thermal stresses and optical distortion. Since the sensor resistance is a function of temperature, sensor self-heating must also be considered. The temperature sensing circuit must be designed to permit no more than 5 milliamperes to flow in the sensor at the lowest ambient temperature in order to prevent appreciable sensor self-heating. The controller should be designed to simultaneously monitor all active sensors in order to continuously receive and process temperature information. Comparisons of sensing element resistances are then made by the controller detection/discrimination circuits. The sensing element having the highest temperature and resistance value will function as the "control" sensing element, thus overriding the other sensing elements in descending order by temperature. This method is expected to produce accurate temperature control, yet to prevent overheating the heater zones. Typical controller power requirement and sensor temperature tolerance ranges for the reference design are shown in Figure 7.16.
7-801.4 **Transparency Terminal Blocks**

The powerlead wires and sensing element wiring can terminate either at embedded terminals in the windshield or may have a length of wire emanating from the windshield and have loose external terminals attached. Because of the fine wires and the handling frequency, the windshield is expected to experience, before installation, the use of loose wiring or "pig tails" is not recommended since breakage is likely and repair would be virtually impossible.

Embedded brass terminals are the recommended practice. Each sensing element wire should be firmly connected to individual terminals. Each power lead wire should be connected to individual terminals, but when two lead wires are used for fired on bus bars, it may be possible to solder two power leads to one terminal.

A special strip should be designed into the windshield that would contain all of the brass terminals. The strip should always be easily accessible from inside the aircraft for maintenance practices.
The terminals should be separated, not clustered, to enable the transparency manufacturer to solder the wires.

NOTE: Manufacture of terminals must be controlled to prevent power heating (IR) of the terminal block. This heating has resulted in local failure of the transparency and smoke in the cockpit.

Experience has shown that the procurement specification should specify that each external terminal installation should be capable of withstanding a 50 inch torque load without failure to the connection and resist a 100 pound load applied perpendicular to the terminal.

7-801.5 Windshield Edge Heater Technical Discussion

The design objective of windshield edge heating is primarily to minimize the temperature gradients that will exist between the heated area of the transparency and its framing structure. The expected benefit from edge heating is a reduction or elimination of edge delamination and an improvement of edge sealing to increase the transparency life as well as reduction of the controller failures by decreasing the amount of on-off cycling.

Some commercial airline operators have experimented with windshield edge heating. Their approach was a form of trial and error experimentation intended to determine the amount of heat that is required to produce the most beneficial results.

The preliminary results of these efforts have shown some reduction in the amount of transparency panel delamination.

It should also be noted that the problems that have been experienced in this area are primarily concerned with large glass panels laminated with PVB interlayers, which have a very limited mechanical performance range compared to the anticipated thermal variations experienced between a heated area and an unheated area. It remains to be seen how some of the new polyurethane interlayers might perform under similar circumstances.
7-801.6 **Recommended Testing**

7-801.6.1 **Transparency Anti-Ice Over-Voltage Requirement**

Each of the electrical conductive coating circuits should be capable of withstanding, without deterioration or failure, the application of an over-voltage of 150 percent maximum operating voltage for the circuit.

7-801.6.2 **Transparency Electrical Insulation Requirements**

As a part of the acceptance criteria for a windshield, certain anti-ice electrical insulation tests should be performed at the transparency manufacturer's facility as detailed below.

a. An alternating current potential of 2500 volts RMS at 60 Hz shall be applied for a period of one minute between the following:

- For each sensing element terminal and each power terminal, the leakage current shall not exceed 0.5 milliamperes.
- For each sensing element terminal and any metal insert (and spacer), the leakage current shall not exceed 0.5 milliamperes.
- For each power terminal and metal insert and spacer, the leakage current shall not exceed 4.0 milliamperes.
- The leakage current shall not exceed 4.0 milliamperes when tested between each power terminal and a metal strip placed in contact with the edge of the glass pane. The entire glass peripheral edge must be tested, but may be tested in sections.
In the event there is also a defog coating, then for each anti-ice and defog terminal the leakage current shall not exceed 4.0 milliamperes.

The rate of potential application shall be at 500 volts RMS per second.

b. Repeat the above test with direct current of 500 volts applied. Resistance of less than 100 megohms shall be cause for rejection.

7-802 AIRCRAFT TO TRANSPARENCY ELECTRICAL SYSTEM DETAIL

COMPONENT DESCRIPTION

The electrical components internal to the aircraft that provide the power and control for the windshield anti-ice system must include the following components:

- Controller
- Wire bundle
- Suppression system
- Power source

7-802.1 Electrical Controller

According to MIL-E-38453A, (Reference 7.13) solid state electronic controllers are to be protected against voltage spikes in the electrical power supply and designed to compensate for any change in ambient conditions surrounding the controller. Modulating type controllers shall not induce unacceptable voltage transients in the aircraft electrical power supply.

In subsequent paragraphs, five practical solid state controllers are described and their respective functions and characteristics are summarized.
7-802.1.1 On-Off Controller Design

The on-off controller (sometimes identified as a bang/bang controller) is basically a two-state device, whose full power output is supplied in response to sensor demand. The mode of operation is very simple - when the sensed windshield temperature is below the designated set-point temperature, full electrical power is switched to the anti-ice heater. Full power is switched off when the sensed windshield temperature reaches the set-point temperature. This constitutes one on-off cycle. This design may incorporate a ramp power warm-up mode. Upon initial system turn-on, the average power delivered to the heater increases with time at a designated rate until the windshield control temperature is reached. Upon reaching the control temperature, the controller goes into an on-off power switching mode. A voltage transformer may be inserted to increase or decrease the heater voltage required for a particular design. Due to the inherently high switching frequency, solid state output devices are normally used to achieve extended equipment life and higher operational reliability, in addition to their lighter weight.

Historically, when bang/bang type controllers have been used to control electrically heated windshields, it greatly reduces the reliability of the windshield. The failure cause is thermal shock that results in cracked glass or a solder joint failure at the internal juncture of the windshield power lead wires and the bus bars.

7-802.1.2 Magnetic Amplifier Controller Design

The magnetic amplifier (MAGAMP) is a variable-reactance device, having no moving parts, whose output power level is proportional to the sensor demand. The MAGAMP is also known as a "Transactor" suggesting that its major components consist of a transformer and a saturable reactor.
7-802.1.3 Pulse Modulated Controller Design

The pulse modulated controller (PMC) is an energy proportional design. This design requires one pair of solid-state switches per phase to conduct current to the load. In order to minimize switching electromagnetic interference, it is usually designed to switch the entire ac cycle current to the load. The number of controller output cycles is made proportional to the temperature sensor demand. Depending on the anti-ice heater voltage required for a specific design, a power transformer can be inserted to achieve the desired voltage conversion. This design can be adopted for a phase-to-neutral, a phase-to-phase, or a three-phase heating arrangement.

7-802.1.4 Variable Firing Angle Controller Design

The variable firing angle (VFA) controller is a solid-state simulation of the MAGAMP type controller. This design provides excellent power proportioning to the heater. Power proportioning is achieved by varying the conduction angle of each cycle of 400 Hz power in proportion to the sensor demand, thus varying the fraction of the energy available in each cycle which is allowed to reach the heater.

7-802.1.5 Static Tap Changer Controller Design

The static tap changer (stc) is a proportional controller designed to switch/select the voltage taps which are available on a multiple tap power transformer through which the heater load current is conducted. The successive taps which are energized and dwell time sustained on each tap are functions of the windshield temperature status and sensor signals.

7-802.1.6 Additional Controller Design Features

The types of controller designs described previously may be incorporated with additional features; among these are the following:
o **Ramp Warm-Up.** Upon initial system turn-on, the controller output power is scheduled at a designated increasing rate until the windshield set-point temperature is reached. The ramp power output is designed to prevent thermal shock to a cold-soaked windshield. A reverse ramp (cool-down ramp) design may also be incorporated to reduce the thermal stress which could occur as a result of rapid cooling under extremely cold environmental conditions.

o **Shorted Sensor Protection.** This type of protection is intended to prevent the windshield from being overheated when a sensor failure causes its resistance to drop, resulting in the controller receiving erroneous temperature information. For a sensor having a positive resistance temperature coefficient, a low sensor resistance is interpreted by the controller as a demand for heating. An appropriate trip threshold can be selected such that an occurrence of this type of failure will automatically enable the controller to inhibit or shutoff power to the windshield.

o **Overheat Protection.** The purpose of this feature is to prevent overheating of the windshield in the event of a control circuit or output switching device failure in the controller. Typically, solid-state logic circuitry is incorporated to detect such failures and either to command the removal of heater power or to establish an inhibit or lock-out function to prevent the application of heater power.

The advantages and disadvantages of the five controller types described above are listed in Table 7.8. A conservative design is required to ensure acceptable operational reliability.
<table>
<thead>
<tr>
<th>Design Type</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>On-Off Controller</strong></td>
<td>• Simple circuitry (solid-state)</td>
<td>• Lacks power proportional control</td>
</tr>
<tr>
<td>(ON-OFF)</td>
<td>• Acceptable option available for ramp warmup</td>
<td>• When a transformer is used, circuit power factor variation requires adequate switching, timing control, and better transformer design to prevent half-cycling and transformer saturation</td>
</tr>
<tr>
<td></td>
<td>• May include a transformer in the power circuit for voltage conversion</td>
<td>• Small envelope size</td>
</tr>
<tr>
<td></td>
<td>• Low unit weight</td>
<td>• High electrical efficiency</td>
</tr>
<tr>
<td></td>
<td>• High electrical efficiency</td>
<td></td>
</tr>
<tr>
<td><strong>MAGAMP Controller</strong></td>
<td>• Simple circuitry</td>
<td>• High unit weight-to-power capacity ratio</td>
</tr>
<tr>
<td>(MAGAMP)</td>
<td>• Inherently immune to electrical transients from external sources, if proper control circuit buffering is provided</td>
<td>• Efficiency problem can be significant for a high output power requirement.</td>
</tr>
<tr>
<td></td>
<td>• Rugged in construction</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• High reliability in power circuit</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Requires little EMI suppression</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Excellent power proportional control</td>
<td></td>
</tr>
<tr>
<td><strong>Pulse-Modulate Controller</strong></td>
<td>• One pair of output switching devices per phase.</td>
<td>• May cause amplitude modulation to the power source and/or phase voltage unbalance</td>
</tr>
<tr>
<td>(PMC)</td>
<td>• Few non-solid-state components</td>
<td>• Requires a design for protection against interference from extraneous electrical transients</td>
</tr>
<tr>
<td></td>
<td>• High potential reliability</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Requires little EMI suppression</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• May interact with the power system regulation system and may aggravate phase load unbalance for single or two-phase design.</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Low unit weight-to-power-capacity ratio, if a transformer is not required</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Small envelope size, if a transformer is not required</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• Energy-proportional control (constant; output voltage level)</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• High efficiency</td>
<td></td>
</tr>
<tr>
<td></td>
<td>• High power factor</td>
<td></td>
</tr>
</tbody>
</table>

7.113
<table>
<thead>
<tr>
<th>Design Type</th>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Variable Firing Angle Controller (VFA)</td>
<td>• Good power proportioning control (voltage and current proportioning)</td>
<td>• Requires additional EMI filtering for either the zero crossover switch-on (switch-off at some conduction angle up to 180° electrical degrees), or switch-on at some conduction angle (switch-off at load current zero crossover)</td>
</tr>
<tr>
<td></td>
<td>• Smooth and linear power ramp warmup</td>
<td>• Low efficiency</td>
</tr>
<tr>
<td></td>
<td>• Low unit weight-to-power capacity ratio</td>
<td>• High switching duty cycle (switches twice in every cycle)</td>
</tr>
<tr>
<td></td>
<td></td>
<td>• Not compatible with the peak-sensing type of power-system voltage-regulator design</td>
</tr>
<tr>
<td>Static Tap Change Control (STC)</td>
<td>• Good power proportioning control (voltage and current proportioning)</td>
<td>• High unit weight-to-power capacity ratio</td>
</tr>
<tr>
<td></td>
<td>• Power ramp warmup in discrete voltage steps</td>
<td>• Medium (moderate) efficiency rating</td>
</tr>
<tr>
<td></td>
<td>• Does not introduce interference in power system</td>
<td>• Requires a transformer for output voltage tapping</td>
</tr>
<tr>
<td></td>
<td>• Input may be either single-phase or phase-to-phase design, if source capacity is adequate</td>
<td>• Prone to cause a tap-to-tap short-circuit, requiring an adequate preventive or protective design.</td>
</tr>
<tr>
<td></td>
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<td>• Medium-to-high circuit complexity</td>
</tr>
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</table>
7-802.1.7 Operational Anomalies and Protective Control Logic

There are three categories of system component failures which could cause a system operational anomaly; these are sensor element circuit failure, controller control circuit failure and controller output stage failure. The reference study temperature - sensing design described in paragraph 7-801.3.1 - would allow up to two sensing element failures in one mission, if a three-zone conductive coating configuration is used, yet would still retain a partial anti-icing capability. Sensing element failures and their consequences must be accommodated in the control circuit design.

7-802.1.7.1 Sensing Element Short Circuit - Physical short circuits (shorting) of the sensing element, or its electrical circuit, would not occur within the transparency without action by some external agent. Physical shorting could occur in the sensing element circuit external to the windshield proper. In either case, a shorted sensing element would not necessarily be the cause of losing all the anti-icing capabilities of the windshield. Such failures have been rare in commercial aircraft experience, but have occurred often enough that redundant sensors should be provided for the anti-icing system to enhance operational reliability.

The sensing element having the higher resistance value when above a designated shorted sensor threshold should be designated as the "control" sensor as previously described. The incorporated logic would allow the controller to ignore up to two shorted sensors in a system similar to Figure 7.14. Therefore, the heater sector having a shorted sensing element would continue to be anti-iced in accordance with the demand of the coldest sector not having a shorted sensing element. A sensing element whose resistance falls below a value designated as the shorted sensing element threshold may be ignored. The correct sensing threshold should be determined from the worst temperature environment in which the aircraft is designed to operate.
Normal system operation would continue and the temperature sensing logic could then allow a second sensor to fail in a shorted mode without losing anti-icing capability. However, in the event that the remaining sensor circuit should also fail in a short or open mode, the controller would inhibit all output power.

7.802.1.7.2 Sensing Element Open Circuit - The open circuited (open) sensing element or open sensing element circuit failure mode should not be allowed to be the cause of windshield replacement. The spare sensing element should be designed to readily replace the failed sensing element. In the event that the active sensing element should fail in an open mode, the windshield anti-icing capability would not be lost. In order to achieve the goal of maintaining anti-icing capability for a system similar to Figure 7.14, with up to two sensing elements failed in an open mode, the failure criterion for an "open" sensing element selected to be 1000 ohms + 10 percent of sensing element resistance. This resistance value provides sufficient margin by which the controller can readily distinguish a high sensing element resistance which could be due to a high ambient temperature condition such as might be caused by aerodynamic heating from an actual "open" sensing element. "Open" sensing elements are rare in aircraft operational experience.

Should the first sensing element fail in the open mode (i.e., sensing element resistance value exceeding 1000 ohms), the controller would ignore it. One of the other two active sensors would then function as the control sensor for a system such as shown in Figure 7.14. Should the second sensing element subsequently fail in the open mode, the remaining sensing element would be the control element and the system would continue to function until this remaining operative element should fail either in the open or the shorted mode. Should the third sensing element also fail, the controller output to all three zones would be inhibited. It should be noted that this system can provide a fail-safe back-up to the conventional or E-MUX control system.
Active Sensing Element Replacement - In the event of an active sensing element failure, the spare sensing element should be activated. It can be placed into active duty either automatically or manually.

The controller should be designed with an additional manual or automatic switching circuit to cut out the failed sensing element and to switch in the spare sensing element(s). This design approach would require six more wires to the controller and at the windshield wire bundle, and a larger aircraft wiring interface. If an automatic switching design is adopted and there is no spare sensing element (e.g., the spare sensing element is already in use as a substitute element), the controller must automatically perform the following sequence of tasks: (1) find a suitable spare sensing element, preferably in the same sector; (2) disconnect the existing sensing element circuit; and (3) reconnect the suitable spare sensing element to the zone control circuit.

If the other sensing element in the affected zone is already failed or was simultaneously failed, a sensing element of another zone may be selected to provide three active sensing elements for control redundancy, as long as there are three functional sensors for a design similar to Figure 7.14.

A second alternative is one that requires manual switching of the failed sensing element then switching the spare sensing element into the feedback loop of the system. This design approach also requires additional wires and a multiple pole selector switch. This switch may be located either on the controller housing or remotely in the flight deck. This design option would avoid additional connector shielding problems, but would require additional crew training, task loading in the flight environment and utilization of cockpit space for the switches.
A third alternative may be the most desirable in terms of cost, considering the frequency of sensor failure. This design option requires manual interchanging of sensing element wires at a wiring connector, preferably at the controller connector in order to permit the circuit change in an accessible location. This design approach to the sensing element replacement problem would require four twisted and shielded pairs of sensing element wires (one pair for the active sensing element and one pair for the spare sensing element) per zone to be terminated at the controller connector. The other two twisted and shielded pairs of spare sensing element wires from each zone would be routed to the connector, but not terminated therein. Instead, each spare sensing element wire would be routed, along with the active sensing element wire ends, to the connector. The insulation would be stripped from the spare sensing element wire ends, connector pins installed, and a jumper connection installed to short each spare sensing element circuit, so that electrical noise voltage would be suppressed. All sensing element wires would require proper identification as to their sector origin, in order to facilitate production and maintenance. On the female half of the mating connector, preferably in the controller housing, there would be a total of only six twisted and shielded pairs of wires (two per zone), and a smaller sensing element wiring connector could be used at the controller. The maintenance procedure would be relatively simple. It would require only that the failed sensing element wires be removed from the male half of the wiring connector, their ends be jumper connected together, and the pinned spare sensing element wires be inserted in their places.

The approach using manual interchanging of sensor wires would have the advantages of cost and installation space reduction, and better accessibility at the controller box. It would also prevent inadvertent reconnection of inoperative sensing element circuits by air or ground crews. The system should also be inherently more reliable, due to the reduction of circuit components. The controller would remain fully interchangeable, while the cable changes would match the associated windshield sensing element condition.
Disadvantages of this design would be connector design requirements and lack of in-flight repair. However, a single or even a double sensing element failure would not alone terminate the windshield de-icing function for a multiple circuit system. This design option would be worthy of future consideration based on component availability.

7-802.1.8 System Operation Under Anomalous Conditions

The system as designated, must not only operate within conditions defined as "normal", but also must be able to survive harmful external transients or temporary failures of the input power supply and must recover to an operational state at the termination of the anomalous condition.

7-802.1.8.1 Power Interruption During Ramp Warm-Up Period

Electrical power interruption as the result of an ac bus transfer, which could occur for a variety of unrelated reasons, may last between 50 and 200 milliseconds, depending mainly upon the operating speed of the bus power contactor used in the ac power system. The duration between power removal and reapplication resulting from a manual command may be 500 milliseconds or longer. If a power interruption occurs for any reason during ramp operation, the transparency temperature will start to drop. The cooling rate will, of course, be dependent on the transparency temperature and the existing ambient temperature. When power is removed and subsequently reapplied, the rate of thermal energy input to the transparency must not be excessive. It is, however, both unnecessary and undesirable to have the warm-up ramp function restarted each time power is only briefly interrupted.

Ideally, the warm-up ramp operation would resume the scheduling of power to the heaters at the rate which would be nominal for the windshield temperature at the time the source power is reapplied. For example, consider the control shown in Figure 7.17, assume that
the ramp is set to minimum rate and that the ramp has been in operation for two minutes. Power is being applied at 50 percent of the maximum rate. If at this instant there is a source power interruption which lasts for one-half minute, the windshield would cool to a temperature where power must be applied at a rate equal to 37-1/2 percent of its maximum. The full regime of percentage output power versus time shown in Figure 7.17 should govern the warm-up ramp recover mode of operation.

![Controller Power Warm-up Ramp](Image)

Figure 7.17. Controller Power Warm-up Ramp.
For the case where the duration of power interruption is equal to or greater than the previously elapsed warm-up ramp time (from initial turn-on to the power interruption), the allowable power levels would, of course, have returned to zero; therefore, the warm-up ramp mode of operation would be restarted.

7-802.1.8.2 Power Interruption During Post-Ramp Operation - In the event that a power interruption, such as an ac bus transfer, occurs during steady-state heater operation, the controller should subsequently resume its proportional control function. The rationale is that the transparency temperature reduction during this normally very short period of time (50 - 200 msecs.) would be insignificant, and the resulting thermal stress would be well within the transparency material tolerance. In steady-state operation (during the post-ramp proportional mode), the transparency would be kept at the controlled temperature. Subsequent to a power interruption, the transparency temperature would drop at an exponentially decreasing rate, commencing at an initial maximum rate which would be dependent only on the difference between the transparency and environmental temperatures.

Subsequently, when power is restored, if the maximum warm-up ramp should be applied, the maximum transparency heat-up rate would be 120°F during the first second after power application. Therefore, in order to minimize the potential for thermal shock, the controller could be allowed to ignore the power interruptions of 200 milliseconds or less. For power interruptions of more than 200 milliseconds, the warm-up ramp power level should be applied as described previously for power interruptions occurring during ramp warm-up.
The electrical windshield heating system design concepts must be integrated with the physical requirements of the windshield. Design recommendations of this concept are shown by example in this paragraph.

With reference to Figure 7.14, a three-sector, three-phase heating design includes seventeen power and sensor terminals and eight grounding terminals. The power conductor bundles are shown in Figure 7.18. The sensor conductor bundles are shown in Figure 7.19 with detailed view of the electrical disconnect terminal designs in Figure 7.20.

The conductive coating which constitutes the heater is divided into three equal areas. Any additional windshield area which proves difficult to incorporate into the heater pattern may be coated for aesthetics or even radar cross-sectional (RCS) control, but this area need not be electrically heated if it is in a noncritical visual area. Deletion lines, each of 30 + 3 mils in width will provide electrical isolation while restraining the electrical field stress to an acceptable level to prevent phase-to-phase or phase-to-frame arc-over.

The reference windshield design (Figure 7.14) shows a common neutral bus bar parallel to the center part edge. Two braided and insulated neutral conductors, one connected to each bus bar extremity, are also connected individually to the two separate neutral terminals identified as "M". One neutral terminal is located at the forward end and the other is located at the aft end of the sill. Three power supply bus bars, one for each phase, are isolated and positioned along the sill. Each power supply bus bar is connected to the corresponding phase power terminal by a braided and insulated conductor routed internal to the windshield. Each of the five power terminals is connected to the windshield power conductor wire bundle (Figure 7.20) with a separate mating electrical terminal block.
The sensor terminals, four per heater sector, as shown in Figure 7.14, are connected to the sensor wire bundles with three electrical terminal blocks, one for each sector and its associated group of four terminals.

The twisted, shielded sensor wires are grouped in one bundle, which is then enclosed by an overall EMP shield. The power wires are grouped in a separate bundle. Both wire bundles, power and sensor, are routed aft toward the side (aft) windshield supporting structure post. The sensor wires and the power wires are terminated in the male half of a connector. The power wires are unshielded and are similarly routed and terminated at the male half of a second connector.

With this reference wiring system design, the electrical terminal blocks are connected to the windshield and the bundles are secured to the sill prior to installing the windshield. The wire bundles are designed with sufficient length to reach the aircraft installed wiring harness connectors. These connectors can be mated, connected or disconnected from the aircraft cables to the electrical heating controller, without removal of or disturbance to the windshield; thus, maintenance electrical tests can be made.

7-802.3 Suppression Hardware

7-802.3.1 Varistors and Zener Diodes

Varistors and zener diodes have transient voltage clamping levels that must be above the peak ac power phase voltage to be effective. This level is dependent to some degree on the transient current and will increase with an increase of transient current. Comparing the varistor and zener diode, the diode generally has the better voltage versus current characteristics; i.e., the clamped transient does not rise as high with a zener diode as it does with a varistor.
Figure 7.18. Power Wire Bundle on Windshield Diagram.
Figure 7.19. Sensor Wire Bundle on Windshield Diagram.
Figure 7.20. Typical Electrical Disconnect Terminal Block Installation and Detail
Spark gaps, unlike varistors and zener diodes, present a very high impedance to the normal ac phase voltage, but develop a very low, and almost constant, voltage drop when triggered into conduction by a transient. For a properly selected and installed spark gap, the clamping level of the voltage across the gap during the transient may be on the order of 15 to 20 volts. This contrasts with a value of 500 to 1000 volts or more for a zener diode or varistor for some windshield applications.

The transient energy that must be dissipated in the suppressor is directly related to the clamping voltage. Therefore, when higher voltage circuits must be protected, a spark gap would be called upon to dissipate less transient energy than would a varistor or zener diode.

Spark gaps have one distinct disadvantage when they are placed across a steady-state power phase voltage. The very low voltage drop across the arc generally will be sustained by the steady-state line voltage long after the transient has subsided. For dc power supply voltages above the arc sustaining level, the arc will continue until the spark gap is destroyed. For ac power supply voltages, the arc may be sustained for approximately one-half cycle of the ac supply frequency. This is much longer than the transient duration. If the spark gap electrodes have become very hot during the transient discharge period and the coincident ac power half-cycle, the ionization level within the gap may be sufficient to trigger the arc into a thermionic run-away condition, thus resulting in the destruction of the spark gap.
7-802.3.2.1 Spark Gap Protection - Techniques are available to confine the arc time to a portion of the half-cycle ac time period containing the over-voltage transient, and thus to protect the spark gap. The most common technique is to place a current limiting impedance or resistance in series with the spark gap or in series with the ac power conductor for the phase across which the gap is placed. The value of this resistance or impedance is selected to limit the ac phase current through the arc to a safe value for the specific gap.

When the current limiting impedance or resistance is in series with the gap (in the transient path to ground), the voltage drop across the series-connected gap current limiter can become quite high and can reduce or obliterate the transient clamping effectiveness of the gap. As an example, one ohm of resistance in series with a specific gap would allow a transient of more than 9000 volts to appear on the ac power phase if a 9000 ampere swept stroke lightning transient were to be impressed on the windshield heating circuit. While the one ohm resistance would protect this gap from destruction by ac follow-on current, the resulting voltage transient would probably destroy the SCR's in the temperature controller.

If the protective resistance were placed in series with the ac power phase feeding the spark gap and windshield, and if the gap were effectively grounded, the transient on the ac power phase would be clamped at a much lower value, close to the arc drop of 15 or 20 volts. A pre-arc very short duration transient of perhaps 800 to 1000 volts might be seen, depending on the specific gap. This circuit location for the protective current limiting resistance would have a distinct disadvantage in that there would be a steady-state loss of approximately 10 volts in the supply voltage to the windshield, and a loss of about 200 watts of power (assuming a 10 amperes load) with an additional requirement for removal or dissipation of this amount of heat.
If an inductance were placed in series with the ac power phase supplying the spark gap, and if the ac impedance of the inductance were also one ohm, the current-limiting action of the inductance would be essentially the same as the case when the protective resistance was in series with the gap, but the effect on the voltage across the anti-icing resistance film on the windshield would be negligible. This is because the voltage drop across the inductance is 90 degrees electrically out of phase with the voltage across the resistance of the anti-icing coating. The series inductance would also be more effective than the resistor in reducing the pre-arc transient seen by the temperature controller. Thus, it may be concluded that protection from ac follow-on current destruction of a spark gap can be effectively accomplished by placing a suitable inductor in series with the windshield heater film and by then placing the spark gap directly across the transient source.

Another, and perhaps an optimum, method of protecting a spark gap from ac follow-through current damage would be to employ over current sensing and electronic shut-down in the temperature controller. Modern controllers usually employ SCR's to control the output current. The SCR gate circuitry could be arranged to inhibit turn-on for the subsequent one-half cycle periods if an over-current is sensed during the preceding half-cycle period. Therefore, the ac current limiting of a "fired" spark gap would only be required to prevent unacceptable damage during one-half cycle of the ac power frequency. Refiring during subsequent half cycles would thus be prevented by the over-current sensing and electronic shut-down circuitry within the controller. This approach requires very close coordination between the transient-suppression design and the temperature-controller design. The feasibility of the spark gap approach is further enhanced by the opportunity to incorporate a current-limiting reactance, an electromagnetic interference (EMI) filter and electronic over-current shut-down within the temperature controller.
A peculiarity of spark gap suppression should be noted. When a spark gap is in the arc (protection) mode, as is the case during most of the suppression time period, the arc produces considerable EMI energy, often referred to as electrical "hash". A detailed oscillographic inspection of the clamped voltage will show an unsteady signal containing many short time bursts. This signal has the potential of producing EMI which is conducted by and radiated from the spark gap wiring. Interference control measures may be required if this hash can be detrimental to the heat controller or other systems within the aircraft. Figure 7.20 shows a schematic diagram of an optimized windshield heater circuit transient protection method.

LEGEND: 1. Spark gap arrester located at windshield.
2. Anti-icing element built into windshield faceply.
3. Hash filter and surge current limiting impedance - included in anti-icing controller.
4. Electronic over-current detector and current shut-down control - included in anti-icing controller.

Figure 7.20. Optimized Windshield Heater Circuit Transient Protection Method
Spark Gap Installation Precautions—The optimized transient protection method for anti-icing circuits requires that only spark gaps be located directly at the windshield/canopy heat terminals. When power current follow a protection is used, the energy dissipation in the spark gap is limited to the original transient energy and one-half cycle of ac power energy. This enables the use of a spark gap that is physically small. Candidate gaps are produced by several manufacturers. Proprietary techniques are used to obtain fast clamping, low overshoot, and high current capabilities. Much of the development for this type of gap was originally done for the communications industry where large quantities of spark gaps are used.

Although only a small space is required for a modern spark gap, one gap is usually required for each wire to be protected. One electrode of each gap must be directly connected to the airframe structure, usually via the metal frame of the windshield or canopy. A wire ground lead of only a foot or so might have sufficient impedance to partially or totally negate the transient clamping action. An integral structure ground is preferable. For the above reason, it is obvious that very early close cooperation among all affected design disciplines is necessary if a good overall installation is to be achieved.

Figure 7.21 is a suggested typical power disconnect terminal block installation with the spark gap integral to the installation. This installation mounts on the windshield frame. One terminal post is integrally grounded to the windshield/canopy frame. The ground terminal post is used for the wire shield ground at the windshield/canopy end of the wire run between the transparency and the heater circuit electrical control box. Each design will have its specific design requirements that may necessitate some modifications to this approach.
Figure 7.21. Typical Electrical Power Disconnect Terminal Block
Installation and Detail With Integral Spark Gap

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Utilizing the thermal analysis noted in Section 5, the detail components of the subsystem may be analyzed as noted in subsequent paragraphs.

7-803.1 Windshield Electrical Heating Design Calculations

The three-phase (three sector) heating design of Figure 7.14 was used as a selected design basis. (Note: Some operational experience with a coating system which utilized deletion lines has been unfavorable because of visual distraction. This type of design must not, therefore, be used indiscriminately.) An example of the environmental exposure conditions and the resulting face ply temperatures to arrive at this point can be found in Reference 7.31. The electrical system analysis and calculations are then shown in the following paragraphs.

7-803.1.1 Sector Resistance Calculations

The electrical system information indicated that the resistance of the power transmission line (between the controller output and the heater) is considered to be negligible, due to the short length of the wire circuit (150 inches). The ac bus voltage available for controller utilization varies within the range of 208 volts RMS to 244 volts RMS. Assuming a two-volt drop within the controller to allow for EMI filtering and a silicone controlled rectifier (SCR), the voltage available for the anti-ice heater would be between 206 volts RMS and 242 volts RMS. Starting the 1600 BTU/HR.-FT², which is the required thermal energy determined from Section 5 analysis, this quantity is corrected into the electrical power requirements as follows:

\[ Q = \frac{kE^2}{RA} \]  

(7.30)
where:

\[ Q = \text{thermal energy, BTU/HR.-FT}^2 \]
\[ E = \text{the applied phase voltage or potential, volts RMS, line-to-neutral} \]
\[ R = \text{the heater bus-to-bus resistance, ohms} \]
\[ A = \text{the heater area, ft.}^2 \]
\[ K = \text{a conversion factor} - 3.413 \text{ BTU/Watt} \]

In a practical design, the voltage drop internal to the controller may be as much as 8 volts RMS. This variation generally depends upon designed load current limitation and EMI filtering.

The transparency is divided into three heating sectors, as previously shown in Figure 7.14, each having 510.32 in.$^2$ or 3.54 ft.$^2$ of heated area for anti-icing. The total area to be heated is 1530.96 in.$^2$ or 10.63 ft.$^2$. The total amount of electrical power, \( P_T \), required to develop 1600 BTU/HR-FT$^2$ per MIL-T-5842 would be:

\[
P_T = \left( \frac{1600 \text{ BTU}}{\text{hr.-ft.}^2} \right) 10.63 \text{ ft.}^2 \left( \frac{\text{watt-hr.}}{3.413 \text{ BTU}} \right) = 4983.3 \text{ watts}
\]

The electrical power delivered to each sector, \( P_T \), is therefore 1/3 (4,983.3) watts or 1661.1 watts.

The heater resistance is essentially constant throughout the temperature range, when electrical power is applied to produce the required thermal energy for anti-icing. Therefore, 1600 BTU/HR-FT$^2$ must be developed at the lowest phase line-to-neutral voltage; i.e., 206 volts RMS. Rearranging the above equation for heater sector, the maximum resistance \( R_{\text{max}} \) is:

\[
R_{\text{max}} = \frac{(3.413)^2 (206)^2}{(1600)(3.54)} = 25.57 \text{ ohms}
\]
Considering the hardware manufacturing tolerance to be 10 percent, which a windshield manufacturer is normally capable of holding with a production yield rate that is economically acceptable, the heater resistance (bus-to-bus resistance per sector) would be between 20.93 ohms and 25.57 ohms; hence, 23.25 ± 10 percent.

7-803.1.2 Heater Power Density

The highest power density which the windshield anti-ice heater conductive coating is required to sustain, without degradation, with the highest phase line-to-neutral voltage (242 volts) and the bus-to-bus resistance at the lowest permissible manufacturing tolerance (20.93 ohms) would be 5.48 watts/in.². These values are computed as follows:

a. At the highest phase line-to-neutral voltage ($E_{\text{max}}$) and the lowest heater resistance ($R_{\text{min}}$), the power density, $P_{d_{\text{max}}}$, would be:

\[
P_{d_{\text{max}}} = \frac{(E_{\text{max}})^2}{(R_{\text{min}})(A)}
\]

where $A$ is area in square inches (7.31)

\[
= \frac{(242)^2}{(20.93)(510.32)}
\]

\[
= 5.48 \text{ watts/in.}^2
\]

*This is the RMS values, which should not be confused with the peak or maximum value ($E_m$) of the sinusoidal waveform. All currents and voltages are stated in RMS values, unless specified otherwise.
b. At the lowest phase line-to-neutral voltage \(E_{\text{min}}\) and the highest heater resistance \(R_{\text{max}}\), the power density, \(P_{d_{\text{max}}}\), would be:

\[
P_{d_{\text{min}}} = \frac{(206)^2}{(25.57)(510.32)} = 3.25 \text{ watts/in.}^2
\]

7-803.1.3 Power Conductor Size

The maximum RMS load current that the heater conductor is required to conduct occurs when the "line" voltage (phase-to-neutral voltage) is the highest, with the heater resistance at the lowest manufacturing tolerance limit. The heater is essentially resistive; hence, the load power factor is assumed to be unity. The total conductor length is expected to be approximately 12.5 feet. The conductor gauge size should be selected to handle the maximum load current, with negligible temperature rise and voltage drop. Therefore, the highest RMS load current, \(i_{\text{max}}\), is:

\[
i_{\text{max}} = \frac{e_{\text{max}}}{R_{\text{min}}} = \frac{242}{20.93} = 11.56 \text{ amperes}
\]  

Allowing a 20 percent conduction capability as a design margin, the power conductors would be No. 16 AWG per MIL-W-81381/12. The conductors for the windshield temperature sensors which are required to conduct no more than 10 milliamperes of direct current signal, should be not less than No. 24 AWG per MIL-W-81381/12. The No. 24 AWG size for the sensor circuit is an over-design electrically, but it is necessary for mechanical strength.
The single-ended, three-sector design serves two important functions, thereby satisfying two important design criteria: (1) The bus-to-bus resistances of all sectors are essentially equal. This presents approximately an equal load demand to each of the three phases of the ac bus, thereby avoiding potential phase load unbalances; (2) The single-ended neutral/ground referenced design offers the advantage of maintaining a low impedance for P-static and EMP-related or lightning-related transients when the system grounding policy permits a structure ground at the windshield frame.
7-900 REFERENCES


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