WADC TECHNICAL REPORT 57-370
ASTIA DOCUMENT NO. AD 155846

AN ENGINEERING STUDY OF AIRCRAFT CRASH-FIRE PREVENTION

R. B. JONES, R. G. DIQUATTRO, AND R. G. TOWLE

*WALTER KIDDE & COMPANY, INC.,
BELLEVILLE 9, N. J.*

JUNE 1958

AERONAUTICAL ACCESSORIES LABORATORY
CONTRACT NO. AF 33(616)-2246
PROJECT NO. 6075
TASK NO. 61340

WRIGHT AIR DEVELOPMENT CENTER
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

McGregor & Werner Midwest Corp., Dayton, O.
13-336 & 337-400-1-5-59

FOREWORD

This report was prepared for the Aeronautical Accessories Laboratory, Wright Air Development Center, Air Research and Development Command, United States Air Force, Wright-Patterson Air Force Base, Ohio, by the Research and Development Department of Walter Kidde & Company, Inc., Belleville 9, New Jersey, under Contract No. AF 33(616)-2246, Project No. 6075, Task No. 61340, "An Engineering Study of Aircraft Crash-Fire Prevention". The position of technical administrator and project engineer for Wright Air Development Center was successively occupied by Messrs. R. A. Dickson, R. Stasiak, and F. W. Thompson, Jr., under the supervision of Mr. H. Klein, Section Chief, Aircraft Fire Protection Section, Mechanical Branch. At Walter Kidde & Company, Inc., the project was administered by Mr. E. Zeek, Assistant Vice-President and Manager, Research and Development Department. Mr. R. G. Towle served as project engineer during the early stages of the program; subsequent project engineers were Mr. R. B. Jones and Mr. R. G. Diquattro. Liaison with WADC was maintained through Mr. D. R. Squier, Manager, Flight Safety Sales, Aviation Division, and Mr. D. E. Dean, Aviation Division District Sales Manager, Dayton, Ohio. Mr. F. Kulima provided technical guidance during the installation of the crash-fire prevention system equipment in the aircraft at WADC. Acknowledgement is made of the valuable efforts of the numerous personnel of the Walter Kidde crash-fire group which permitted the demonstration of the feasibility of providing an airworthy and reliable crash-fire prevention system for aircraft.

Acknowledgement is also made of the cooperative efforts of Mr. I. I. Pinkel and his staff at the NACA Lewis Flight Propulsion Laboratory, and of the valuable contributions of the Experimental Fabrication Division of the Directorate of Support and the Cargo Operations Branch, Cargo Maintenance Branch, and Test Engineering Division of the Directorate of Flight and All-Weather Testing during the installation and flight test phases. First Lieutenant Joel Tumarkin and Mr. Jesse Martin of the Test Engineering Division were most helpful during the flight test phase. The Fairchild Engine and Airplane Company rendered valuable assistance on questions relating to the C-119 aircraft which were used as the flight test vehicles.

This report was prepared by Mr. H. Dalalian under the guidance and supervision of Mr. R. B. Jones and Mr. R. G. Diquattro. The project was started in September 1953. This report was completed on June 27, 1958.

ABSTRACT

Utilizing principles previously established by the National Advisory Committee for Aeronautics and the Walter Kidde & Company, Inc., an aircraft crash-fire prevention system was designed, developed and flight tested. The object of the program was to demonstrate, through the use of pre-production type equipment, the feasibility of providing an airworthy and reliable aircraft crash-fire prevention system. A transport type airplane, the USAF C-119G, was used as test medium for the system. The system was designed to eliminate, in the event of a survivable type crash, the potentially dangerous crash-fire hazards usually found in an airplane. This was accomplished by providing for the inerting of all possible sources of ignition, as well as the shutoff of combustibles in nacelles and de-energizing of the airplane's electrical system. The system also included crash damage sensing devices coordinated by a control unit to automatically initiate the inerting and shutoff equipment in the event of crash. Circuitry for semi-automatic checkout of the system was incorporated in the control unit. The flight tests which concluded the program, demonstrated that an aircraft crash-fire prevention system can be made airworthy and reliable.

PUBLICATION REVIEW

The publication of this report does not constitute approval by the Air Force of the findings or conclusions contained herein. It is published only for the exchange and stimulation of ideas.

FOR THE COMMANDER:

Frederick W. Thompson, Jr.

FREDERICK W. THOMPSON, Jr.
Project Engineer
Aircraft Fire Protection Section
Mechanical Branch
Aeronautical Accessories Laboratory

SUMMARY

This summary is intended for the reader who is interested in only the major details of the crash-fire prevention equipment, and its installation and flight test on a C-119 airplane, and who wishes a general review of the entire crash-fire prevention research, development and test program which culminated in these flight tests. For this reader, therefore, considerable background material on the NACA research and the C-82 prototype equipment and test crash is included which is not found in the report itself but is contained in the referenced literature.

Survivable-type airplane crashes, that is, crashes in which the occupants might survive the impact, are frequently followed by fire. A practical way of preventing such fires has long been needed, but the means have been unavailable until, in recent research with full scale crash tests, NACA succeeded in identifying the origins of crash fires and devised experimental means for preventing them. Fuel, spilled from ruptured fuel tanks, is atomized and spread by the crashed airplane's deceleration until it is ignited by contact with an ignition source. The ignition sources were found to be primarily engine torching and backfiring, hot engine exhaust systems, and sparks from the airplane's electrical system. Small oil or dripping fuel fires, ignited from these ignition sources, could also act as intermediate ignition sources themselves to ignite the cloud of atomized fuel. The crash fires could be prevented by controlling the spillage of combustibles in places close to ignition sources such as in nacelles, and inerting the ignition sources before contact could be made with them by the atomized fuel.

Inerting, NACA found, could be accomplished by relatively simple but effective means. Engine torching and backfiring could be stopped by quickly shutting off the fuel supply and diluting the mixture within the engine with an agent such as CO_2. Water sprays would cool hot surfaces below fuel ignition temperatures, the steam generated serving as an interim protective blanket. Electrical arcs could be prevented by disconnecting and grounding the airplane's electrical system. Combustible spillage into the nacelles could be reduced by rapid fuel and oil shutoff at the nacelle firewall.

Speed was all-important. Fuel from ripped tanks could be expected to reach ignition sources in the nacelles in a matter of seconds, spilled combustibles in the nacelles even sooner. NACA's experimental inerting devices were fast - electrical shutoff was accomplished in .1 second, cooling of hot exhaust system surfaces started in .25 second, and the engine mixture was diluted with CO_2 in .34 second - times found fast enough to beat ignition.

A contract was awarded to Walter Kidde & Company, Inc. to produce proto-

type equipment in an integrated system for a test crash to be conducted by NACA. The equipment was to provide inerting and fuel and electrical shutoff as performed experimentally by NACA. In addition, recognizing that human reaction would be too slow, equipment was to be provided for automatically initiating the inerting and shutoff devices.

The prototype equipment was produced and installed in a C-82 airplane for the test crash. The engine exhaust system was cooled by spray nozzles fed from pressurized tanks controlled by cartridge-operated valves. Three and one-half gallons of water, with metallic salts added to depress the freezing point, cooled each engine. Three pounds of CO_2 in a standard fire extinguisher bottle diluted the engine mixture. Special high speed shutoff valves were developed to shut off fuel and oil at the nacelle firewalls, and fuel at the engine intake. Electrical shutoff was accomplished with standard aircraft devices.

The initiating system devised for this test crash consisted of four crash sensing elements: (1) inertia switches mounted in the fuselage and nacelles which would be operated by crash deceleration, (2) engine mount reaction switches which would sense unusual movement of the engine caused by the propellers striking an obstacle, (3) cable-operated switches in the wings which would detect penetration of the wing by a pole or tree, and (4) strip switches along the belly of the fuselage which would detect deformation of the fuselage or would be operated by the weight of the airplane on them. These devices, connected in four circuits, would automatically trigger the inerting system when crash conditions set off any two circuits.

The C-82 airplane carrying this prototype inerting and initiating equipment was test crashed by NACA in July, 1953 and it was successful in preventing the occurrence of a crash fire. The results of previous experimental crashes indicated that the crash conditions imposed always resulted in fire. The fact that fire did not occur during this crash indicated that the crash-fire inerting system functioned satisfactorily as a complete unit. The prototype equipment functioned with a rapidity equal to or greater than that of the experimental systems used by NACA in their studies.

The successful crash test, with prototype inerting and automatic initiating systems preventing the usual crash fire, demonstrated that means were now available to prevent aircraft crash fires. The means, represented by the prototype equipment, could reasonably be carried by the airplane and could be depended upon to prevent the fires if it operated reliably. The crash test demonstrated also that the equipment could be subjected to the rigors of a crash and still perform its function.

To be proven was whether such equipment could be depended upon not to malfunction during flight, where its possible inadvertent operation could cause damage to or even loss of the airplane. Such equipment would have to be built to withstand aircraft environmental conditions of vibration, temperature extremes, humidity, etc., and still operate dependably. This was Kidde's next assignment - a contract with the Air Force to build crash-fire prevention equipment, based on the prototypes, for flight test in a C-119F airplane. The equipment, designed for the requirements of that airplane, was to satisfy the requirements of the environmental test specification MIL-E-5272A. This is the equipment and the flight test that is the subject of this report.

With Wright R-3350 turbo-compound engines in the C-119F, the power recovery turbines presented a somewhat new problem in exhaust system cooling. NACA crash-fire research with jet aircraft, however, indicated that turbines could be cooled by spraying with water without any damage to the turbine. The power recovery turbines were, therefore, cooled with internal sprays. The prototype exhaust cooling system of using spray nozzles trained on the exhaust stacks were not entirely satisfactory from a practical standpoint, so a new and better method of cooling the hot exhaust system was developed in which the surfaces were covered with fine mesh screening and aerated coolant distributed under pressure through perforated tubes attached directly to the screened surfaces. The coolant tanks were designed to fit around the engine close to the exhaust system. Pressurization was supplied by a separate high pressure air tank also attached to the engine. Discharge of coolant was controlled by a squib-operated valve in the air tank and burst disc valves in the coolant tanks. A CO_2 bottle was attached to the engine at the rear case and the carbon dioxide introduced through ports in the induction casing, again using a squib-operated valve.

Fuel shutoff at the engine was accomplished by modifying the spinner discharge valve (a valve located at the point of entrance of fuel to the blower section and held open by the fuel pressure) so that it could be driven closed by the high pressure air from the exhaust cooling system. Firewall shutoff valves for fuel, oil, and hydraulic fluid were redesigned to achieve the necessary reliability and to conform, as far as crash-fire prevention considerations would permit, to the fuel shutoff valve specification MIL-V-8608.

Combustion-type cabin and de-icing air heaters required hot surface cooling, accomplished by internal and external coolant sprays.

The auxiliary power plant, normally operating at take-off and landing, required mixture inerting with carbon dioxide and hot surface inerting. Here the hot surface, being small, was effectively inerted by its wrapping of asbestos tape.

Electrical shutoff was accomplished with relay trip circuit breakers to release the battery contactor and interrupt the main engine and APP generator fields. The bus was grounded by a contactor mechanically interlocked with the battery contactor so that both contactors could not be closed at the same time.

The initiating system was extensively revised from that used on the C-82 test crash to reduce the possibility of inadvertent actuation of the system in flight. The strip switches on the belly of the fuselage were connected into a controlling circuit so that the complete inerting system could be tripped only if two of the three fuselage switches were actuated simultaneously, a condition that could be met only by the fuselage coming in contact with the ground. The engine reaction switches and the wing deformation switches were connected in circuits in such a way that actuation of both types on one side of the airplane would produce instant inerting of the nacelle on that side, actuation of either one would provide an arming signal that, in conjunction with the fuselage switch circuit, would inert the entire airplane. This arrangement, worked out by NACA, the Air Force and Kidde in collaboration, provided for immediate local inerting of ignition sources in areas where severe damage had been sustained and fire was imminent but would not permit total inerting with total loss of power until it was positively established that the aircraft was no longer airborne. Inertia switches were eliminated pending further study because NACA research indicated that crash deceleration could be so low as to require inertia switch settings that would make them susceptible to inadvertent operation in turbulent flight.

A checkout circuit was provided for preflight checking of the system including checking, electrically, the condition of the various initiators and inerting devices and, functionally, the operation of all the circuits. The preflight test gave assurance that crash-fire protection was provided and that no condition was present that would give inadvertent operation.

Numerous safety features were incorporated in the system. One for example, was in the pneumatic mechanism that closes the spinner discharge fuel valve. Provision was made for the closing pressure to be released after a period sufficient to stop the engine so that the valve would reopen, and, if shutdown had been inadvertent, the engine could be restarted. Another safety feature was in the bus grounding contactor being mechanically interlocked with the battery contactor so that the bus cannot be grounded until the battery is disconnected. Another important safety feature was the routing through separate connectors of all wires to the control unit that could, by coming in contact with each other, operate the inerting system.

A control panel was provided for the pilot that contained an indicator light and switch for each initiator circuit. The switch permitted the pilot to cut out any crash-sensing initiator circuit whose indicator light warned of malfunction in flight. Also provided on this panel was a manually-operated crash switch to permit the pilot to actuate the crash-fire prevention system in anticipation of a crash.

The equipment as described was designed and fabricated. It was put through, successfully, the qualifying tests of the standard Air Force environmental conditions as required by MIL-E-5272A. In the latter half of 1957, the equipment was installed in two C-119F airplanes and flight tested by WADC under varying flight conditions. The equipment was installed in its normal aircraft environment, but in such a manner that if any of it inadvertently operated, the aircraft's normal operation would not be affected. For example, the crash sensing devices were connected to a monitor panel rather than to their regular control box so that their operation would be indicated but could not trigger any part of the system. The firewall shutoff valves were paralleled with the airplane's existing firewall valves so that inadvertent closing could not affect the airplane, but a connection to the monitor panel indicated the position of the valves, either open or closed. The engine CO_2 system was dead-ended in a pressure transducer which signalled the monitor panel. Similarly, the compressed air which operated the engine exhaust cooling system and the engine fuel shutoff valve, the CO_2 bottle for APP inerting, and the coolant tank for the combustion heaters were dead-ended in transducers that signalled inadvertent operation to the monitor panel, and thus could not actually perform their function. Each test flight was accompanied by a flight test engineer who made a preflight checkout of the equipment using the semi-automatic checkout circuit and who observed the monitor panel during flight to detect any malfunctions or inadvertent operations. A total of 282 hours of test flight time was accumulated.

Difficulty with some of the equipment was experienced during the flight tests and was cleared up on all but two items. During the first test flights the engine reaction switches tripped on normal take-offs and landings. Investigations revealed that, of the two switches installed on each engine, the bottom one was being subjected to relative engine motion greater than allowed by the setting. The affected lower switches were reworked and reset and no further trouble was encountered.

On two occasions during the flight test, one of the firewall shutoff valves failed to operate properly. On each occasion the trouble was traced to faulty electrical connections.

The coolant tank burst discs, designed to burst and release the coolant when pressurized from the compressed air source, failed by bursting under flight imposed impact loads without pressurization. The coolant tanks themselves were strapped to brackets and attached to the engine using available studs on the rocker arm boxes. The straps proved to be too flexible, allowing the tanks to move under engine vibration and eventually wear the tank wall so thin as to produce a failure. The brackets and studs also proved inadequate, breaking under the stresses imposed by the loosely attached tanks. The stud failures made it necessary to remove the tanks from the airplane so that flights could continue. As a result, corrections could not be made and tested for either the brackets or the burst discs. However, the failures indicate the nature of the fixes required.

For the burst discs, the flight impact loads are too close to the pressurization force so that fatigue failures result. The cure is either to reduce the impact loads by impact absorbing devices or revert to a different type of valve operation such as a piston-supported burst disc, a cartridge-operated frangible disc valve, or a lock open type poppet valve.

The tank wear problem can be corrected by interposing a wear absorbing pad between tank and bracket, by using a more rigid support bracket to reduce movement, or by using a flexible mounting to absorb the movement. The stud and bracket problem can be cured in the case of this engine either by spreading the load over more studs and brackets, using a more rigid tank mounting to reduce load concentration on a single stud or bracket, or by providing stronger points of attachment. Should the latter prove to be necessary, the engine would require modification to provide the required support.

Thus, of the troubles experienced in the flight test only the coolant tank burst disc failures and the coolant tank mounting remain to be cured. In both of these cases the cure is apparent and the troubles can be solved with ordinary application engineering, not development.

After the necessary readjustment of the lower engine reaction switches, all the initiating devices withstood the flight tests satisfactorily. The control unit, monitored for any signs of a signal output that would have caused the actuation of any inerting device had it been connected, performed normally including its checkout circuit which was exercised for each flight. Inadvertent operation of the firewall shutoff valves during flight was a fault, not of the component but rather of the installation wiring.

Since only one component, the engine exhaust coolant tank, showed an uncorrected deficiency, but because equipment with comparable features - the engine CO_2 bottle mounting, and the combustion heater coolant tank valve - withstood comparable flight conditions, it can be stated that the crash-fire prevention equipment withstood the flight tests satisfactorily. As the purpose of the flight test was to demonstrate the feasibility of providing airworthy and reliable crash-fire prevention equipment for aircraft, it can therefore be concluded that the demonstration was successful.

Further, the research completed on jet-engined aircraft shows that the same principles of inerting apply, so that similar equipment with similar problems would be used. All the engineering knowledge that is needed to provide crash-fire prevention for any piston engined, prop-jet or turbo-jet aircraft is now available as a result of the program of research, development and flight test just concluded.

TABLE OF CONTENTS

TABLE OF CONTENTS (continued)

TABLE OF CONTENTS (continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (continued)

LIST OF ILLUSTRATIONS (continued)

LIST OF ILLUSTRATIONS (continued)

I. INTRODUCTION

The entrance of Walter Kidde & Company, Inc. into the field of aircraft crash-fire prevention dates back almost to the inception of that phase of aircraft safety research. Following the resolution of the mechanisms of start and the means of prevention of crash fires by the National Advisory Committee for Aeronautics (Reference 1), the United States Air Force recognized the merits of providing crash-fire prevention systems for aircraft. As a result, Walter Kidde & Company, Inc. was awarded a contract (Reference 2) by USAF to study the feasibility of integrating crash-fire prevention equipment into an automatic system. This system was to include prototype inerting, cooling, and shutoff equipment and automatic crash-sensing devices for the C-82 airplane, the type of aircraft being used by the NACA in their full-scale testing at the time.

Working in close cooperation with USAF and NACA, and utilizing data from the numerous NACA reports on the subject of aircraft crash fires, this project was concluded by a full-scale crash test in July 1953 in which the occurrence of fire, where fire otherwise would have occurred, was prevented by the prototype equipment, and which demonstrated the effectiveness of crash-fire prevention systems for aircraft. A final report (Reference 3) on this work was submitted to the Air Force by Walter Kidde & Company, Inc. under USAF contract No. AF 18(600)-167. In addition, the results of the crash test and a report on the performance of the equipment are presented in NACA RM E55B11 (Reference 4).

Following the conclusion of this program, the USAF awarded Walter Kidde & Company, Inc. a second contract,No. AF 33(616)-2246 (Reference 5), calling for the development and flight test of pre-production type equipment to determine the feasibility of providing an airworthy and reliable aircraft crash-fire prevention system, which is the subject of this report. This equipment was to be developed using USAF C-119F type airplanes (Figure 1) as the test vehicle.

Specifically, the contract called for the following:

1. An engineering study of the C-119F airplane, consisting of a complete analysis of the aircraft to determine all possible hazards insofar as crash-fire prevention was concerned.

2. A design analysis of the crash-fire prevention system on paper prior to the intensive study of the aircraft.

3. Special attention to the combustion heaters on the C-119F airplane.

4. Engineering design of a sub-system to handle each hazard as it was uncovered and understood; the characteristics and performance of the equipment in each sub-section were to be established and the final design characteristics worked out.

5. Integration of the various sub-systems after the above was accomplished so that a complete fire prevention system was established and design requirements determined.

(Manuscript released by authors 18 Sepbember 1958 for publication as a WADC Technical Report)

6. Consideration of the various pieces of prototype equipment from the C-82 tests as a starting point in development of the pre-production equipment for the C-119F aircraft. The pre-production equipment developed had to meet the performance requirements of the C-119F, the environmental conditions, and the applicable procedures of MIL-E-5272A.

7. Three complete sets of hand-made pre-production type equipment were to be fabricated and assembled. (Subsequent revision of the contract required only two sets).

8. The complete crash-fire prevention systems were to be flight tested and a final report was to be issed in conclusion of the overall program.

According to the original thinking on the system, inertia switches were to be installed as part of the initiating sub-system, wing cable switches were to be installed only outboard of the engines, and complete inerting of the entire airplane would have been the only possible mode of operation. As development proceeded and more knowledge was gained on the subject, it was mutually agreed upon by USAF, NACA and Walter Kidde & Company, Inc. that the inertia switches (Appendix D) would be omitted so as to eliminate the possibility of inadvertent operation, that the cable switches would be installed both inboard and outboard of the engines, and that the additional capability of local inerting would be included in the system.

During the final stages of development, the flight testing phase of the program was amplified. The contract specified the use of a monitoring system to check the functioning of individual crash-fire prevention system components during the flight test, and thus detect any malfunction or inadvertent operation of the equipment during flight. The various components of the system were to be installed in the aircraft in such a manner that malfunction or inadvertent operation of any one or several of them would not in any way affect the normal operation of the aircraft. This preventive type of installation was to be accomplished by the use of dummy lines and circuits which would simulate the actual circuits and permit the components to be monitored for satisfactory operation under true flight conditions. The system was to be flight tested on two C-119F aircraft for a total of approximately 300 hours. Upon completion of the flight tests, the equipment was to be removed from the aircraft and thoroughly inspected.

II. ANALYTICAL STUDY OF CRASH-FIRE HAZARDS IN THE C-119F AIRPLANE

The C-119F airplane, for which the crash-fire prevention system was to be designed, was a high wing, twin engine cargo airplane, successor to and very similar to the C-82 "Flying Box Car". The major differences were the lengthened fuselage, some changes in aerodynamic surfaces, and the substitution of more powerful engines, Wright R-3350-85 turbo-compounds. The C-119F contained the same hazards that were previously identified in NACA studies and, with two additions, were identical to those encountered in the C-82 crash test program. These hazards may be sub-divided into two categories, combustibles and ignition sources.

Fuel, engine oil, and hydraulic oil were the combustibles present in the subject aircraft just as they generally are in most aircraft. Control of fuel and oil spillage, insofar as possible, must be considered in the design of an efficient crash-fire prevention system.

Regarding ignition sources, the R-3350-85 engines of the C-119F possessed the same basic hazards as did the R-2800 engines of the C-82. These consisted of the hot metal surfaces of the engine exhaust systems, and engine exhaust and induction system flames. In spite of the fact that the hazards presented by the two types of engines were the same, the engines themselves were different. The R-3350-85 turbo-compound engines of the C-119F had a different exhaust system and, in addition, included three power recovery turbines which further complicated the problem. Because of these dissimilarities, a somewhat different treatment was required for the R-3350-85 engines than for the R-2800 engines.

A source of ignition not encountered in the C-82 crash-fire studies was constituted by the airplane's combustion type heaters. Eight internal combustion heaters, installed in a group within the fuselage of the airplane above the auxiliary floor level and aft of the crew's compartment, provided hot air heating for cold weather operation of the aircraft. Heat is generated by the induction and ignition of fuel in the heater's combustion chamber. A temperature survey indicated that the heater inner section surfaces become hot enough to ignite a combustible mixture.

Still another ignition source not involved in previous crash-fire studies was the auxiliary power plant. This unit, an internal combustion engine-generator combination, provides electrical power independent of the airplane battery for starting the engines, ground checking of various electrical equipment, and augmenting the engine generators during takeoffs and landings. The problems relating to the auxiliary power plant are similar to those presented by the airplane's engines; i.e., hot exhaust stacks, and the danger of backfires and torching.

As in the C-82, the C-119's electrical system provided a source of ignition. Severance of wing wiring, windmilling of generator rotors, or re-orientation of or damage to the batteries during a crash could cause electrical ignition of crash-spilled combustibles.

Other prominent ignition sources present were incandescent light filaments, particularly those of landing lights because of their position directly in front of fuel cells, and friction and electrostatic sparks possibly generated during a crash. These sources of ignition had been identified in NACA studies, but no means for inerting them had been developed during the C-82 program, nor was an attempt made to inert them in the C-119. This was so because they were considered to be ignition sources incapable of being inerted in aircraft of existing design.

III. DEVELOPMENT OF CRASH-FIRE PREVENTION SYSTEM FOR THE C-119F

Crash-fire studies conducted by NACA (Reference 1) have revealed several factors that produce the fire following a crash accident. Basically, they are as follows:

1. The combustibles contained in the aircraft are released either by rupture of the containers and lines due to the impact forces present or by actual physical damage caused by external objects such as poles, trees, etc. in the crash path.

2. The released combustibles atomize into the surrounding atmosphere and, by virtue of the high decelerations that are present during a crash, spread forward and engulf a major part of the aircraft.

3. The combustible mist contacts an ignition source resulting in fire. Fire can be prevented if either the combustibles can be prevented from reaching the ignition sources or the ignition sources can be removed or inerted.

As previously mentioned, the ignition sources in aircraft have been identified as the hot surfaces of engines and heaters, engine exhaust and induction flames, incandescent lamp filaments, electric arcs, and friction and electrostatic sparks. The dangers inherent in the presence of most of these ignition sources may be nullified by the introduction of a crash-fire prevention system. Such a system is designed to inert the sources of ignition during an otherwise survivable crash and thus prevent the occurrence of fire. A crash-fire prevention system should consist of the following:

1. An initiating system which can sense when the aircraft has suffered damage severe enough to result in fire and which transmits this information to --

2. A control system which receives the crash-damage signals, interprets them as to extent of damage, and actuates appropriate portions or all of the --

3. Inerting and shutoff systems which inert the various ignition sources, shut off fuel, oil and hydraulic fluid at the nacelle to control spillage, and shut off electrical power.

In providing an integrated system of pre-production type crash-fire prevention equipment for testing in the C-119, it was proposed to take the following actions in treating the hazards discussed in Section II above:

1. Inject CO_2 into the engine manifold and inert the combustible mixture in the engine induction system (to prevent fire in the event of break-up of the engine casing or detachment of the carburetor).

The CO_2 system is to be mounted directly on the engine in such a position that, in the event the engine is torn from its mount, the CO_2 system will remain with the engine and perform its function.

2. Cool the engine exhaust system to a temperature below the spontaneous ignition temperature of any combustible present; in this case 500°F was chosen because of the lubricating oil. Cooling is accomplished by directing a flow of coolant, such as water, against the hot surfaces to extract the heat from the metal by vaporization of the coolant.

   The coolant system, both supply and distribution, should be so attached that, in the event of engine separation or break-up, the coolant system will remain with the hot metal and continue to cool it. In the case of the R-3350 turbo-compound engine, the exhaust collector system is so divided by the three power recovery turbines that break-up would result in many separate parts of hot metal, none large enough to warrant its own complete coolant system. However, this turbine arrangement also made it less likely that exhaust parts would separate from the engine. It was decided, therefore, that the coolant system would be attached to the engine and would, in the event of engine separation, continue to cool those hot exhaust parts that remained with the engine.

3. Close fuel, oil, and hydraulic lines at the engine firewall.

4. Shut off fuel as close as possible to the point of induction into the engine to shut down the engine as rapidly as possible. Because there was no external fuel line from the carburetor to the intake, a valve could not be interposed as in the C-82; however, there was an existing internal valve between carburetor and intake which could readily be modified to be closed by the crash-fire prevention system.

5. Shut down the auxiliary power plant, and prevent contact of spilled combustibles with its hot exhaust system by covering the exhaust system with insulation.

6. Inert the combustion heaters by closing the duct valves to prevent combustibles from entering and contacting hot heater surfaces. Subsequently, it was decided that this was impracticable and unreliable, and that cooling of the heaters would be a more satisfactory means of inerting them.

7. Inert the electrical system as in the C-82 -- note that provisions had to be made for disconnecting power from the crash-fire equipment after actuation. (In the C-82, a motor driven switch was employed for this purpose; the motor was powered from the battery and operated continuously. The timing switch was closed as a result of the energizing of a magnetic clutch by means of a crash impulse).

The various electrical and electro-mechanical components of the subject crash-fire prevention system were designed to operate satisfactorily under a minimum applied voltage of 12 volts DC although the nominal system voltage was to be 28 volts DC. It was considered desirable to do this because the C-82 crash test indicated the possibility of considerable voltage variation during a crash as a result of momentary shorting. To insure reliable operation of the system components and therefore of the system, 12 volts DC was selected as the low voltage limit on the basis of the C-82 experience.

The crash-fire prevention system described in this report possesses the additional feature of being able to provide local inerting of either engine nacelle in case in-flight damage is sustained, and overall inerting of the airplane once ground contact is made and engine power is no longer required. A discussion of the various components of the initiating, control, and inerting systems and how they were all integrated into an effective crash-fire prevention system is presented in the following paragraphs.

A. Initiating System

The automatic initiating system used in the C-82 test crash functioned well. Because the C-119 airplane was so similar to the C-82, it was planned to use the same system, making only certain minor modifications in the circuitry which would improve the reliability. In that system, the actuation of any two of the four initiating circuits within a ten-second interval would cause the inerting and shutoff systems to operate and inert the entire plane.

Considerable attention was given to inflight reliability of this system, particularly to the concept that no single failure of a component or a wire should cause the inerting and shutoff systems to operate while the airplane is in flight. The basic premise that two of four initiator circuits had to be actuated within ten seconds of each other gave a certain measure of safety in flight. As an additional precaution, an indicator panel was to be provided that would indicate to the pilot when any initiator device tripped. While in flight, such tripping would be presumed to be inadvertent and switches would be provided by which the pilot could switch the offending device out of the circuit. Even with these precautions, there was still a possibility of progressive failures by which one initiator circuit could be tripped and not be detected until a maintenance check, meanwhile leaving the airplane vulnerable to the inadvertent tripping of a second initiator circuit which would trigger the inerting and shutoff systems and cause immediate loss of power. It became apparent, therefore, that some provision for check-out of the initiator system before flight would be desirable. Study indicated that check-out of the electrical integrity, not only the initiator circuits, but of the entire system was feasible and could be made "push-button" automatic so as to relieve the pilot or flight engineer from needing a detailed understanding of the system and circuitry.

Later on, a unique method of detecting incipient short circuits due to chafing of wires was conceived. Its value in detecting conditions which could cause inadvertent operation or malfunction in circuits to critical devices was immediately apparent. The circuitry required was not excessive, and the need for special shielded wire was offset by the fact that armored wire would probably be desirable anyway for those devices that controlled shut-down of the engine, and the armored wire would not be as effective as the special shielded wire. Accordingly, provision was made in the control circuitry for incorporation of the short-anticipator system.

The possibility of triggering the crash-fire prevention system inadvertently while the airplane was in flight seemed remote with these safety precautions. Nevertheless, knowing that inadvertent operation in flight would result in total loss of power and control created grave doubts of the advisability of an arrangement that would permit such a possibility. A particular possibility would be for an aircraft to suffer damage by contact with trees while airborne at take-off or on landing approach. It is conceivable that such damage could be light enough to allow the airplane to continue in flight but severe enough to actuate the crash-sensing devices. Early in 1955, NACA proposed an initiating system arrangement that would allow for such in-flight damage. Basically, the concept was that localized damage to the aircraft while in flight would result in immediate localized inerting in the damaged area to prevent fire only in that area where it was imminent, while inerting protection for the entire airplane would be provided when it reached the ground. NACA suggested use of additional sensing devices (References 6 and 7) to detect break-up of the propeller reduction gear housing and to detect loss of the landing gear, and a modification of the engine reaction switch to sense engine breakaway rather than unusual movement. It was also suggested by NACA that a device be installed in each wing, actuated by the pull cable of the wing deformation switch, to disconnect the electrical wiring in that wing in the event the wing is penetrated. To further guard against fire starting from severed electrical wiring, NACA suggested that the wiring be encased in a flexible sheath and re-routed aft of the rear spar.

Consideration of these suggestions by WADC and Walter Kidde & Company, Inc. led to the adoption of a compromise initiating system for installation in the C-119 airplane for the flight tests. Development of initiating system components had progressed nearly to completion and it was realized that development of new devices would hold up the program excessively. Break-up of the propeller reduction gear housing was less likely with the Wright R-3350 engine because of its different design than with the R-2800 engine used by NACA in its test crashes. It was thought that the deformation switches on the belly of the low slung C-119 fuselage would adequately detect conditions that would result from torn-away landing gear. The concept of detecting local damage and providing localized inerting until ground contact was established could be achieved using the equipment already developed. While the crash protection would perhaps not be the ultimate envisioned by NACA, it would be as adequate as that originally planned. Furthermore, the purpose of the flight test program was to test reliability of the equipment against inadvertent operation in flight, and this purpose would not be served any better by the recommended changes.

The compromise initiating system (Figure 2) devised for the C-119 flight tests by the joint efforts of WADC, NACA, and Walter Kidde & Company, Inc., functions as follows:

(a) Cable-operated deformation switches were to be strung singly through the wing in front of the fuel cells rather than doubly as previously planned.

(b) Engine reaction switches were to be mounted on the engine mounts to detect unusual movement of the engines as originally planned.

Operation of either one of these two types of devices would provide an arming signal to the initiating system control box. Operation of the remaining one of the two types of devices on the <u>same side</u> of the airplane would trigger the inerting system for the nacelle on that side, and would shut off electrical power in that wing. To guard against loss of the crash sensing signal as a result of severance of the signal-carrying wires, a ten-second memory circuit was provided for each of these two types of switches.

(c) Three fuselage deformation switches were to be mounted on the belly of the fuselage, one located centrally up forward, and one on each side at the rear to sense contact of the fuselage with the ground.

Simultaneous operation of any two of the three fuselage switches for only an instant was required to signal the fact that ground contact had been established. This signal to the control box, in conjunction with a previous arming signal, triggered the entire airplane inerting system. Provision was also made for the reverse procedure - that is, the ground contact signal would be held for ten seconds and if the damage signal from a wing or nacelle was received in that period, the entire inerting system was triggered.

To guard against a false ground contact signal resulting from a brush with tree tops, for example, the fuselage deformation switches were utilized only as momentary contact devices and the individual circuit holding relays originally planned for these switches were eliminated. It was thought unlikely that any contact with the belly of the fuselage during flight could simultaneously operate two of the three widely spaced switches even for the required instant. It was necessary, however, to provide a memory circuit for the ground contact signal in order to insure provision of a signal of sufficient strength and duration to reliably trigger the inerting system and to provide for the reversed sequence of operation previously noted. Thus, if two momentary contact switches were actuated in a simultaneous instant, the resulting signal was held for ten seconds.

The inertia switches (Appendix D) originally planned for the initiating system were eliminated in this revised concept. While the switch itself is an accurate and reliable device, the low settings necessary to detect survivable type crashes (1.5 g's) are low enough that the switch could be tripped by the impact of a hard landing or even possibly by severe turbulence in flight. For this reason the device was considered unreliable as a detector of crash conditions and was eliminated. A discussion of the initiating system components and their development follows:

1. Rubber-Strip Type Deformation Switch (Figure 3)

   Three rubber-strip type deformation switches were mounted on the belly of the aircraft fuselage (Figure 4). One was located on the centerline toward the fuselage nose (Figure 5), while the other two were situated on either side of the fuselage centerline toward the tail (Figure 6). These switch locations were selected after consideration of all the possible modes of ground contact which the subject aircraft might conceivably experience during a crash. Located in the selected positions, at least two of the three switches would probably function in a crash, regardless of the attitude of the fuselage, provided it was not inverted. The rubber-strip type deformation switch consists of three thin strips of phosphor bronze, 5/16 inch wide x 0.010 inch thick, separated by a 1/4 inch wide strip of silicone rubber spirally wound, with a pitch of 5/8 inch, around the center strip. Enclosing the sandwich thus formed, is an extruded sheath of Neoprene. One end of the sheath is hermetically sealed around a solid copper wire which connects the center bronze strip to a wire braid grounding strap. At the other end of the sheath, a rubber patch, previously sealed around a two-pin electrical connector, is fused into the sheath completing the hermetic seal. The two outer bronze strips and the two pins of the connector are connected in common. The two-pin connector, with the pins connected together within the sheath, makes it possible for the checkout circuit to check continuity of wiring through the connector into the switch and thus establish the integrity of the electrical connection as well as the wiring.

When pressure is applied to the deformation switch, the outer strips contact the center strip thereby completing an electrical circuit. These switches were designed to remain inoperative at deformation pressures below 150 psi. The permissible minimum local bend radius for these switches is approximately 6 inches, while the minimum coil diameter for transporting would be 3 feet or greater. This type of switch may be fabricated in any desired length, so that its use is not restricted to any one particular model of aircraft.

2. Cable Type Deformation Switch (Figures 7 and 8)

Two cable type deformation switches were installed in each wing of the airplane just forward of the main spar. One switch was located outboard of the engine nacelle (Figure 9) while the other was located inboard of the nacelle (Figure 10). Penetration of the wing by any obstacle to a depth sufficient to threaten damage to the main spar and subsequent rupture of the fuel cells will cause the switches to operate.

These switches consist, in part, of a length of stainless steel cable housed in an aluminum tube. In the case of the inboard switch, one end of the cable assembly is securely anchored to the main spar near the fuselage. The switch end of the assembly is rigidly attached to the main spar adjacent to the engine nacelle wall, as shown in Figure 11. The outboard switch is similarly mounted except that its anchor end is located approximately 12 to 18 inches beyond the last fuel cell (Figures 9, 12 and 13). At the switch end of these units, the cable is connected to a spring-loaded plunger; movement of the plunger in either direction along its centerline causes it to trip a microswitch.

The minimum plunger movement of ±1/2 inch required to operate the switch was determined on the basis of the distance between the cable and the main wing spar, and the minimum spacing between any two adjacent cable housing support points. This travel is sufficient to permit plunger movement as a result of normal wing flexure or thermal expansion without causing operation of the switch. Operation of the switch can be precipitated by pulling on the cable as might occur due to excessive wing deformation, by complete severance of the cable as may happen when an obstacle is encountered, or by shearing of a pin located at the point where the cable is attached to the plunger. This pin has an 80 pound shear load capacity. Flared fittings at the ends of the aluminum tubing insure that the unit is moisture-proof when completely assembled. Adjustment of cable tension is made in one simple operation within the switch housing at the time of installation.

3. Engine Reaction Switch (Figures 14 and 15)

Two reaction switches were mounted on each engine, one on the rear oil pump body cover plate on the lower end of the engine vertical centerline (Figures 16 and 17), and the other on the right side fuel injection drive gear housing substitute cover plate approximately 45° from the upper end of the vertical centerline (Figures 18, 19 and 20).

Located in these two positions, one or the other or both of the switches are effective for all possible modes of excessive movement of the engine in its mounts. The switches bear against reaction plates on the engine mount and are operated when severe movement of the engine occurs.

Similar to the cable type deformation switch, except for the absence of a cable, the engine reaction switch contains a lever arm-roller type microswitch. The roller at the end of this spring-loaded lever arm rides along a spring-loaded necked plunger. Normally, the roller rests in the neck of the plunger. Any longitudinal movement of the engine (relative to its mount) greater than that experienced during normal operation of the aircraft but short of bottoming will cause the switch to operate. Originally, all the reaction switches were set for an allowable plunger movement of ±0.125 inches before they would operate, this setting being based on stationary engine tests and engine mount data. However, the flight tests (See Section V) indicated this setting to be unsatisfactory for the lower reaction switches. After a study of the problem, the lower switches were set to operate whenever the plunger displacement caused by engine movement exceeded 0.200 inches forward or 0.130 inches rearward. This setting proved satisfactory in subsequent flight tests. The plunger and plunger spring are enclosed by a Neoprene dust cover; the switch itself is sealed against moisture by gaskets and "O" rings.

B. Inerting and Cooling (Actuation) Systems

1. Engine Exhaust Cooling

In an aircraft crash-fire situation, the hot surfaces of the engine exhaust system are the most prevalent sources of ignition. One of the prime requirements of a crash-fire prevention system, therefore, is an engine exhaust inerting system. Such a system should simultaneously inert and cool the hot surfaces of the engine exhaust assembly until they are cooled to below the spontaneous ignition temperature of the fuel-air and oil-air mixture.

The most direct method of accomplishing this desired inerting and cooling would appear to be the use of numerous, low-capacity, spray nozzles attached to a tubular ring circumscribing the exhaust system and directed at the hot surfaces. In spite of the fact that this method had been successfully employed experimentally at both NACA and Walter Kidde & Company, Inc. on the C-82 crash tests, the application of such a system to flight aircraft was considered undesirable and impractical for several reasons. The Pratt & Whitney R-2800 engine used in the crash tests utilized a collector ring type of exhaust disposal system. This type of exhaust system concentrated the hot metal surfaces in a relatively confined volume within the nacelle. The Wright R-3350-85 engine, in contrast, utilized a system of separate exhaust stacks, six stacks leading into each of the three power recovery turbines. This system distributed the hot metal surfaces within a much larger volume within the nacelle.

Spray nozzles, positioned to cover the hot surface areas, would have to be placed very close to the hot metal surfaces to avoid deflection of the coolant spray by the air flow within the nacelle. Their positioning about the exhaust surfaces was found to be very critical. Use of these nozzles would require an extensive network of coolant supply lines, supported at the proper distance from the exhaust stack surfaces.

In this arrangement of nozzles and supply lines, inadvertent displacement which could very likely occur during maintenance checks on the engine or as a result of vibratory loads imposed by flight conditions could reduce the effectiveness of the cooling system to the point where it might not prevent the occurrence of a crash fire. To optimize the quantity of coolant would require nozzles of varying sizes and spray configurations, such as solid cone, hollow cone, or fan shaped spray. In many instances, the spray nozzle orifice would have to be so small that the problem of clogging would arise. The resulting end product of such a system would be extremely complex. It would present problems in maintenance and would add considerable weight to the aircraft.

In order to be better able to develop a practical exhaust cooling system for the R-3350 turbo-compound engine, a team of Walter Kidde engineers was sent to the NACA Lewis Flight Propulsion Laboratory to make a detailed study of the possible method of de-energizing hot surface ignition sources for the purpose of arriving at a suitable solution to the problem of de-energizing the exhaust system of this engine. Small scale tests were conducted on each idea that appeared to hold promise. Both preventive type concepts and positive inerting and cooling methods were considered.

The study at NACA led to the selection of an aerated coolant sprayed onto screen-covered stack surfaces by means of simple orifices from distribution tubing attached directly to the stack surfaces. Covering the exhaust surfaces with wire mesh retains the coolant more effectively. By dimpling the wire mesh to form a waffle pattern, cross channels were provided adjacent to the metal surfaces which allowed the coolant to flow over the metal surfaces beneath the wire mesh, thus reducing the coolant spillage. Mixing the air with the coolant under pressure (aerating) prior to discharge improved its spreading qualities.

A mockup (Figure 21) consisting of exhaust stacks of the No. 1 turbine (3 o'clock position as viewed from the rear of the engine) from the R-3350-85 engine was set up to determine the following:

1. The actual quantity of coolant necessary to cool the exhaust disposal system of the R-3350-85 turbo-compound engine to safe limits.

2. The wire mesh which most efficiently retained the coolant and the weave and kind of material which would have the best durability.

3. The diameter, length, and best routing of the distribution tubing attached to the stacks to obtain the proper coolant coverage.

4. The diameter and location of the coolant spray orifices in the tubing to assure adequate distribution of coolant.

5. Ratio of the diameters of air and coolant metering orifices necessary to produce the most efficient aerated solution.

6. Optimum driving pressures of aerated coolant.

In order to determine the quantity of coolant required to cool the exhaust disposal system for this engine, it was necessary to know the quantity of heat that had to be removed from the hot exhaust system in order to render it inert. Thus a profile of maximum obtainable temperature of the exhaust system was required. A temperature survey was therefore conducted, the results of which revealed that the average maximum temperature was approximately 1000°F.

The temperature to which the exhaust stacks were to be cooled was selected as 500°F while the weight of the metal to be cooled was found to be approximately 258 pounds. Using the foregoing information, the heat content which was to be removed from the exhaust system was calculated and the quantity of coolant needed was determined to be 2-1/4 gallons. This quantity was doubled to 4-1/2 gallons to account for spillage and incomplete vaporization. Although the coolant developed under the previous contract (Reference 3) would be used eventually, water was used for the preliminary tests conducted on the engine exhaust system, since both the coolant and water had equal per volume cooling capacity.

In selecting the wire mesh that was most suitable from the standpoint of coolant retention qualities and durability, samples of different types and sizes were tack welded to the stacks of the mockup. The wire mesh finally chosen was 80 mesh waffled twill weave of 0.007 inch diameter stainless steel wire.

The size of distribution tubing was selected on the basis of the smallest practical size usable without incurring large pressure losses. With the possibility of engine re-orientation during a crash, it was necessary to insure coolant coverage even in the event of repositioning of the exhaust stacks. Because of this, the distribution tubing was originally wrapped spirally about each stack. This method was not very successful in that the spray from the distribution orifices located about the bottom side of a stack section would spill away, thus wasting a large quantity of coolant. It was necessary, therefore, to compromise coolant coverage during engine re-orientation in order to conserve coolant. This was done by wrapping the tubing only on the top side of normally horizontally positioned stacks and retaining the spiral concept on normally vertically positioned stacks. This arrangement, of course, meant that the exhaust stacks with distribution tubing attached were no longer interchangeable between the three turbine sections. From the cold flow coolant distribution tests conducted on the exhaust stack mockup, it was established that spiralling the tubing around the vertically positioned stacks with a pitch of approximately 4 inches and "snaking" the tubing across the tops of the horizontally positioned stacks, using an approximate pitch of 4 inches with an amplitude of 45° either side of the vertical centerline, gave adequate coolant distribution and provided for coolant distribution in the event the engine is partially re-oriented during a crash landing.

The size of the coolant distribution orifices were determined on the basis of the smallest size that experience had indicated as being acceptable from the standpoint of clogging. Their respective locations were experimentally selected to provide the optimum coverage of coolant. The number of orifices per stack section was determined on the basis of best coverage without sacrificing coolant flow toward the extremities of the distribution tubes.

The tubing eventually selected for coolant distribution was standard wall, 3/16 inch O.D., stainless steel. A drilled hole of 0.022 inch was selected as the minimum orifice size.

In determining the ratio of air and coolant metering orifices to provide proper aeration of coolant, several factors were considered. First, to optimize the quantity of coolant, the maximum rate of coolant discharge was limited by the maximum rate of heat transfer from stacks to coolant. Second, the air metering orifice size was selected on the basis of that size which yielded a sufficient pressure within the distribution tubing to provide adequate coolant coverage at the extremities of the tubes. Finally, the selection of the driving or upstream air pressure for metering both the coolant and air was based on obtaining optimum coolant coverage. The sizes of the orifices were established to be 0.070 inch diameter for air and 0.110 inch diameter for coolant. The best driving pressure was determined to be 300 psi.

The tests conducted on the modified mockup (Figure 22) consisted first of flowing the aerated mixture over cold stacks to observe the overall coverage. When the cold flow coverage was considered adequate, the stacks were heated by using "weed burners" to simulate engine exhaust stack temperatures. The "weed burners" were basically blow torches using kerosene for fuel. A "weed burner" was located at each stack so as to simulate the exhaust gases emitting from the engine cylinders. Thermocouples were attached to the stack surfaces and temperatures were recorded by utilizing a pyrometer panel with camera facilities (Figure 23). By incorporating a timer on the pyrometer panel, it was possible to obtain "temperature decay vs. time" data. Utilizing this data, modifications in coolant distribution were made until the required temperature reduction on all surfaces was obtained.

While the exhaust stacks could be cooled effectively by flowing coolant over them, external application of coolant to the power recovery turbines would not remove the heat from the large internal masses at a satisfactory rate. The turbines were well cooled with air except for those parts directly in contact with the exhaust gases. It was reasoned, therefore, that if adequate coolant were injected into the hot gas stream, cooling might be accomplished. Three nozzles, one placed in each of the inlet scrolls to the turbine nozzle box and directed downstream against the stationary vanes, cooled the turbine, not only rapidly, but with very little loss of liquid coolant, vaporization being practically complete. The cooling air cap and downstream shroud ring, being somewhat shadowed from direct impingement of the hot gases and, therefore, from the cooling spray escaped complete cooling. The addition of another nozzle in the flight hood, directed so that its spray impinged on the cooling air cap and the shroud ring, cooled these portions satisfactorily making inerting of the turbine complete.

Internal spray nozzles had been developed for the C-82 test crash wherein the nozzle orifice was protected from the hot exhaust gas blast during normal operation by a frangible ceramic disc that shattered upon application of coolant pressure and exposed the nozzle orifice. This nozzle was considered adaptable to the flight hood application, although certain major modifications were necessary in order to provide a wide angle spray, to make the disc resistant to breakage from vibration and heat distortion, and to greatly reduce the weight of the unit. A ceramic disc nozzle, described later in this section, was developed in which the above objectives were achieved.

To secure effective cooling from the nozzles in the inlet scrolls in the nozzle box, the spray had to be directed downstream to impinge directly on all the stationary vanes. This meant that the nozzle, when spraying, would have to be located out in the hot exhaust gas stream. However, in normal operation it would soon burn away. A retractable nozzle was devised, therefore, that under normal conditions was flush with the scroll inside surface; when actuated by the coolant pressure, it extended into the gas stream enough (3/4 inch) to properly direct the coolant spray downstream. The nozzle orifice was protected in normal use by being withdrawn into its housing and the sliding surfaces were protected from corrosion and carbon build-up by a protective lip similar to a poppet valve. Because of this construction, it was called the "Poppet" nozzle and its development is described more fully under a section so titled.

The development of the cooling system for the No. 1 turbine exhaust section finally reached a stage where fabrication of a prototype of this system for installation and testing on an R-3350-85 engine power package at Walter Kidde (Figure 24) was in order. In accordance with WADC preference, the prototype fabrication work was to be performed by a manufacturer of engine exhaust systems. The design of the exhaust disposal system of the R-3350-85 engine was considered to be critical in nature. It was felt that a manufacturer of exhaust systems, being cognizant of this fact, would therefore provide the required modifications in the best possible manner.

Accordingly, a subcontract for the work required was awarded to the Ryan Aeronautical Company. The required modifications consisted of tack welding the waffled pattern wire mesh to the exhaust surfaces, routing and properly securing the distribution tubes, and welding bosses into the turbine nozzle boxes and flight hoods to accommodate spray nozzles. A Walter Kidde engineer was sent to the Ryan plant to provide technical coordination.

While work was in progress at Ryan on the No. 1 turbine exhaust section, mockups of the Nos. 2 and 3 turbine exhaust sections were set up at Walter Kidde to develop the exhaust cooling system for these sections. When the development of the cooling systems for these turbine exhaust sections was complete, they were also sent to Ryan to be used as guides. The completed prototype exhaust assemblies received from Ryan were installed on the R-3350-85 engine power package at Walter Kidde.

By means of cold flow tests, heat tests, and fire tests, the routing of coolant distribution tubing and the locations of spray holes were optimized. The cold flow tests consisted of spraying a colored coolant solution onto the unheated exhaust assemblies to check the coolant coverage provided by the distribution system. Thermocouples on the exhaust surfaces were used to record "temperature vs. time" during the heat tests which consisted of operating the engine at full power until exhaust stack temperatures were stabilizied, then simultaneously cutting the engine and actuating the cooling system.

The fire tests consisted of first running the engine at full power until temperatures were stabilized. At engine shutdown, SAE No. 10 oil (oil having a lower spontaneous ignition temperature than gasoline) was sprayed onto the stacks until a fire occurred. The occurrence of fire proved the presence of an ignition source and a combustible mixture. At that time, the exhaust cooling system was actuated; it extinguished the fire and cooled the stacks. After the fire was extinguished, oil was resprayed during cooling. When the oil vapors did not ignite, the exhaust system was considered inerted. By moving the oil spray to various positions, all stack surfaces were surveyed. Difficulty was experienced in igniting the fuel spray on any but the hottest screen-covered surfaces, where prior to screening, the surfaces were above the ignition temperature of fuel. This would seem to indicate that the screening in itself provides an inerting effect.

Duplicating the final configurations (Figure 25) of the modified exhaust assemblies presented some problems, since it would have been practically impossible to modify additional assemblies from sketches or photographs. Therefore, the three mockup sections originally sent to Ryan were stripped of surface covering and tubing and returned to Walter Kidde so that the final tube routing and orifice and fitting locations of the prototype could be painted onto the respective stack sections. These painted stacks were then returned to Ryan and served as master templates for fabricating the additional exhaust systems for flight testing.

In addition to the modified exhaust assemblies already discussed, the complete engine exhaust cooling system included the following major components:

a. Air Bottle (Figures 26 and 27)

For an exhaust cooling system to operate satisfactorily when employing the principle of coolant aeration, a large volume of air is needed. Providing this volume of air within the coolant containers would have made them excessively large and heavy. Also, utilizing the air from pre-pressurized coolant containers for aeration would have progressively reduced the driving pressure of the coolant, thus seriously decreasing the effectiveness of aerated coolant distribution. To overcome these difficulties, a separate high pressure storage source for the air was used. The air was stored at 3000 psi and when released, it passed through a pressure regulator, exiting at 300 psi. In this way, driving air was maintained at relatively constant pressure during the period of aerated coolant distribution.

An air bottle of 200 cubic inches capacity was required. It was a cylindrical tube with welded hemispherical ends, bent to conform to the peripheral outline of the power section of the engine. It was constructed of 0.095 inch thick heat treated SAE 4130 chrome-moly seamless steel tubing to withstand a hydrostatic proof test pressure of 5200 psi. A coating of Araldite, an epoxy resin, was applied to the interior of the bottle to prevent corrosion. The outlet boss was located centrally on a side of the bottle so that when positioned on the engine, the outlet valve would be located between the inlet ends of the two coolant containers.

b. Air Bottle Discharge Valve (Figures 26 and 27)

The discharge valve used with the air bottle was a squib operated valve identical in design to that developed for the CO_2 engine inerting system except for the attachment fitting. This valve is fully described in the section on CO_2 system components.

c. Air Pressure Regulator (Figure 26)

An air pressure regulator was used in the engine exhaust cooling system for the purpose of supplying 300 psi air from the 3000 psi air storage bottle. The regulator was an existing one of relatively simple design. With a few minor modification, it was adapted to this purpose.

d. Coolant Containers (Figures 26, 27 and 28)

The two coolant containers of 2-1/4 gallons capacity each were fabricated of Type 347 stainless steel tubing, 3.5

inch O.D. and 0.042 inch wall thickness. They were bent to conform to the peripheral shape of the engine power section, and were mounted on the engine by means of brackets fastened to the engine cylinders. The mounting arrangement proved to be unsatisfactory, however; the difficulties experienced with the mounting during the flight tests are described in Section V. Tank operating pressure was 300 psi. The tanks withstood a hydrostatic proof pressure test of 500 psi and a burst pressure test of 1000 psi. Confirming the design analysis, the tanks exhibited no tendency to straighten under pressure. The air inlet ends of the tank were approximately 8 inches apart at the top of the engine. A check valve in the air supply line and a fill opening were located at the upper end of each container. The bottom end of each tank contained a burst disc type valve for coolant discharge.

e. Burst Disc Type Coolant Discharge Valve (Figures 26 and 28)

The body of this valve was an integral part of the coolant container, while the valve itself consisted of a burst disc arrangement with screen. A nylon seat check valve was incorporated in the discharge fitting. The burst disc was fabricated of 1/4 hard 2S aluminum with a coined circular notch. It withstood the effects of vibration up to the required 500 cps and 20 g's and burst at 250 ±10 psi static pressure. However, flight tests proved this valve to be unsatisfactory. The difficulties experienced are described in Section V.

f. Aerators (Figures 29 and 30)

There were three aerators, one for each turbine section. Each unit consisted of a 5/16 inch stainless steel tee with two ends used to house the air and coolant metering orifices of 0.070 inch and 0.110 inch diameter respectively. The third end of the tee was connected to the supply manifold. The aerators were attached to studs at the base of the engine fire shield.

g. Manifold (Figures 29-34 inclusive)

An engine exhaust cooling system manifold assembly was provided to transfer air and coolant from the storage tanks to the coolant distribution points. The manifold, which was made of stainless steel tubing, was attached by means of clamps to the forward side of the engine fire shield and followed the contour of the engine at that point. Various sizes of tubing were as follows:

1/2 inch diameter from the coolant tanks, into
5/16 inch diameter branches to the three aerators
1/4 inch diameter from air line cross fitting to aerators
5/16 inch diameter sub-manifolds out of aerators
3/16 inch diameter from sub-manifolds to stacks

1/4 inch diameter from 1/2 inch diameter coolant line to flight hoods and turbine nozzles

Standard AN fittings were used to connect the tubing with the exception of a special sleeve which was used to adapt the 1/4 inch diameter air line to the 5/16 inch aerator (tee).

h. Ceramic Disc Nozzle (Figure 35)

This nozzle is used for spraying coolant internally in the engine power recovery turbine flight hood. It consists of a sliding nozzle tip contained in a housing which screws into a boss welded to the flight hood. The sliding nozzle is spring-loaded against a frangible ceramic disc which covers the opening in the housing, and protects the nozzle orifice from the effects of the hot exhaust gases. The coolant pressure provided upon actuation of the system produced sufficient force to shatter the ceramic disc and force the piston-like nozzle to the end of its travel with the tip protruding from the housing and providing for an unobstructed spray of coolant.

i. Poppet Nozzle (Figure 36)

In order to get a spray nozzle out into the gas stream at the turbine nozzle box with the spray directed downstream and yet not have it burn up during normal engine operation, a retractable nozzle assembly was designed. From the development work done on the ceramic disc nozzle, it was known that a piston arrangement was reliable under the high heat of the exhaust gases as long as the gases were excluded from the clearance space, so a retractable nozzle was feasible from that standpoint. However, it was known that the turbine nozzle boxes were a critical section of the engine. Attaching additional weight in the form of nozzles to this critical section might materially shorten the life of this unit. A study of the nozzle box and consultation with the engine manufacturer led us to believe that the structure would take the added load, although the weight to be added had to be kept to a minimum.

The poppet nozzle consists of a piston contained in a housing that screws into a boss welded to the inlet scroll of the turbine nozzle box. A protective lip is provided over the clearance between piston and housing. A hold-back spring provides closing force on the protective lip and seals against the entry of hot exhaust gases. Pressure of the coolant upon actuation of the system extends the piston into the gas stream where the nozzle orifice is then in position to spray and cool the hot turbine vanes.

j. Coolant Solution

The coolant used was a water-base salt solution with a specific gravity of 1.48 and a freezing point of -40°F. This solution was developed by Walter Kidde & Company, Inc., under contract No. AF 18(600)-167.

To recapitulate with regard to operation of the engine exhaust cooling system, it is actuated by means of an electrical impulse from the control unit. This impulse detonates the two squibs in the air discharge valve; the explosive force ruptures the two burst discs and the air is allowed to flow to the pressure regulator. From the pressure regulator, part of the air goes to the coolant containers to pressurize them, while the remainder flows to the aerators. When a specified pressure is reached in the coolant containers, the retaining discs burst and release the coolant to the supply lines. The released coolant passes through a screen and a check valve at the outlet of each solution tank. Fragments of the ruptured disc are trapped by the screen, while the check valve prevents pressure surge from one coolant container to the other through the common manifold in the event that one disc bursts before the other. The released coolant flows to the internal spray nozzles at each turbine nozzle box and flight hood and also to the aerators. From the aerators, the mixture flows through the exhaust stack distribution tubing and is sprayed into the wire mesh through drilled holes in the tubing. Coolant spray at the exhaust stacks was found to start at 0.23 seconds after the electrical impulse is received at the air discharge valve.

2. Carbon Dioxide Engine Induction Inerting (Figures 37-40 inclusive)

The second source of ignition which previous study had identified in regard to aircraft engines was the possibility of back-fire and torching within the engine induction and exhaust system. In view of the success achieved with the CO_2 engine induction inerting system used in the C-82 test, it was decided in the case of the C-119 to take the same approach - that is, to inert the induction and exhaust system by diluting the entering fuel-air mixture to the point of non-flammability with CO_2. As a starting point in the development work on the engine induction inerting system, the solenoid-operated flood valve and the CO_2 bottle assembly from the C-82 test were evaluated for possible use on the R-3350-85 engines of the C-119F airplane.

a. CO_2 Discharge Valve

It was required that the CO_2 valve be able to operate at a minimum applied voltage of 12 VDC in the C-119F installation. Since the C-82 valve was designed for operation at 18-20 VDC, modification was necessary. However, since the necessary modification would add considerable weight to the valve solenoid, and since it was found also that the valve neck required strengthening to withstand the engine vibration requirement, it was felt that a different approach should be taken to the CO_2 valving problem.

A squib-operated valve was considered. A valve of this type had been recently developed by Walter Kidde & Company, Inc. The existing design, which utilized a single squib, was modified to provide for double squib operation to increase its reliability. The double squib valve functioned properly and passed the environmental tests with no difficulty. The valve was light enough to permit the use of standard pipe threads in connecting it to the CO_2 container.

Aside from having different size pipe threads, the same valve was also used as the CO_2 discharge valve for the auxiliary power plant shutoff system. In addition, the valve was found acceptable for use as the air bottle discharge valve of the engine exhaust cooling system. In this latter case, the connecting threads were designed as standard AN pneumatic type.

The valve design consisted of a "Y" shaped aluminum body containing a squib and burst disc assembly in each of the two arms of the "Y"; the leg of the "Y" was threaded to fit the CO_2 storage tank. Escape of the CO_2 from the container is blocked by the two burst discs until the CO_2 is required. Detonation of the two parallel connected squibs by means of an electrical impulse from the crash-fire prevention system control unit results in rupture of the two burst discs due to the high intensity pressure wave developed. The CO_2 is then free to flow through the valve passages to the common discharge line.

b. CO_2 Bottle

In the C-82 test crash, a CO_2 container having a capacity of 3-1/2 lbs. of CO_2 was adequate to inert the induction system of each of the two Pratt & Whitney R-2800 engines. The Wright R-3350-85 engines used in the C-119F had a greater induction system displacement. The quantity of CO_2 necessary to inert this induction system was arrived at by determining the actual quantity of CO_2 needed to purge the fuel-air mixture in the induction system volume plus the cylinder displacement volume and was determined to be 4-1/2 pounds. This quantity of CO_2 was contained in a standard 5 lb. capacity CO_2 cylinder having a 1 inch NPT thread.

With the requirement that the CO_2 bottle should be attached to the engine, a mounting location was selected on a blank pad on the right side of the rear cover of the accessory section of the engine. In order to mount the bottle in this location, it was necessary to relocate the engine generator on the left side of the rear cover because of insufficient clearance.

The bracket to hold the CO_2 bottle assembly was first made of sheet metal for lightness. However, vibration tests showed it to be too weak. Several attempts were made to strengthen this sheet metal design, without much success. The stronger it was made, the heavier it got until it became obvious that no weight saving was being realized by using a sheet metal construction. Consequently, a magnesium casting was designed and found to satisfy the requirements of MIL-E-5272A.

Two standard 1/8 inch pipe fitting ports leading into the engine induction system were located on the lower left side of the engine casing. These ports and passages were originally designed into the engine by the manufacturer for use in sensing induction diffuser pressure, but were later plugged when not needed. These ports, therefore, were used to inject the CO_2 into the induction system.

From the squib operated discharge valve, a supply line consisting of 1/2 inch diameter stainless steel tubing was run by the most direct route to these ports. At the ports, the supply line branched into two 1/4 inch diameter stainless steel lines which were attached by means of standard 1/8 inch pipe fittings.

3. Combustion Heaters (Figures 41-44 inclusive)

An extensive study of the C-119F heating and ventilating system indicated that the eight Janitrol combustion type heaters could become sources of fire in an otherwise survivable crash by virtue of the hot metal surfaces developed during normal operation. To preclude the possibility of this occurrence, two approaches were considered:

1. Prevent fuel mists from reaching the combustion heaters by installing shutoff dampers in the ducts leading to the cold-air plenum chamber.

2. Install coolant spray nozzles in each heater for the purpose of rapidly cooling the hot surfaces to a temperature below the spontaneous ignition temperature of the fuel mist.

Although the first arrangement would have the advantage of being the simplest and lightest, the second approach would be more positive by way of cooling the hot surfaces and thus eliminating them as sources of ignition. The second method also would have the advantage that the coolant spray striking the hot metal surfaces would produce an inert atmosphere throughout the heater and heater exhaust system. In addition, the latter method would not prevent subsequent operation of the heaters in the event of inadvertent discharge of coolant into them. It was decided, therefore, that the coolant spray method would be used to prevent the occurrence of crash fires which might otherwise be precipitated by the presence of the eight Janitrol combustion type heaters.

The initial step in the design and development of the coolant spray system for the combustion heaters was to determine which surfaces of the heater would become hot enough during normal operations to cause the ignition of crash-released combustibles. A temperature survey was conducted for this purpose using an S-200 Janitrol heater, the type installed in the C-119F airplane. The heater was set up so that the actual operating conditions experienced in the aircraft could be simulated. Numerous thermocouples, located at random on the heater surfaces, were connected to a pyrometer board for recording the surface temperatures during operation of the heater. Results of the survey indicated that surfaces hot enough to ignite a fuel-air mist were confined to the inner section of the heater. Except for a small area near the air intake end, cooling of this section would be required in view of the recording of temperatures as high as 1500°F.

Basically, the heater consisted of a combustion chamber and a heat exchanger. When the heater was in operation, air entered the combustion chamber, mixed with sprayed fuel, and was ignited by means of a spark plug. The burning mixture thus produced traveled to the end of the combustion chamber where it was made to turn 180° and flow back in the opposite direction through an annular passage surrounding the combustion chamber. At the downstream end of this passage, the mixture crossed over by means of four small tubes to an outer annular passage and flowed back in the original direction to the exhaust end. The air which eventually heated the interior of the aircraft, entered the inlet end of the heater assembly, became heated as it made one pass over the inner and outer passes of the burning mixture, and left the heater at the outlet end to be distributed.

Having ascertained which of the heater surfaces would require cooling, spray tests were conducted on the heater. Fifteen thermocouples were then symmetrically located on the combustion chamber and the corrugated shell of the inner passage in the areas where the high temperatures had previously been recorded. As in the temperature survey, the thermocouples were connected to the pyrometer board; this arrangement permitted the recording of "temperature vs. time" curves for the various spray configurations tested. The cooling tests were conducted by simultaneously releasing a precharged solution of water (wetting agent added) to spray nozzles, shutting off the heater ignition and recording "temperature vs. time". Also, airflow to the heater was controlled to simulate the reduced airflow experienced during a crash. The airflow was reduced from normal to cut-off in approximately seven seconds.

Various spray systems were tested before satisfactory cooling was achieved. The most troublesome problem encountered was that of the water spray ricocheting off the corrugated surfaces. This was overcome by having the spray from several small solid-cone pattern nozzles impinge on the corrugated surfaces from a nearly perpendicular direction. The nozzles were attached to a straight tube which was inserted into the top of the annular space between the inner and outer passes for the full length of the heater. The spray nozzles were directed inward toward the corrugated shell of the inner pass. To complete the cooling of the inner section of the heater, and in effect the entire heater, a single solid-cone pattern spray nozzle was inserted into the combustion chamber through an unused spark plug well. The spray from this nozzle, directed downstream, was quite effective in cooling the combustion chamber.

Because the coolant is sprayed in passages through which the cabin conditioning air passes, as well as in combustion air passages, vaporized coolant would enter the cabin. A coolant should, therefore, be chosen that will avoid any possible irritating effects from its vaporized state. If the coolant is to be stored in an unheated part of the airplane, a low freezing point salt solution would be required.

The cooling tests established the following specifications and requirements for the combustion heater cooling system:

1. 1-1/2 pints of water-detergent solution (0.2% detergent by weight serves as wetting agent) for each S-200 Janitrol combustion type heater.

2. A 500 cubic inch capacity coolant storage tank containing 345 cubic inches of coolant solution and 155 cubic inches of charging air at a pressure of $125 \begin{smallmatrix}+25 \\ -00\end{smallmatrix}$ psi.

3. Three nozzles attached to the straight tube, providing a combined flow rate of 0.35 gallons per minute at a pressure of 100 psi.

4. One nozzle for mounting in unused spark plug well of heater, providing a flow rate of 0.35 gallons per minute at a pressure of 100 psi.

Based on the foregoing experimentally determined specifications and requirements, final design of the spray system was completed and hardware was fabricated and tested. Upon the satisfactory completion of these final tests, additional hardware was made for installation in the two C-119 aircraft. A detailed description of the final design spray system illustrated in Figures 41-44 inclusive follows:

a. Internal Nozzle Assembly

The internal nozzle assembly was an all stainless steel unit, appropriately silver plated to prevent seizure between the nozzle body, the bushing, the locknut, and the spark plug well of the heater. The assembly consisted of a nozzle, a tube, and an adapter brazed together and inserted into a bushing. After the nozzle assembly was positioned in the spark plug well so that the centerline of the 30° solid-cone spray pattern was parallel to the longitudinal axis of the heater, it was secured in place with a locknut.

b. External Nozzle Assembly

The external nozzle tube assembly consisted of a preformed 1/4 inch O.D. tube to which three nozzles, a threaded bushing, and a mounting hook were brazed. With the exception of an aluminum AN sleeve and coupling nut, the unit was fabricated entirely of stainless steel. The spray pattern of each nozzle on the assembly was an 80° solid-cone. The nozzles were so positioned in the tube assembly that the center line of each cone-shaped spray pattern made an angle of 55° with the longitudinal axis of the heater. Once the external nozzle assembly was installed in the heater, it became an integral part of the heater, and did not need to be removed separately prior to removal of the heater from the airplane, should such become necessary.

c. Coolant Storage Tank

The coolant storage tank was a 500 cubic inch capacity, two piece, resistance welded, stainless steel sphere. A coating of Unichrome B194 on the inside of the sphere protected the weld seam from crevice corrosion. Four mounting lugs welded to the exterior of the sphere provided for bolting to a supporting box structure which in turn was attached to the aircraft. The tank was equipped with a pressure gage, an air filler valve, and a standard Walter Kidde cartridge valve for coolant fill and discharge. After filling the tank with 1-1/2 gallons of water-detergent solution through the fill connection, the filler neck was sealed with a frangible disc and capped with a cartridge type discharge valve. The tank was pressurized through the air filler valve to 125 psi. An electrical impulse from the control unit was required to cause a projectile in the discharge valve to rupture the frangible disc, thereby releasing the water-detergent solution to the distribution lines. A large screen in the valve trapped fragments of the frangible disc and the projectile in order to prevent them from clogging the distribution lines. The discharged solution flowed into a manifold from which it was conducted through individual lines to the eight heaters. A fine wire screen type strainer located at each heater prevented dirt particles from clogging the spray nozzles in the heaters.

4. Auxiliary Power Plant (Figures 45, 46 and 47)

The auxiliary power plant in the C-119F airplane represented another possible source of fire during crash landings by virtue of its hot exhaust surfaces, possible exhaust and induction flames, and electrical arcs and sparks. An electric generator and a two-cylinder, four-stroke cycle internal combustion engine comprised the auxiliary power plant which was mounted on the aft right section of the accessory equipment compartment floor of the airplane.

As with the main power plants, it was planned to inert the APP to prevent the ignition of fuel mists or vapors by any hot surfaces or by back-fires or exhaust torching. The generator would be inerted in the same manner as the main power plant generators, described in Section III-C-3, "Electrical Shutoff"

It was found that the only hot surfaces present were those of the exhaust stacks and manifold, areas normally insulated with a lagging of asbestos tape and waterglass. As the insulation surface was well below fuel mist ignition temperatures, it was considered that proper maintenance of this lagging would be sufficient to insure neutralization of the exhaust stacks as an ignition source.

APP engine shutoff is accomplished by closing of its solenoid-operated fuel valve upon de-energization of the airplane's electrical system. To inert the fuel mixture within the engine and thus prevent the possibility of back-fires or exhaust torching, a charge of CO_2 is injected into the air intake filter until the unloaded engine coasts to a stop.

A 4-ounce charge of CO_2 was stored in a standard 14.6 cubic inch capacity CO_2 bottle which was fitted with a squib-operated discharge valve that was identical, except for the screw end, to those used on the main engine and CO_2 bottles. Upon release the CO_2 would be conducted through 3/16 inch tubing to a nozzle built into the air intake filter.

C. Fuel, Oil, Hydraulic Fluid, and Electrical Shutoff

1. Fuel, Oil, and Hydraulic Fluid Shutoff Valves (Figure 48)

As previously noted, crash-fire prevention is accomplished by inerting ignition sources or by preventing combustibles from reaching ignition sources. Under crash conditions, the severe distortions in the nacelle may result in rupture of the lines carrying combustible fluids to the engine. As the spillage of these combustibles in a nacelle may very likely result in fire before the nacelle ignition sources can be inerted, it is desirable to prevent such spillage as far as possible. In the C-82 crash test, high speed shutoff valves were used in each fluid line in place of the usual firewall shutoff valves. These valves closed in .017 seconds after energization by the crash-fire prevention system.

The C-119F had three combustible fluids present at the nacelle; fuel, lubricating oil, and hydraulic fluid, each controlled by the usual firewall shutoff valve.

While the high speed shutoff valves used in the C-82 test crash functioned properly, it was realized that a very high degree of reliability was required for this device in view of the disastrous consequences that might result from inadvertent closing in flight. A very critical analysis was made, therefore, of the valve design. The following general requirements were established:

1. The valve should conform to the applicable portions of MIL-E-5272A and MIL-V-8608. These specifications involve a number of standard details that limit overall design.

2. Size all valves to the 2 inch O.D. tubing of the oil line requirements (the largest) so as to avoid three different size designs, but have three different sets of elastomer seals to be compatible with the three liquids.

3. The valve must be quick closing, combined with prompt automatic reopening so as to prevent interruption of service in case of inadvertent closing. The automatic reopening feature was to be negated by the crash impulse. A closing time of .017 seconds, as in the C-82 valve, was set as a goal, but to achieve reasonable cycle life, this was subsequently increased to .04 seconds.

4. The valve must be of a "fail-safe" design to minimize accidental closing. This design would include a separate electrical connector and wiring for the closing impulse so that a connector failure could not initself cause the valve to close.

5. The closing solenoid was to operate reliably between 12 and 30 volts, the rewind motor could operate at normal aircraft voltages of 18 to 30 volts.

The means for shutting off the fuel supply at the firewall consisted of a high-speed gate valve, similar in some respects to the shutoff valve used in the C-82 test. In this high-speed shutoff valve for the C-119, the flow of fuel is barred by the closing of a flat tear-drop shaped anodic "hard-coated" aluminum gate. Leakage around the gate is prevented by teflon seals in the valve body on either side of the gate. Motivating power for the gate is supplied by two inter-threaded helical coil torsion springs, and is transmitted through a spring retaining drum, a shaft, and a crank to the gate. A 12 to 28 volt DC motor winds the springs until the required energy has been stored; at this time, a torque sensing device attached to the motor contacts and opens a single pole, single throw switch in the motor circuit, thereby stopping the motor. The motor will be in continuous operation except when the springs are fully wound. The energy stored in the fully wound spring is of such magnitude that the valve closing time ranges from 55 to 35 milliseconds as the operating voltage is varied from 12 to 28 volts DC. Slightly longer periods of time would be required to open the valve immediately following its closing.

Movement of the gate is prevented until the desired moment by two latches, located 180° apart at the end of the drum. One latch controls closing of the gate, and the other latch controls opening. Each latch is released by a separate solenoid. When the gate is open, the closing solenoid will operate either when it receives a crash impulse from the control unit, or when a switch on the pilot's panel is manually moved to the "close" position. A microswitch in the closing solenoid circuit is in the closed position when the gate is open; thereby, a circuit is completed upon receipt of the crash impulse through a holding relay, or upon closing of the manual switch on the pilot's panel. Another microswitch in the opening solenoid circuit is in the open position when the gate is open; when the gate closes, this switch also closes. When the opening solenoid switch is in the closed position, set of contacts on the initiator relays to "lock-in" the coil upon actuation and hold the relay closed. In series with this "lock-in" were placed the contacts of a thermal time delay relay energized by the initiator relay that would open and release the initiator relay after 10 seconds. To provide continuous energy during the contact transfer time when the relay coil is otherwise not energized, a capacitor is placed in parallel with the coil of the initiator relay.

An indicator light is placed in parallel with the heater of the time delay relay, but physically located on the pilot's control panel (Figure 53), to indicate when an initiator relay is energized. Also located on the pilot's control panel is a switch in each initiator relay coil circuit, normally closed, which the pilot can open and thus de-energize any initiator relay should conditions warrant. The operation of an initiator relay while the airplane is in normal flight would presumably be inadvertent operation, leaving the crash-fire prevention system partially triggered - a condition under which the pilot would switch out the offending initiator circuit.

Additional contacts are provided on these initiator relays to provide for actuation of the master relays to accomplish the selectivity described in Section III-A.

No holding or "lock-in" feature is used for the individual initiator relays of the fuselage deformation switches. Here the contacts are merely arranged so that closing of any two will energize another relay. This other relay, labeled "strip delay relay" on the Schematic Diagram of Figure 51, provides the "lock-in" feature, holding again for 10 seconds.

In the C-82 test crash, overload circuit breakers were employed in the line to each actuator, their purpose being to open in the event that such a line was shorted during the crash, and so protect the line from fire and prevent a drain on the power supply. The weight of these devices, in the numbers needed, was considerable. Further, overload circuit breakers require appreciable time to trip so that it was believed they would be practically ineffective as protection during the 0.1 second that the actuators were energized. However, some protection is needed to prevent a drain on the available power supply for actuators if one or more should be shorted during a crash. Accordingly, in the C-119 control box design, resistors were placed in series with each actuator line to limit short circuit current to a safe value. Because such resistance provided a voltage drop in normal use and thus reduced the voltage available to the actuator, a resistance of one ohm was selected which permitted a maximum current drain and a normal voltage drop that could be tolerated.

2. Checkout System

The automatic checkout for continuity basically inserted a high impedance sensitive relay in series with each initiator and actuator circuit. If the circuit was continuous, enough current would flow to pull in the sensitive relay but not enough to actuate the initiator relay or actuator (even squib actuators). For the automatic continuity check, pull-in of the "sensing" relay, as it was called, merely advanced a solenoid driven selector switch (Figure 54) another step and then examined another circuit for continuity in the same manner.

Checking out the initiator relay circuit was done by energizing each relay in turn and checking with the sensing relay to see if it operated, and checking to see that the master relays did not operate. Initiator relays were then checked in all possible combinations, each time checking to see that the master relays were or were not operated as required. By choosing normally open or normally closed contacts of the sensing relay at each position with a bank of the stepping switch, the sensing relay could be used to advance the stepping switch one position if the proper result occurred, and thus provide automatic operation. If the switch "hung-up" at any position, it indicated that the proper result had not occurred -- in other words, a fault existed in the circuits indicated by that position.

To insure that power was not applied to actuators if the master relays were tripped during check, either deliberately or inadvertently, the first act of the checkout system was to remove the power from the master relay contacts, and the second step was to check with the sensing relay that the power was actually removed. The power was removed by using a relay-operated circuit breaker, thus making the last step of the checkout process the restoration of such power by manually resetting the circuit breaker, hence its designation "Arm System" circuit breaker.

With the revision of the initiating system as described in Section III-A, provision for automatic functional checkout became too complex to work out in the short time available, so a manual checkout system was devised. In this system, two rows of indicator lights were provided, one row amber, the other red. In each check position of the manually-operated selector switch (Figure 55) initiator relays are operated in various combinations and the resulting actuation of master relays recorded by the red bank of indicator lights, each marked with the relay it indicates. A separate section of the selector switch lights the amber indicator lights corresponding to the red lights which should be on at that position. Thus proper operation of the circuit is indicated by the lighting of an amber light and its corresponding red light. An amber indicator light without the corresponding red indicator lighted, or vice versa, would indicate a circuit malfunction. Certain non-checking intermediate steps were required to set up the circuits for functional tests. For example, an initiator relay would be energized at one position and checked at the next position to see if it was holding. For the first of these positions, an amber and corresponding red indicator (marked "Normal Operation") are lit directly by the selector switch merely to give the operator the usual indication to proceed to the next position.

In each case, the numbered position of the selector switch, both automatic and manual, and the indicator lights on or off, indicate which circuits are being tested and consequently, if a malfunction occurs, they indicate which circuit is faulty. From this information, if the checkout shows a malfunctioning circuit, a decision can be made in an emergency whether or not it is dangerous to fly with the crash-fire prevention system in operation,or whether only some part of the protection is not available.

3. Construction of Control Unit (Figures 56-59 inclusive)

During fabrication of a prototype control unit to test various parts of the circuitry, it was realized that it would be very difficult to wire the complete final units with any degree of assurance that the wiring would be intact when finished or that it would remain intact in service. The mechanical design of the final control unit was, therefore, based on eliminating this problem.

The complexity of the relay interwiring prompted the consideration of printed circuits.

To permit mounting the relays directly to the printed circuit board, a mounting was devised which removed the stress from the soldered relay terminal connections at the board. The relay was mounted by its mounting flange on stand-offs. A bracket was then clamped around the relay case and fastened to the board to provide three point support. Epoxy and fiberglass were selected for the board material as having the best strength and durability.

Toggle switches, indicator lights, and circuit breakers were mounted to the front panel. The rear panel was selected for the large number of electrical connectors needed for the connections in and out of the control unit.

A box construction was designed in which the sides and bottom were formed of one piece with rolled-over edges to receive the front and rear panels and the top cover.

Two channel sections extending from side to side and welded to the bottom and sides formed a rigid base on which to mount the checkout switches and a resistor and capacitor mounting terminal board. The initiator and master relay sub-chassis were of box cross-section themselves, and were bolted to the top rolled-over edges of each side. These chassis with the channel section across the bottom and with the front and rear panels and the top cover bolted in place made an extremely rigid box structure for the control unit that prevented flexing and distortion of the components that might interefere with their operation. Such rigidity further improved the reliability of the wiring connections eliminating possible damage to them from relative movement of components. At assembly, the wire cables were well supported at suitable points to eliminate any cable motion that would put stress on connections or abrade the wires.

The extreme precautions taken in the wiring of these control units proved their worth. In wiring the three units fabricated, there was not a single wiring failure encountered, and neither was there a wiring failure in the environmental testing or the flight tests.

The benefits to be derived from the proper operation of the crash-fire prevention system, should it ever be called upon to operate, and the hazards inherent in its inadvertent operation make reliability of the electrical circuitry as well as the mechanical components of utmost importance. Consequently, the control unit was designed and built with reliability and safety as the foremost consideration. It is believed that this goal was achieved to an outstanding degree.

E. Monitor Unit

The monitor unit is not part of the crash-fire prevention system. It was designed and built expressly and solely for the purpose of permitting the observation of inadvertent operation, should such occur, of the various initiating, actuating, and shutoff devices and the control unit master relays during flight testing of the system. As previously mentioned, the numerous components of the system were developed and tested in accordance with the requirements of environmental specification MIL-E-5272A. In addition, the parts were also required to withstand the performance characteristics of the airplane. Conformance to this latter requirement, therefore, was checked with the use of the monitor unit during flight tests.

In general, the monitor unit consisted of a number of relays, switches, indicator lights, AN connectors, and terminal boards all housed within a special cabinet. The wiring to and within the unit was so arranged that the unit could detect inadvertent operation of the following components:

1. The initiator (crash sensing) switches.
2. The actuator devices (air, coolant, and CO_2 discharge valves).
3. The shutoff valves (fuel, oil, and hydraulic fluid).
4. The actuator or master relays (in control unit).
5. The battery contactor.

A detailed discussion of the monitor unit circuitry (Figure 60) is in order here.

The initiator switches were connected to the monitor unit using the same wiring which would ordinarily be used to connect them to the control unit. Actual connection of these wires to the monitor unit was made with AN connectors. The internal circuitry was such that upon inadvertent closing of an initiator switch, for even a fraction of a second, a relay similar to the initiator relays in the control unit would be energized and thereby light an indicator light on the front of the monitor unit. This light would remain lit and the relay energized until notice was taken and they were shut off by momenterily removing the power to the unit. To permit continuity checks of the control unit initiator relay circuits during the monitored flight testing, each AN initiator connector on the rear panel of the control unit was shunted.

Monitoring of the various actuating devices was slightly more involved than monitoring the initiator switches. In the first place, it was necessary to insure that monitored flight testing could be performed without any possibility of losing the aircraft as a result of inadvertent operation of any of the actuators affecting the engines. To accomplish this and at the same time provide for positive detection of inadvertent operation of any of the actuators, the discharge lines from all the CO_2, coolant, and air storage tanks were plugged upstream of their respective delivery points. Pressure switches were then inserted in the CO_2 and air supply lines downstream of the discharge valves, and liquid-level switches were inserted into the coolant supply lines in similar locations. All these switches were individually connected to a terminal board on the back of the monitor unit. Internally, the circuitry from the terminal board was similar to that for the initiator switches described in the previous paragraph. Inadvertent operation of a discharge valve or rupture of a coolant retaining disc would result in closing of the contacts of the pressure of liquid-level switch in the particular supply line involved. This would then energize a relay, causing an indicator light to go on and stay on. Again, momentary power shutoff would be required to de-energize the relay and put out the light.

The engine fuel, oil, and hydraulic fluid shutoff valves were monitored by means of an arrangement of relays, indicator lights, and switches somewhat different from the arrangement used for the initiating and actuating devices. During "monitored" operation of the system, an indicator light for each valve remained "on" continuously as long as the valve stayed open. Each indicator was lighted through the normally open contacts of a relay which was in series with the corresponding valve closing solenoid. Inadvertent closing of the valve was immediately indicated by the closing of the corresponding light as a result of breaking of the circuit through the closing solenoid and de-energization of the relay. Two switches in each valve monitor circuit were included to permit manual opening and closing of the valves by the flight engineers; otherwise, the valve switches on the pilot'a panel could have been used. An inadvertently closed valve could be reopened by moving the "open" switch to its normally open position and then releasing; the indicator light would then relight.

For monitoring the master relays in the control unit, four additional relay-indicator light circuits in the monitor unit were connected to the master relay output terminals on the control unit rear panel. Thus, any inadvertent operation of one of the control unit master relays would result in lighting of an indicator light which would remain on until the power to the monitor unit was turned off. Included in the circuitry relating to the four control unit master relays were three switches which could be used to duplicate the actions of the pilot's crash switches and thereby permit operation of the master relays for checking.

In addition to the various other components of the crash-fire prevention system, it was also desired to monitor the operation of the battery and ground contactor unit during flight test. Because of safety considerations, the battery and ground contactor unit was connected so that it exercised no control over the aircraft electrical system. The existing battery contactor and pilot's battery switch were retained for control of aircraft electrical power. An indicator light and a battery disconnect switch were installed side by side on the panel of the monitor unit. The indicator light was controlled by the contacts of the battery contactor, and was continually lighted while the battery contactor was closed. The battery disconnect switch was connected in the circuit of the battery contactor coil and could be used to de-energize that coil and thereby cause the battery contactor to open. Inadvertent closing of the ground contactor would trip its overload circuit breaker and would be indicated by the lighting of the "bus grounding circuir breaker tripped" indicator light on the front panel of the control unit.

The various generator field coil circuit breakers which would be included in an armed crash-fire prevention system were left unconnected for the monitored flight test because of flight safety considerations.

IV. INSTALLATION

To test the airworthiness and reliability of the crash-fire prevention system, two identical systems were installed on C-119G aircraft, Serial Nos. 51-8040 and 51-8046. Although the contract originally specified the use of C-119F aircraft for flight testing, the C-119G's were available at the time of installation and since there was no difference between the two models in regard to the crash-fire prevention system, the latter model was used. Installation work was performed by the Experimental Fabrication Division, Directorate of Support of the Wright Air Development Center. Walter Kidde personnel provided technical guidance during the installation. Installation drawings were prepared by Walter Kidde as the various phases of installation work were completed. The crash-fire prevention equipment was installed for monitored flight testing in such manner that the malfunction of any component or group of components would not in any way affect the normal operation of the aircraft. Those components that would affect the normal operation of the aircraft if they were to malfunction, such as fuel shutoff valves, CO_2 injection system, and electrical power shutoff system, were to be installed temporarily in dummy lines, circuits, and systems in the proximity of their final locations, and wherever possible installed so as to simulate their normal functioning.

A. Initiating System Components

The rubber-strip type deformation switches were attached to the aircraft fuselage in their predetermined locations by means of rivets and aluminum retaining strips (Figures 4, 5 and 6).

Installation of the wing cable deformation switches involved opening the lower skin of the wing (Figure 61) and attaching the cable housing, the cable anchor, and the switch itself to either the ribs or the main spar. Hat shaped brackets (Figures 9 through 13) were used to mount the switch and the anchor on the forward face of the main spar, while angle brackets and clamps secured the cable housing tube to alternate ribs (Figure 62).

The engine reaction switches were installed in their proper locations in the region of the engine accessory section as shown in Figures 16 to 20, inclusive.

B. Inerting System Components

1. Engine Exhaust Disposal System (Figures 26-36 inclusive)

All existing exhaust stacks, flight hoods and turbine assemblies were removed from the engine. The turbine assemblies were disassembled and the modified nozzle boxes were installed in their respective positions. The internal spray poppet type nozzles were installed in the bosses on the nozzle box and positioned. The coolant and air manifolds, aerators, and mixing manifolds were attached to the base of the engine diaphragm; then, the modified turbine assemblies were fitted in their respective positions and the exhaust stacks were installed. The modified flight hoods each containing an internal spraying ceramic disc type nozzle, were also installed. All coolant distribution tubing was connected from the manifolding to the stack sections and spray nozzles.

The coolant containers and air bottle were attached to the engine power section by means of brackets fastened to the extra threaded holes on the rocker arm boxes, some of which mount the cowl ring to the cylinder heads. The two air bottle brackets were mounted on studs which hold the ignition coil saddle.

For monitoring purposes, the coolant manifold was not connected to the coolant discharge valve. Instead, the coolant discharge port was plugged with a liquid level switch to monitor possible inadvertent rupture or leakage through the discharge valve burst disc. In like manner the outlet port of the air bottle discharge valve was dead-ended to a pressure switch.

2. CO_2 Engine Induction Inerting System (Figures 37-40 inclusive, and 63)

The engine generator was relocated to the left side of the rear cover of the engine accessory section and the CO_2 bottle and bracket were fitted in the vacated position, the cast magnesium bracket acting as the cover plate on the power take-off pad. The dual-fitting CO_2 manifold was screwed onto the two induction system inlet port fittings and routed to the discharge valve outlet. Connection, however, was not made to the valve. Instead, the CO_2 discharge valve outlet was capped with a pressure switch for monitoring purposes.

3. Combustion Heater Inerting System

The details of the installation of the combustion heater inerting system are shown in Figures 41-44 inclusive. In order to install the crash-fire prevention inerting and cooling system on the combustion heaters, it was necessary to remove all eight heaters from the airplane. Each heater was equipped with its separate spraying system -- i.e., internal and external coolant spray nozzles -- and re-installed in its respective position. Since the combustion heaters'mounting was arranged in the form of an arc, the inerting and cooling spray system for each heater had to be installed so that the external spray nozzles were located at the top of the vertical centerline plane when the heater was re-installed. The coolant tank was located forward of the combustion heater cold air plenum chamber on the upper instrument deck of the airplane. Manifolding was routed from the coolant tank discharge valve to the individual heater spray systems. The coolant tank discharge valve was not connected to the manifolding, but was dead-ended to a liquid level switch for monitoring. For the flight test, a water solution,with wetting agent added, was used as the coolant.

4. Auxiliary Power Plant

Details of the APP shutoff and induction inerting system are shown in Figures 46 and 47. The CO_2 injection nozzle was attached to the carburetor air cleaner and the CO_2 bottle was mounted by means of a bracket to overhead fuselage structure. The supply line from the CO_2 discharge valve was capped with a pressure switch and secured to the bracket. The bottle was installed in the inverted position since it contained no syphon tube.

C. Fuel, Oil, Hydraulic Fluid and Electrical Shutoff Systems

1. Fuel, Oil, and Hydraulic Fluid Shutoff Valves (Figures 48, 64-66 inclusive)

The high-speed fuel, oil, and hydraulic fluid firewall shutoff valves were mounted in parallel with the existing firewall valves of the aircraft on special manifolds and installed on a specially constructed platform located just aft of the wheel well front bulkhead in the general area of the original valve installation.

The purpose of mounting the crash-fire valves in parallel with the ship's valves was to subject them to normal pressures encountered in flight conditions, but not interfere with normal flow of fluids should a crash-fire valve inadvertently close. The crash-fire valves were electrically controlled from the monitor unit and were cycled during flight without interfering with normal flight operations. An indicator light on the monitor unit identified the position of each valve gate -- i.e., open or closed.

2. Spinner Discharge Valve (Figure 50)

The original spinner discharge valve was removed from each engine and the modified stem and cover assembly was attached. The valves were bench tested according to the Maintenance Instructions (Reference 6) and re-installed. However, the air line to the valve was not connected; instead, the air inlet opening was capped so that the valve would be unaffected by inadvertent operation of the crash-fire engine exhaust cooling system.

3. Electrical Shutoff

The combination battery and grounding contactor and the 100-ampere overload circuit breaker were mounted in a metal box and located on the right side of the cargo compartment near the existing battery contactor (Figures 67 and 68). This box also contained three other circuit breakers: 50 amps for power to crash-fire system, 5 amps for battery center tap (14 v) for the short anticipator circuit, and 15 amp for power to the monitor unit. These three circuit breakers are not part of the crash-fire prevention system, but were added as safety precautions during the flight test.

D. Control Unit

The control unit, with its vibration isolator, was mounted on structural channels and installed on the middle shelf of the equipment rack at the rear of the pilot's compartment (Figure 69). Control unit wiring (Figures 70-77 inclusive) was installed as follows:

1. Initiating Circuits

The wires from the initiators (crash sensing devices) were run directly to the monitor unit using the special shielded wire for the short anticipator feature.

2. Actuating Circuits

Special cables were run from the firewall shutoff valves (fuel, oil, and hydraulic fluid) to the monitor unit terminal strip, using standard aircraft wire, to permit operation of the valves from the control switches on the monitor unit panel, and to indicate, with lights, operation of the valves, both normal and inadvertent.

The various actuator connectors (master relay outputs) at the back of the control unit were connected to the monitor unit so as to permit monitoring of the master relays.

The squibs of the coolant pressurizing air bottle valves, the engine CO_2 bottle valves, the APP CO_2 bottle valve, and the combustion heater coolant tank valve were connected directly to the control unit connectors using the special shielded wire for short circuit anticipation. The pressure switches which would detect inadvertent operation of any of these valves, as well as the liquid level switches which would indicate inadvertent discharge of the coolant, were connected, with standard aircraft wire, to the monitoring unit where an indicator light would indicate operation of the monitoring device. Having the squibs electrically connected would test for the presence of stray electrical fields which might induce enough current in the connecting wires to fire the squibs.

3. Pilot's Panel

The pilot's panel was installed in the center of the windshield between the pilot and co-pilot, necessitating relocation of the magnetic compass. The electrical conductors from the pilot's panel through the junction box to the control unit were terminated at the rear of the control unit in electrical connectors. However, connection was not made to the control unit since a duplication of the pilot's panel was provided on the monitor unit for use during the monitoring of the flight tests.

E. Monitor Unit

The monitor unit was mounted directly to the upper shelf of the equipment rack just above the control unit (Figure 69). The monitor wiring cables were routed from all sections of the aircraft in parallel with the crash-fire cables (Figures 70-77 inclusive) passing through the same conduits, clamps, and hardware. Electrical connections to the monitor unit were made on a terminal strip at the rear of the unit, with the exception of the special shielded wire from the initiators which terminated in connectors on the monitor unit. These were interchangeable with the control unit plugs.

F. Checkout

As electrical connection was made to each component of the crash-fire prevention system, the wires and connections were "rung out" to be sure that the wires were properly routed and connected. Upon completion of this check, power was applied to the system for a checkout of the circuitry.

Each initiator was tripped mechanically to test for proper indication at the monitor unit.

Each monitoring pick-up (pressures and liquid-level switches) was shunted to test for proper indication at the monitor unit.

All shutoff valves were operated electrically using the switches provided to test for proper operation of the valve and proper indication on the monitor unit.

The battery and ground contactor was operated with the switch provided on the monitor unit to test for proper operation, and for indication on the monitor unit. The overload circuit breaker was manually tripped to test the indication on the control unit panel.

The control unit system checkout was tested. Because the initiators were connected directly to the monitor unit instead of to the control unit, it was necessary to provide jumpers at their connectors on the control unit to permit the automatic continuity check to function.

V. FLIGHT TEST

As previously mentioned, the crash-fire prevention system was required to satisfactorily withstand the performance characteristics of the C-119 airplane. Compliance with this requirement was checked by means of the monitor unit during flight tests at Wright-Patterson Air Force Base. The flight testing was performed by personnel of the Directorate of Flight and All-Weather Testing of the Wright Air Development Center. The flight tests consisted of the following specified modes of operation of the aircraft:

1. Short field take-off (excess power)
2. Maximum climb configuration
3. Maximum dive configuration
4. Short field landing (hard brakes, full prop reversal)
5. Simulated ILS landings (gear down, flap down, high engine rpm)
6. Climb-out (170 knots, indicated airspeed)
7. A/C normal flight pattern, normal maneuvers
8. 1000 foot descents at 200 knots indicated airspeed
9. 90° flight turns
10. Ground control approaches
11. Single engine at altitude (other prop feathered)
12. Simulated landing approaches and go-arounds
13. Climbing and descending turns
14. Slow speed stalls
15. Hard landings
16. Heavy to moderate turbulence (bad weather flying)
17. Cross country (6 to 10 hours each flight, cruise 150 to 190 knots true airspeed)
18. Accoustical and vibration tests during flight (maximum vibration condition, out-of-phase props)

A combined total of 282 hours and 45 minutes of flight testing were performed on the crash-fire prevention systems installed in the two C-119G aircraft, serial Nos. 51-8040 and 51-8046. The equipment in the former plane was monitored by a WADC flight engineer for 76 hours. After this time, the equipment was removed and inspected. In the latter airplane, the equipment was monitored for 206 hours and 45 minutes by a WADC flight engineer.

The original flight test plan for each of the two complete systems, as stated in the contract, was to accrue 50 hours of monitored flight time, then modify the installation to the actual crash-fire prevention system configuration and fly approximately 100 hours with the system in the armed condition. From the paragraph above, it is obvious that the flight test plan was partly revised. The decision to forego the 100 hours of system armed flight testing was made by WADC.

The status of the flight testing at the completion of 50 hours on each of the two aircraft was as follows:

1. The majority of the crash-fire prevention system components had satisfactorily demonstrated their airworthiness by withstanding the rigors of the flight test.

2. A few components had not demonstrated their airworthiness; they were, generally speaking, components whose design had necessarily been tailored around the R-3350 engines.

3. A few other components showed some deficiencies to be present, but the solutions to these problems were apparent.

In order to fly a completely armed system, further development of the troublesome components would have been necessary. Also, considerable time and money would have had to be expended to change the installations to the final armed configuration. Since both the funds available and the period of assignment of the two aircraft to WADC were limited, and in view of the fact that armed system flight would provide no additional proof of the airworthiness of the crash-fire prevention system components beyond that shown by the monitored flight testing, it was decided to exclude armed system flight testing from the program. Instead, the systems installed for monitoring were to be flown for as long as possible to determine the endurance of the components.

During the approximately 282 hours of monitored flight testing, several different troubles arose. The deficiencies encountered during flight testing and corrective actions taken are described in the following paragraphs.

A trouble which showed up frequently during the early phase of the flight tests was that of false indications from the 6 o'clock position engine reaction switches. The plunger setting of ±0.125 inches allowable fore and aft engine mount movement during normal operation of the aircraft was evidently not large enough, although the maximum fore and aft movement of the mounts to full snubbing was given by the manufacturer as ±0.180 inches from a no-load position, and measurements taken of the reaction switch probe movement, on the test engine installation at Walter Kidde test grounds,revealed a maximum total movement of only 1/16 inch.

Measurements made by WADC of the actual plunger movement during flight indicated that the switch should be set to allow a plunger displacement of 0.200 inches forward or 0.130 inches rearward. This setting was made and no further triggering was encountered from the 6 o'clock position reaction switches during subsequent flight tests. The upper or one o'clock position reaction switches gave no trouble with the original plunger setting of ±0.125 inches.

Another area of difficulty was encountered with the breaking of the burst discs in the coolant discharge valves during flight as a result of flight-induced hydraulic impact loads.

Further difficulties were evidenced in the cracking of coolant tanks, the breaking of coolant tank mounting brackets, and the failure of the mounting studs. Flexing of the coolant tank bracket strap clamps allowed the tanks to move in their brackets producing a metal-to-metal rubbing action between tanks and brackets, which resulted in tank wall wear and subsequent failure. The sheet metal coolant tank mounting brackets proved to be marginal in design while the failures of the mounting studs occurred because of the high stress condition experienced by these studs due to excessive vibratory and "g" loads induced by flight.

As a result of breaking of mounting brackets, mounting studs, and cracking of coolant tanks, the tankage and associated bracketry was removed from the engines for the duration of the flight testing. Because the tanks were removed, it was no longer possible to provide flight test experience for any correction being considered for the coolant discharge burst disc valves. Therefore, no correction was made, although the following possible solutions were advanced:

1. Reduce the impact load on the burst disc by employing an impact absorber in the form of a molded rubber hemispherical hat over the disc, the brim furnishing the seal, and the dome providing an air cushion to absorb impact and yet transmit the pressurization force to the disc for bursting. Upon bursting of the disc, the rubber would follow through the opening and also burst.

2. Provide a support under the burst disc which would be removed by the pressurizing air. Thus, impact loads would be absorbed by the support until the tank was pressurized, at which time the support would be released by the force of the air permitting the disc to burst.

3. Provide a cartridge-operated frangible disc type valve, as used in standard aircraft liquid agent fire extinguishers. This type of valve employs a heavier disc which can withstand the high hydraulic impact loads developed. The disc is shattered by the force of a cartridge pellet fired at it by an explosive; and thus is not limited to the breaking force that can be developed from pressurization.

4. Provide a pneumatically-operated poppet type valve in which the moving poppet is opened and locked by the pressurizing air.

Although correction of the deficiencies noted in the coolant tanks and their mounting was not accomplished, the failure provided the knowledge and experience necessary to permit successful installations on this or other engines. Corrective action for this particular engine, or for other engines, would be to provide additional points of support to further distribute the load, the use of high strength studs and, if necessary, the use of a flexible mounting to reduce the effect of vibration and high "g" loads imposed by flight.

During the flight testing, there were two instances of inadvertent operation of a high-speed firewall shutoff valve. In both instances, the fault was traced to shorted terminals in the crash-fire system junction box. Excessive vibratory loading was experienced by the wire terminals because the junction box was located directly in the plane of vibration of the propellers. It was established that in neither of these instances did the valve itself malfunction.

VI. INSPECTION

In January 1958, it became necessary for WADC to remove the crash-fire prevention system from aircraft No. 51-8040 because that particular airplane was to be reassigned. After the entire system was removed from the aircraft, Walter Kidde engineers inspected the various components with respect to physical condition and functionability after the 76 hours of flight testing.

The following is a report on the inspection of the crash-fire prevention system components removed from that aircraft.

A. Initiating System

1. Rubber Deformation Switches

Each of the three rubber deformation switches was tested for continuity with the use of an ohm meter while pressure was applied at numerous random points along their length. All three switches were found to be in good condition and in working order.

2. Cable Deformation Switches

The cable deformation switches were inspected for mechanical operation and electrical continuity and were found to be in satisfactory condition.

3. Reaction Switches

The reaction switches were inspected for mechanical operation and electrical continuity and were found to be satisfactory. The associated bracketry was visually inspected and found to be in good condition.

B. Inerting and Cooling (Actuation) Systems

1. Engine Exhaust Cooling

A representative sample (approximately one turbine section) of exhaust stacks was flow checked using water. Coverage appeared satisfactory and there were no indications of clogged distribution holes. The screen ribbon covering the distribution tubing was found detached in a number of places. The primary function of this ribbon covering was to minimize the possibility of clogging of the coolant distribution holes. The fact that clogging had not occurred (as indicated by flow test results) where the ribbon had become detached suggests the possibility that the ribbon covering may not be necessary.

There was one exhaust stack from which the screen covering had deteriorated over an area of approximately one square inch. Examination revealed that this region of the stack had experienced excessively high temperatures causing the screen covering to burn away.

The coolant and air manifolds were in excellent condition.

Of the eighteen poppet spray nozzles spraying coolant into the turbine nozzle box scrolls to cool the vanes and turbine blades, the threads of two nozzle assemblies froze in their respective bosses and sheared in the process of removal. There were five nozzles in which the poppet protective end flange had become detached. No poppet nozzle was found extended. In spray testing, the nozzles from which the protective flanges had become detached were found partially clogged, while all nozzles that had retained the protective flange extended and sprayed properly under the design driving pressure of 200 psi.

Of the six frangible ceramic disc spray nozzles from the flight hoods, five discs were found broken with two of the five nozzles being partially clogged. The unbroken disc burst at 150 psi pressure and the nozzle sprayed properly.

2. Carbon Dioxide Engine Induction Inerting

Both CO_2 bottles and brackets were found to be in good condition. Approximately one pound of CO_2 had leaked out of one bottle.

3. Combustion Heaters

The inerting and cooling system for the combustion heaters was found to be in excellent working order. Several spray nozzles, both internal and external, were flow checked using water. There were no clogged nozzles evident.

4. Auxiliary Power Plant

The CO_2 charge in the bottle was found to have leaked out.

The modification to the air cleaner, i.e. addition of a CO_2 discharge nozzle, was in excellent condition.

C. Fuel, Oil, Hydraulic Fluid, and Electrical Shutoff Systems

1. Fuel, Oil, and Hydraulic Fluid Shutoff Valves

Prior to removal of these valves from the aircraft, an operational check was performed. No further functional checks were conducted during inspection. A visual check revealed no defects.

2. Spinner Discharge Valve

At the time of inspection, neither of the two spinner discharge valves had been removed from their respective engines and, therefore, were not available for inspection.

3. Battery and Ground Contactor

The battery and ground contactor and the grounding contactor overload circuit breaker were functionally checked prior to removal from the aircraft and were found to be operating satisfactorily. Visual inspection revealed no physical defects.

D. Control Unit

The control unit was checked, after removal from the aircraft by using the special checkout unit originally constructed for instruction purposes during the installation period. It performed satisfactorily. The short anticipator system circuitry was checked and found in working order.

VII. WEIGHT BREAKDOWN

The weight of the crash-fire prevention system discussed in this report was 564.37 pounds, including electrical wiring. A breakdown by sub-systems follows:

Initiating System	59.00
Inerting and Cooling (Actuation) Systems	
Engine Exhaust Cooling	240.74
Carbon Dioxide Engine Induction Inerting	54.30
Combustion Heaters	36.23
Auxiliary Power Plant	6.39
Fuel, Oil, Hydraulic Fluid, and Electrical Shutoff	
Fuel, Oil, and Hydraulic Fluid Shutoff Valves	90.32
Spinner Discharge Valves	1.50
Electrical Shutoff and Control Unit	75.89
System Total Weight	564.37

VIII. CONCLUSION

The feasibility of providing an airworthy and reliable aircraft crash-fire prevention system was demonstrated by the flight tests conducted on the pre-production type crash-fire prevention equipment which was developed and integrated into a completely automatic system. The two major troubles encountered -- namely, the coolant tank burst disc and mounting failures -- can be remedied by ordinary application engineering. Both the principles and the majority of the system components are applicable to aircraft other than the C-119 which was used as the flight test vehicle.

Furthermore, even though the engines involved in these tests were of the reciprocating type, the established crash-fire prevention principles of automatically detecting crash damage, inerting ignition sources, and providing combustible fluid and electrical shutoff would apply to jet engine powered aircraft as well (Reference 8). Although the methods of accomplishment might vary, the devices used would be essentially similar and equally reliable and airworthy.

APPENDIX A

LIST OF REFERENCES

1. NACA Report 1133, "Mechanism of Start and Development of Aircraft Crash Fires", I. I. Pinkel, G. M. Preston, G. J. Pesman, 1954

2. USAF Contract No. AF 18(600)-167, "Aircraft Crash-Fire Protection System".

3. Walter Kidde & Company, Inc., Report, "Aircraft Crash-fire Protection System", R. G. Towle, R. G. Diquattro, and K. Fritsch, January 6,1954, (Contract No. AF 18(600)-167, E.O. No. SR-12RDO-664.80A-4).

4. NACA RM E55B11, "Full Scale Performance Study of a Prototype Crash-Fire Protection System for Reciprocating Engine-Powered Airplanes", D. O. Black and J. C. Moser, 1955.

5. USAF Contract No. AF 33(616)-2246, "An Engineering Study of Aircraft Crash-Fire Prevention", December 1, 1953.

6. Maintenance Instructions, USAF Series C-119F Aircraft, AN01-115CCB-2, (Restricted).

7. NACA TN 3774, "Proposed Initiating System for Crash-Fire Prevention Systems", J. C. Moser and D. O. Black, 1956.

8. NACA TN 3973, "Origin and Prevention of Crash Fires In Turbojet Aircraft", I. I. Pinkel, S. Weiss, G. M. Preston, G. J. Pesman, 1957.

APPENDIX B

LIST OF SPECIFICATIONS

1. MIL-E-5272A, 15 July 1955, Amendment I - General Specification for Environmental Testing, Aeronautical and Associated Equipment.

2. MIL-E-5400, 25 January 1954, Amendment I - General Specification for Electronic Equipment, Airborne.

3. MIL-S-8484, 25 June 1954, Seals and Seal Testing Procedure (For Electronic Inclosures).

4. MIL-V-8608, 19 April 1957, Valves, Fuel Shutoff, Electric Motor Operated, 28 VDC.

APPENDIX C

LIST OF ENVIRONMENTAL TESTS REQUIRED

The contract, AF 33(616)-2246, covering the development and manufacture of the components of the crash-fire prevention system specified that environmental tests be performed on the various components in accordance with MIL-E-5272A.

Specifically, the tests performed were as follows:

1. Humidity - Procedure I
2. Salt Spray
3. Vibration - Procedure I or II as applicable
4. Low Temperature - Procedure II
5. High Temperature - Procedure I
6. Sand and Dust - Procedure II

The table below indicates which environmental tests were performed on each particular component of the system.

TABLE OF APPLICABLE ENVIRONMENTAL TESTING PROCEDURES

EQUIPMENT	HUMIDITY	SALT SPRAY	VIBRATION	LOW TEMPERATURE	HIGH TEMPERATURE	SAND & DUST
1. High Speed Shutoff Valves and Brackets	I	S(1)	I	II	I	II
2. Spinner Discharge Valve	--	--	II	--	--	--
3. Exhaust System Cooling System:						
a. Air Bottle Squib Discharge Valve	I	S	II	II	I	II
b. Pressure Regulator	I	S	II	II	I	II
c. Poppet Nozzle	I	S	II	--	Special(2)	II
d. Ceramic Disc Nozzle	I	S	II	--	Special	II
e. Coolant Discharge Valve	--	--	II	--	--	--
4. CO_2 Induction Inerting System - CO_2 container, Valve and Bracket	--	--	II	--	--	--
5. Control Unit and Bracket	I	--	I	II	I	--
6. Time Delay Unit	I	S	I	II	I	II
7. Inertia Switch and Bracket	I	S	I	II	I	II
8. Reaction Switches and Brackets	I	S	II	II	I	II
9. Cable Type Deformation Switch	I	S	I	II	I	--
10. Rubber-Strip Deformation Switch (3)	I	S	I	II	I	--

NOTE:

(1) S -- Standard procedure followed.

(2) -- MIL-E-5272A procedure not valid - nozzle soaked in 1100°F to duplicate engine operating temperature environment.

(3) -- Immersion tested - switch immersed in water and 13 inches of mercury vacuum created to test end seals.

APPENDIX D

DESCRIPTION OF INERTIA SWITCH

By sensing the airplane's deceleration during a crash, the inertia switch serves as an overall crash detector, but not necessarily as a damage detector. Because crash deceleration can be quite low (1.5 to 2.0 g's), an inertia switch set this low could possibly be inadvertently tripped by a hard landing or even turbulent flight. With the initiating system concept of sensing localized damage for local inerting plus establishing positively that the airplane is on the ground for total inerting, a more positive ground contact detecting device, such as the strip deformation switch, was preferable to the inertia switch.

With the elimination of the inertia switch for the C-119 program, work on it was terminated. However, as the development had been completed and pre-production models were being fabricated, a description of the device is included here.

The basic mechanism of the inertia switch is a pendulum mounted to swing in a horizontal plane, but restrained by a torsion spring. Inertia forces, acting on the pendulum, overcome the spring restraining torque at the pre-set value and cause the pendulum to swing in a small arc, making electrical contact at the end of its travel. Using a torsion spring with a very low spring rate, and keeping the arc of movement small results in essentially constant restraining force throughout the pendulum's travel. Suspending the pendulum between precision ball bearings, run dry, kept friction low throughout the temperature range and provided the necessary ruggedness. To provide insensitivity to shock, the time required for the pendulum to swing through its arc was controlled by an air dashpot. The delay time was controlled both by length of arc and rate of damping. Both were made adjustable, the former by a moveable contact, the latter by a valve in the dashpot. Dashpot friction was kept low by a teflon coating on its piston.

This design, coupled with precision fabrication, permitted a high order of accuracy, both for the "g" setting and time setting.

The switch was set using an accurately calibrated centrifuge, and the time of pendulum swing was electrically measured. To insure maintenance of the setting under varying environmental conditions, the switch was hermetically sealed in its case after setting. Although environmental qualification tests were not completed, similar tests conducted during the development program indicated that the switch had a high order of accuracy and reliability.

Figure 1 - C-119F Airplane

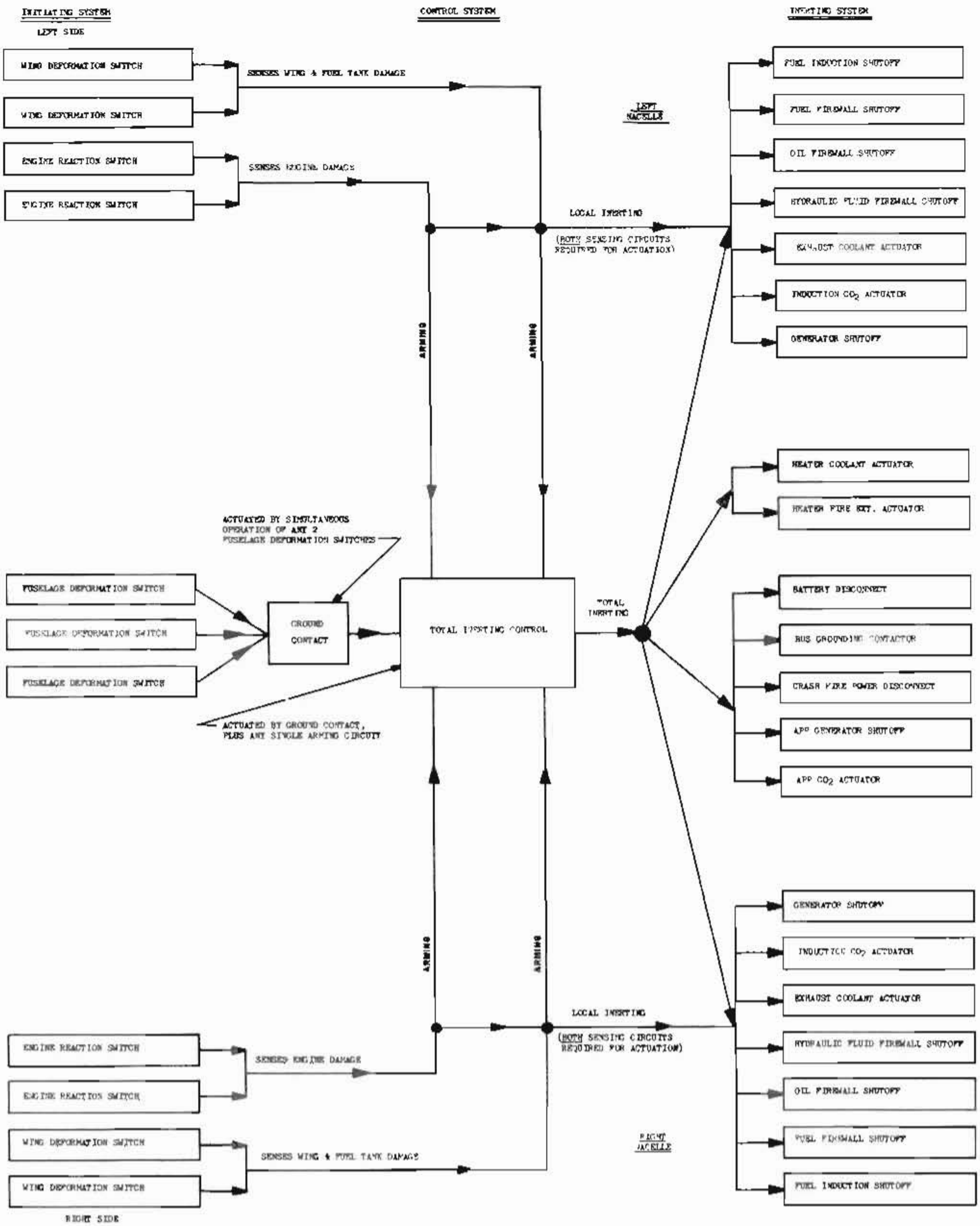

Figure 2 - Block Diagram of Initiating System

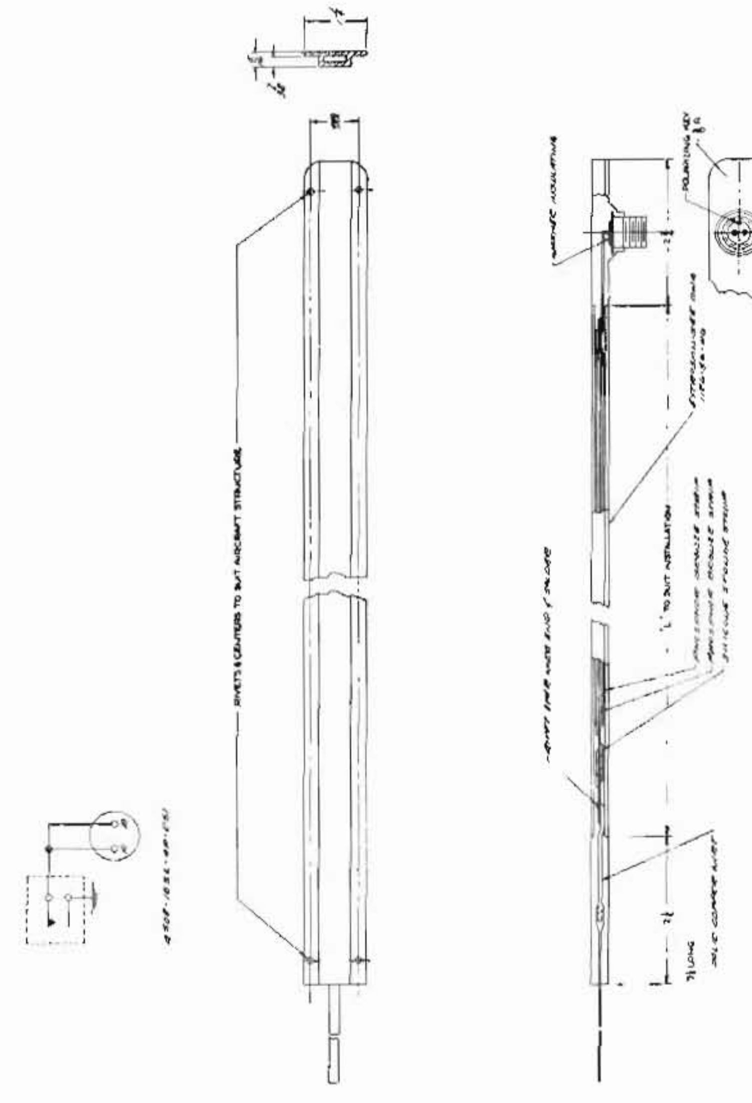

Figure 3 - Rubber-Strip Type Deformation Switch

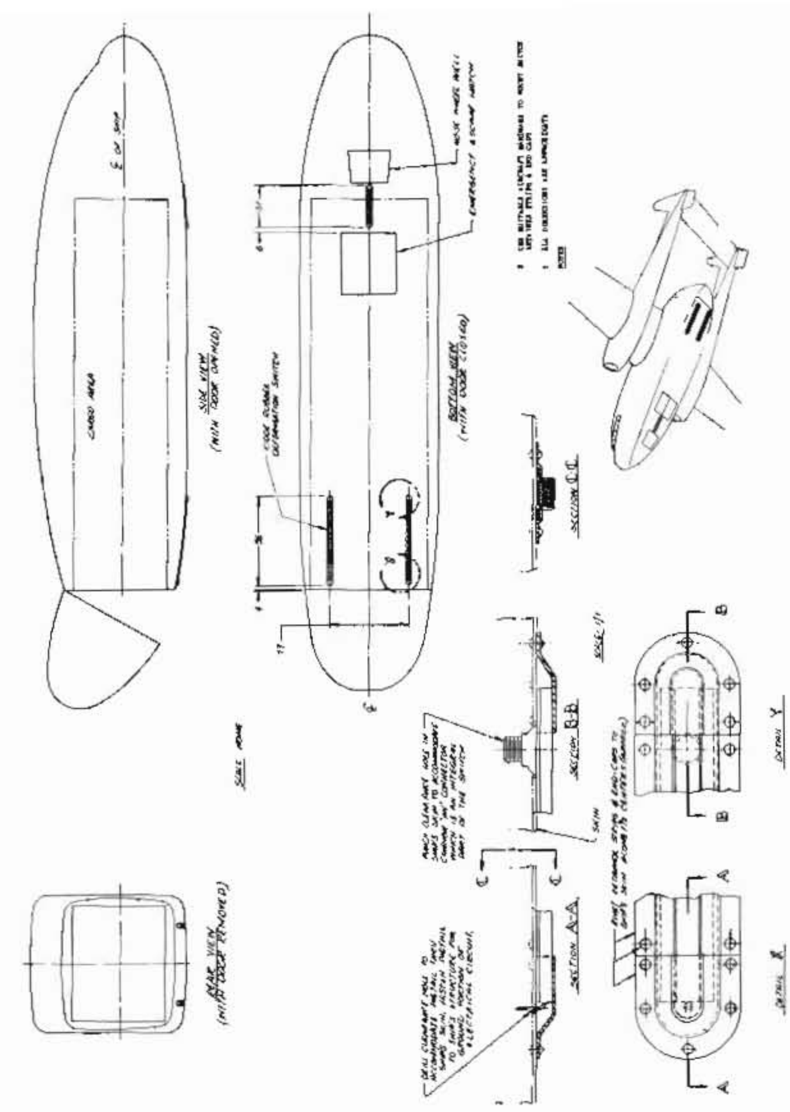

Figure 4 - Rubber-Strip Deformation Switch Installation

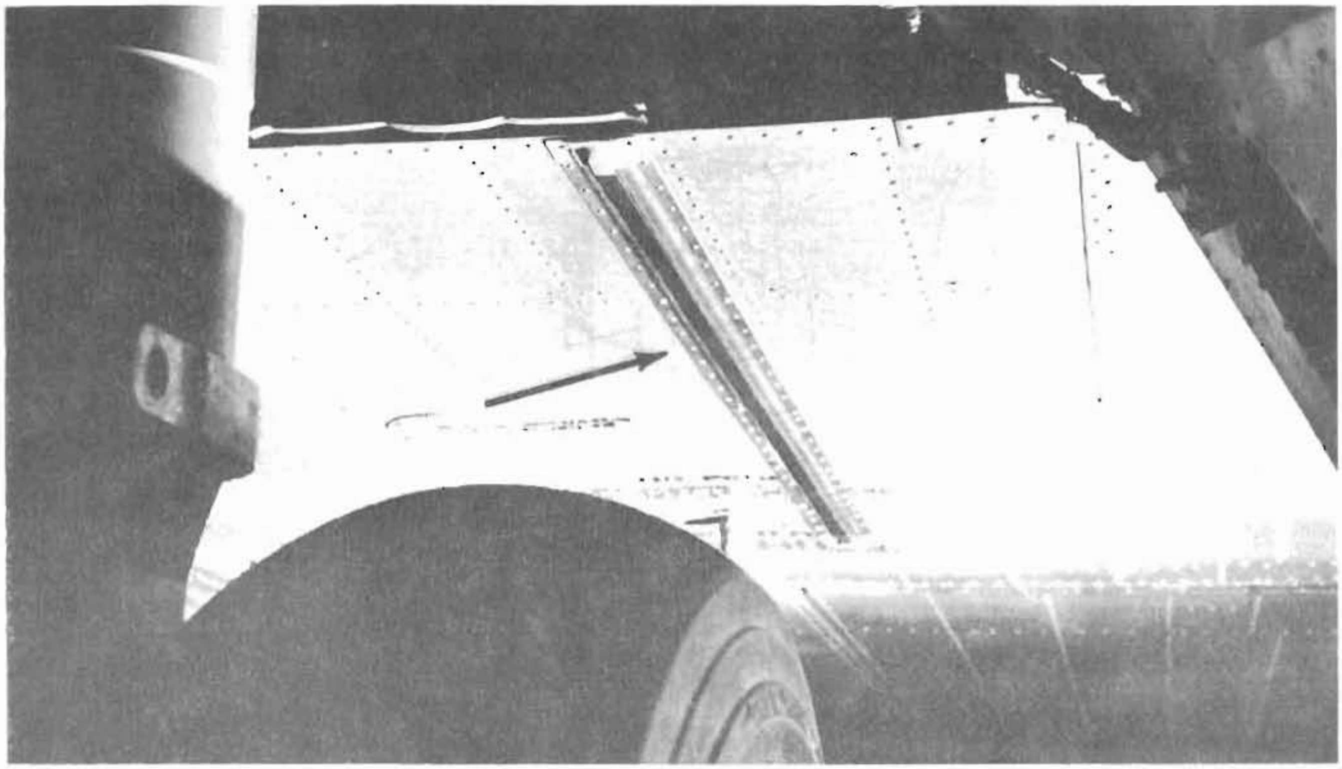

Figure 5 - Forward Rubber-Strip Deformation Switch

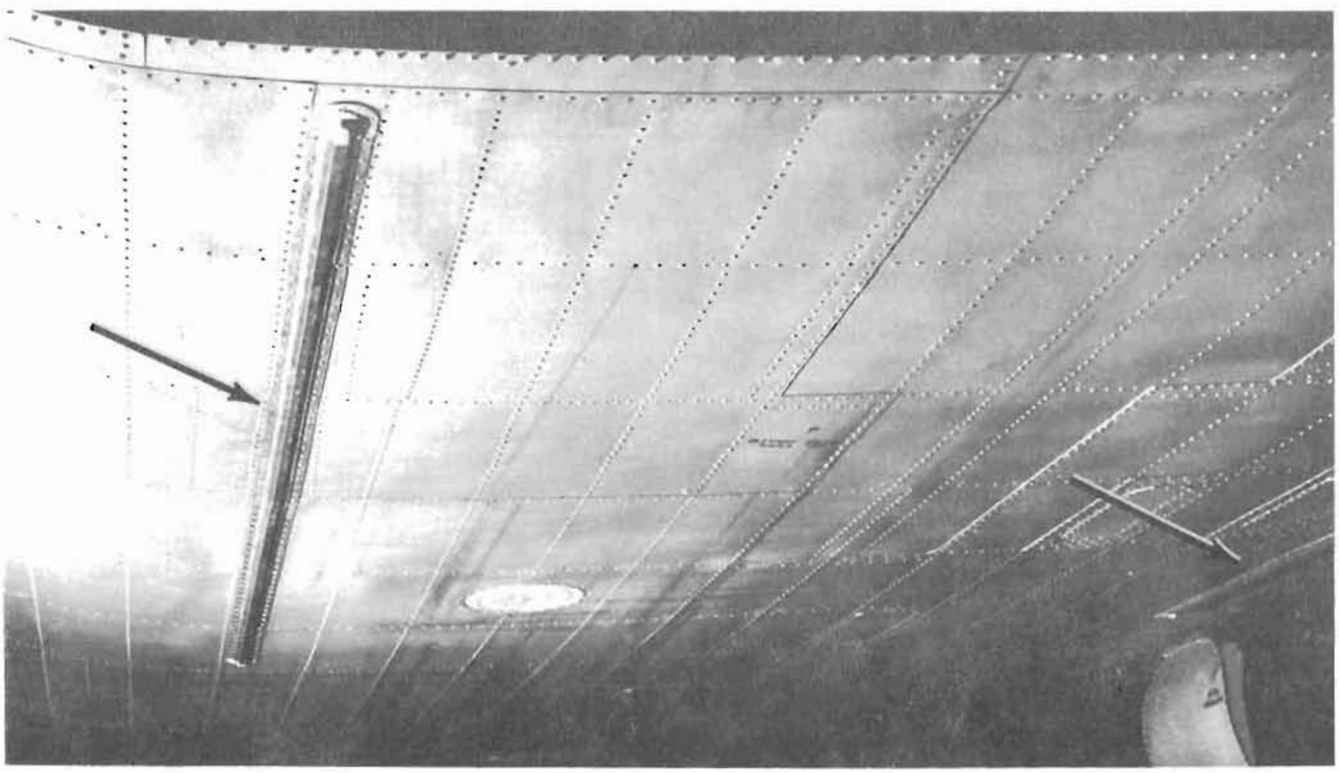

Figure 6 - Aft Rubber-Strip Deformation Switches

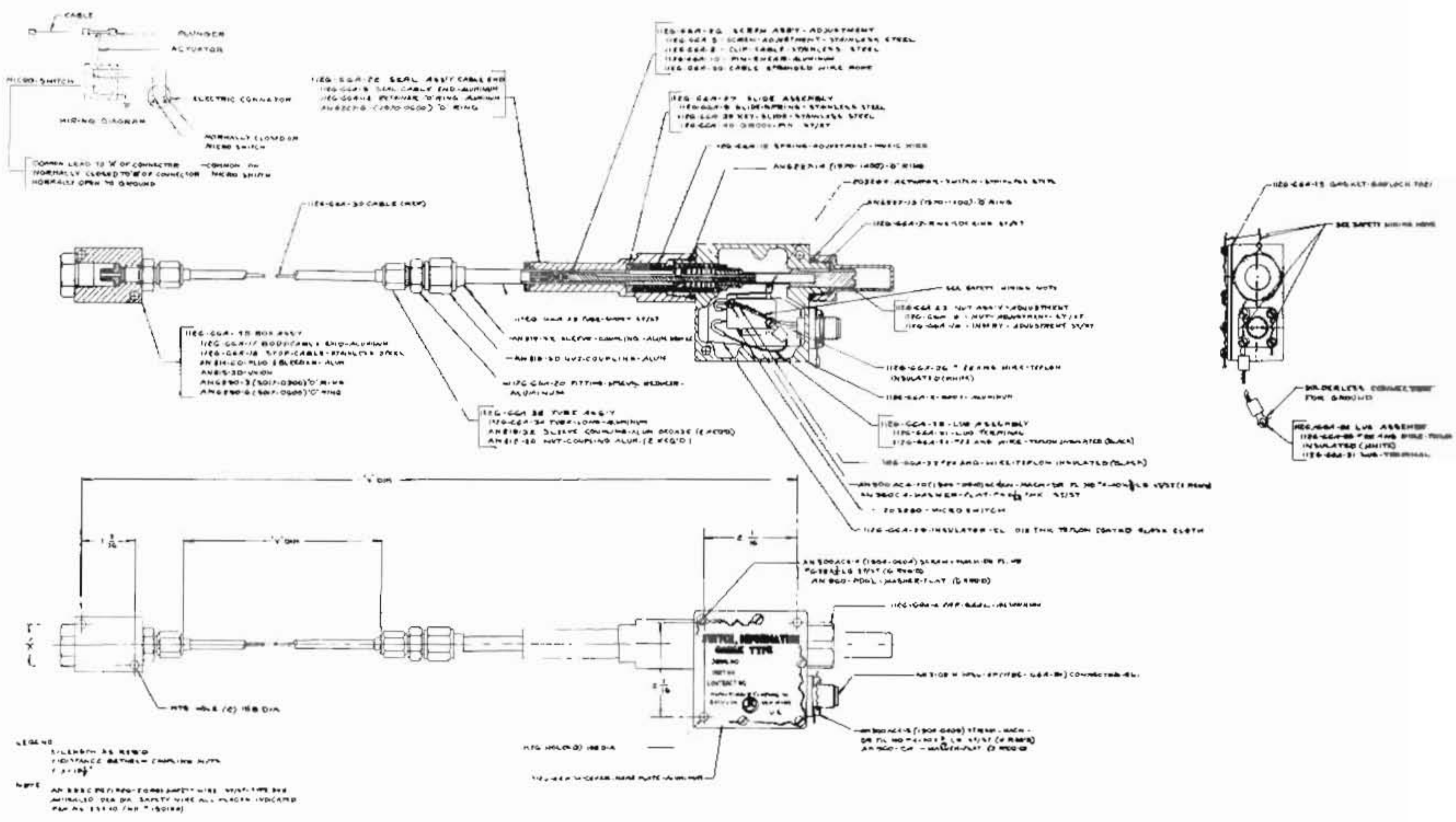

Figure 7 - Cable Type Deformation Switch

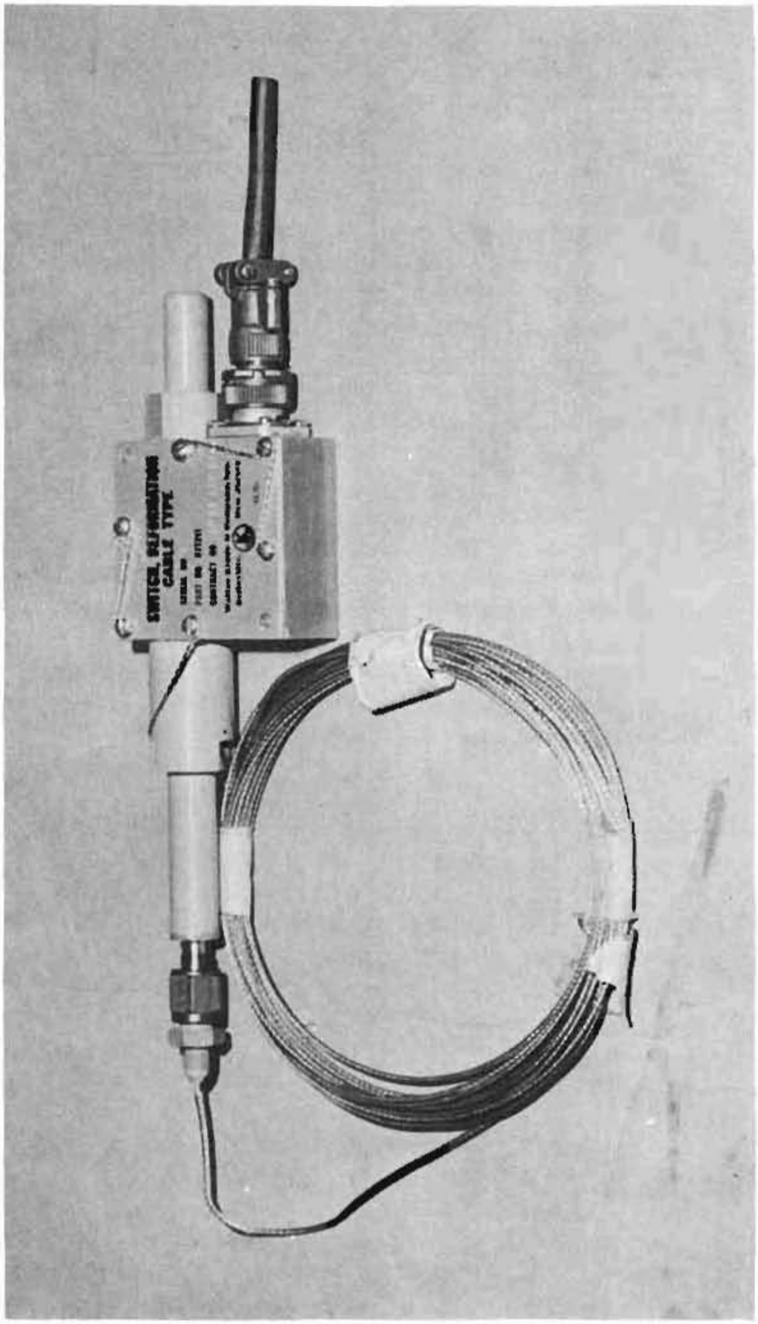

Figure 8' - Cable Type Deformation Switch

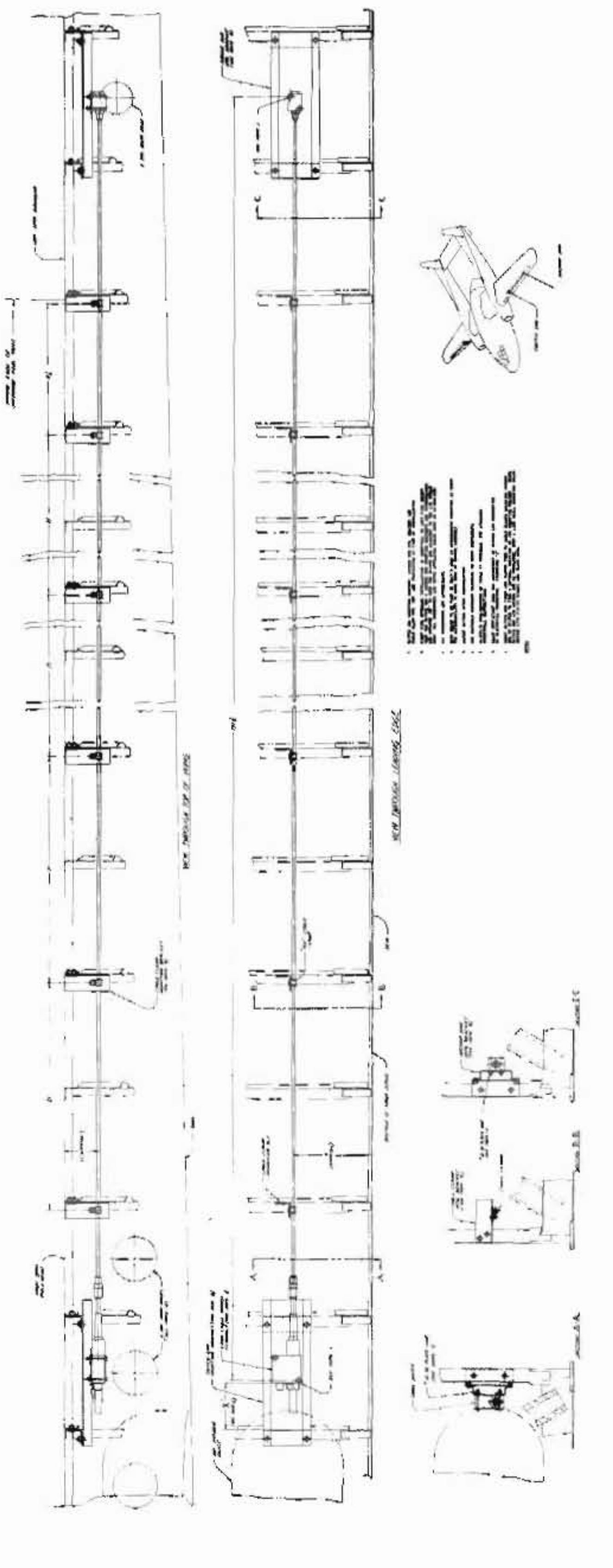

Figure 9 - Left Wing Outboard Cable Switch Installation

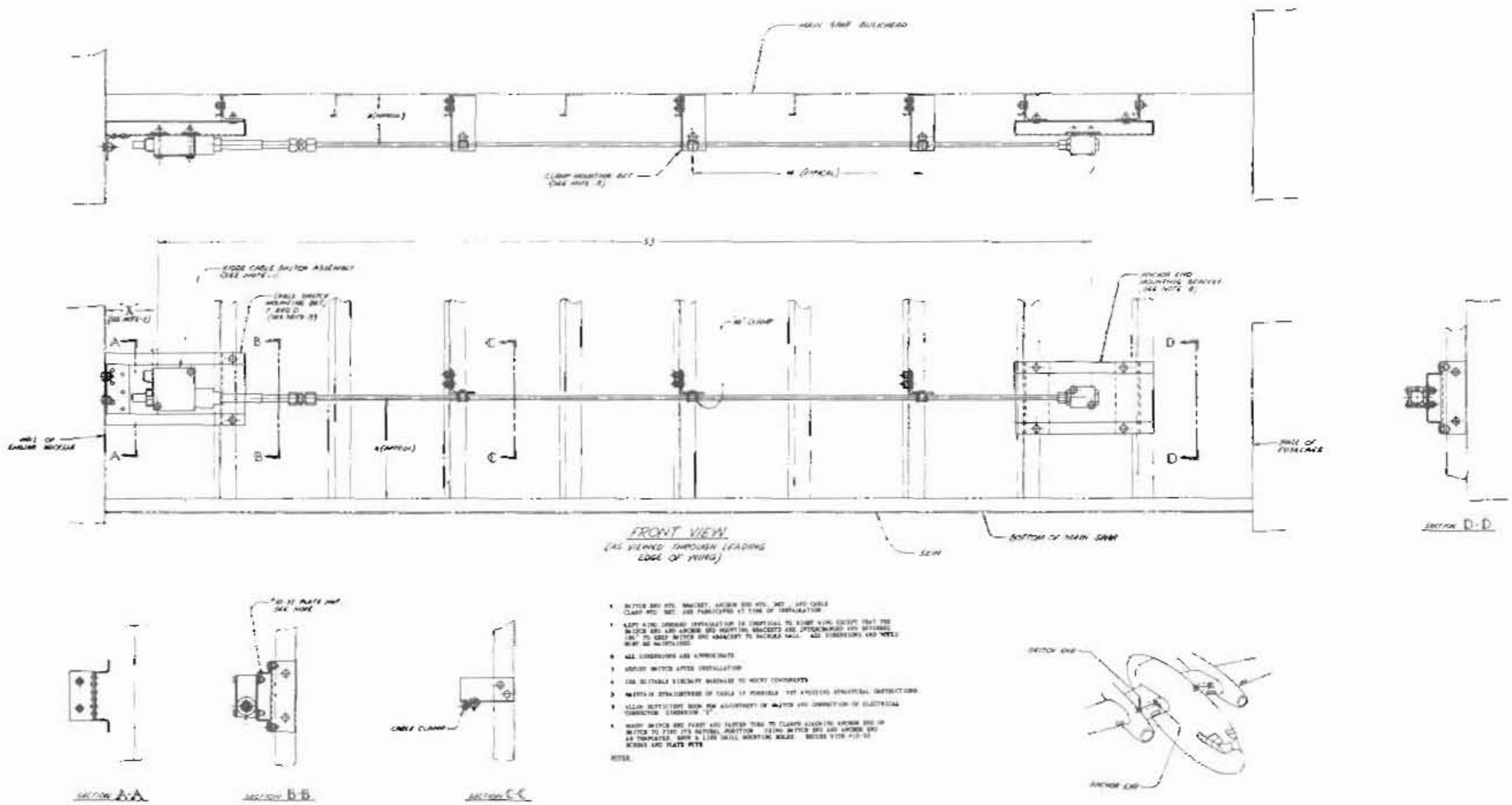

Figure 10 - Right Wing Inboard Cable Switch Installation

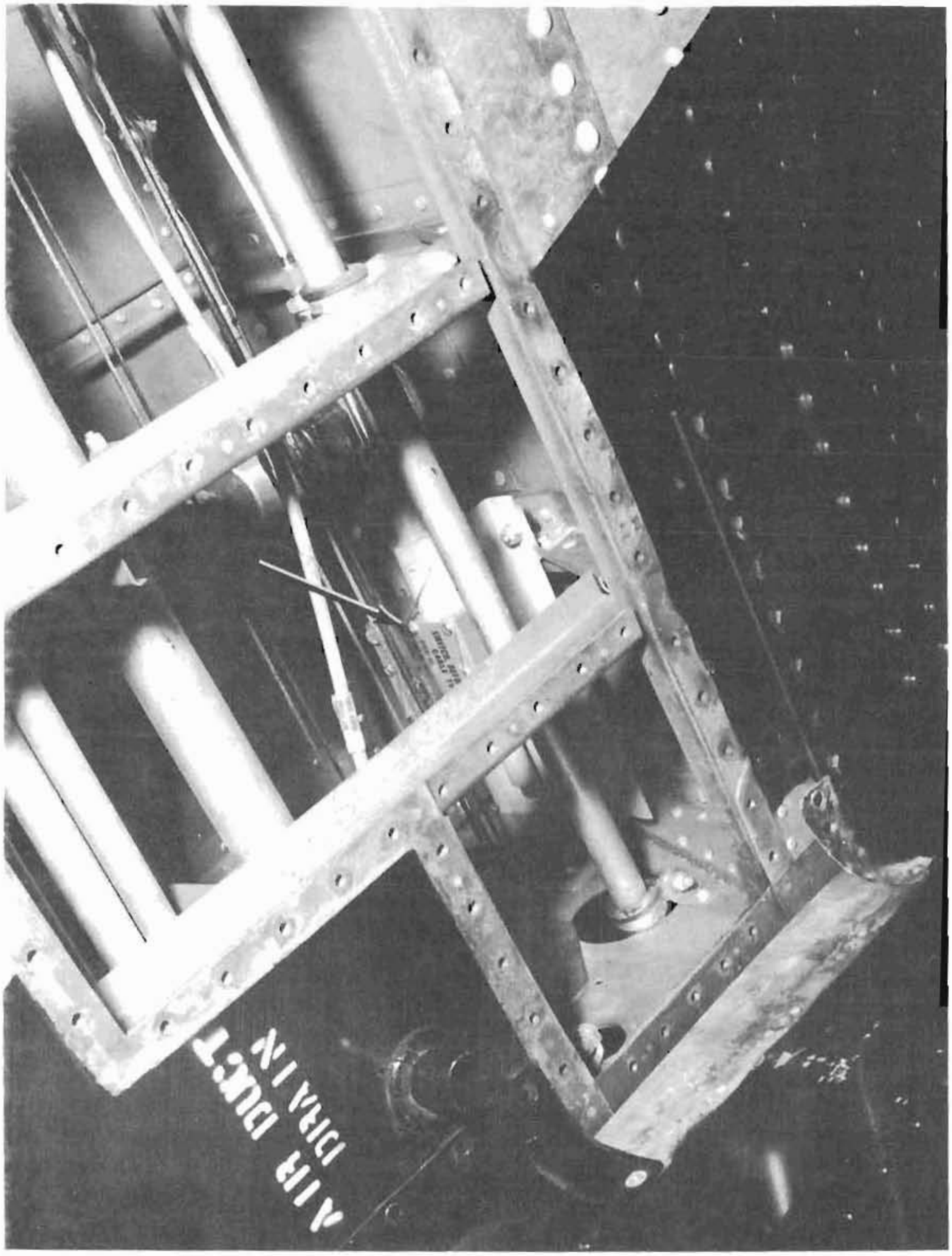

Figure 11 - Right Wing Inboard Cable Switch (Switch End)

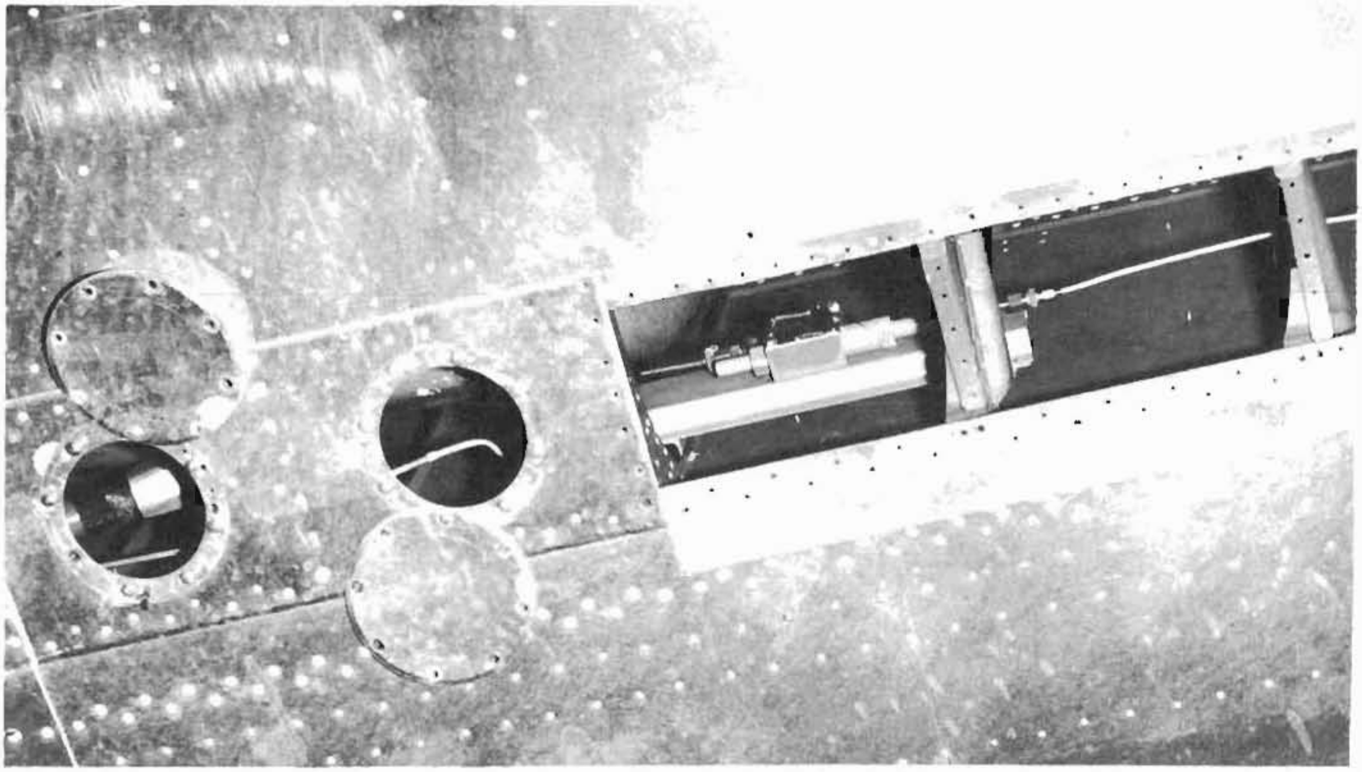

Figure 12 - Left Wing Outboard Cable Switch (Switch End)

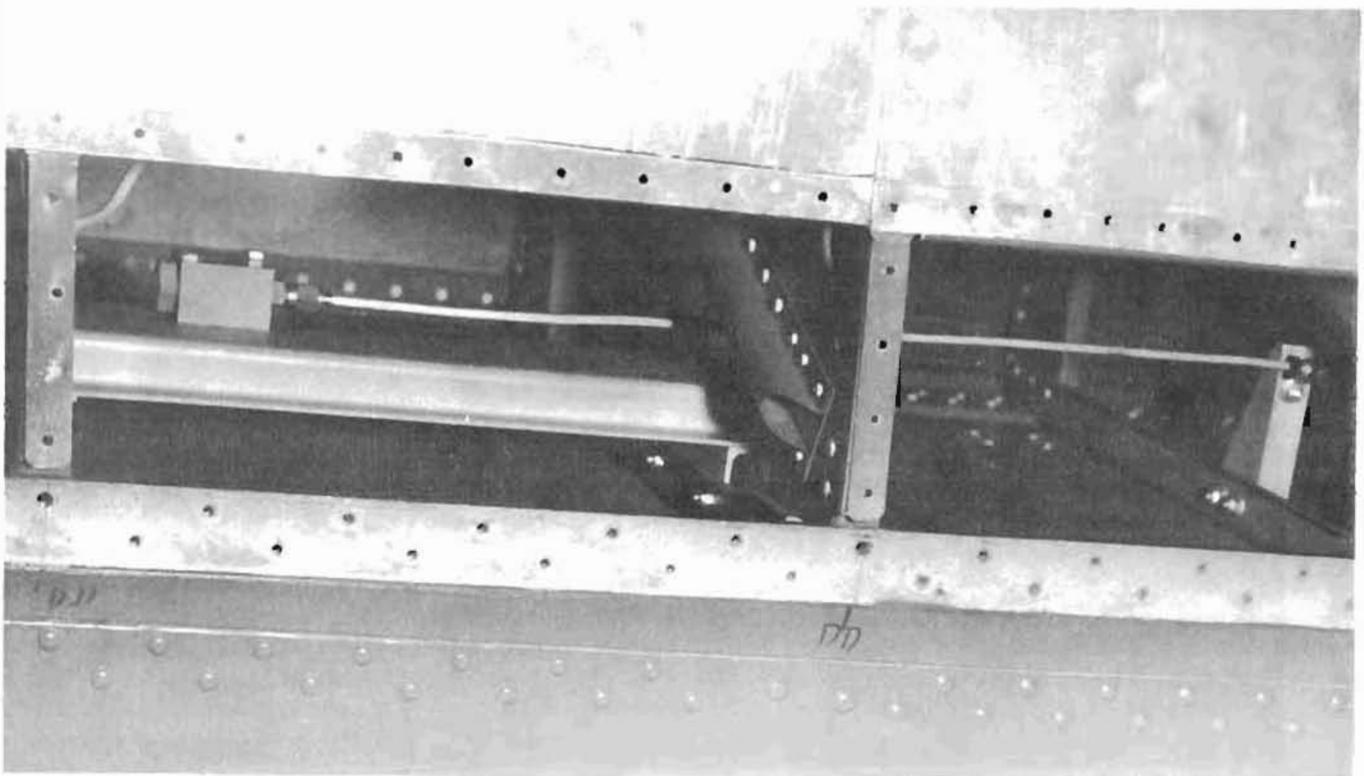

Figure 13 - Right Wing Outboard Cable Switch (Anchor End)

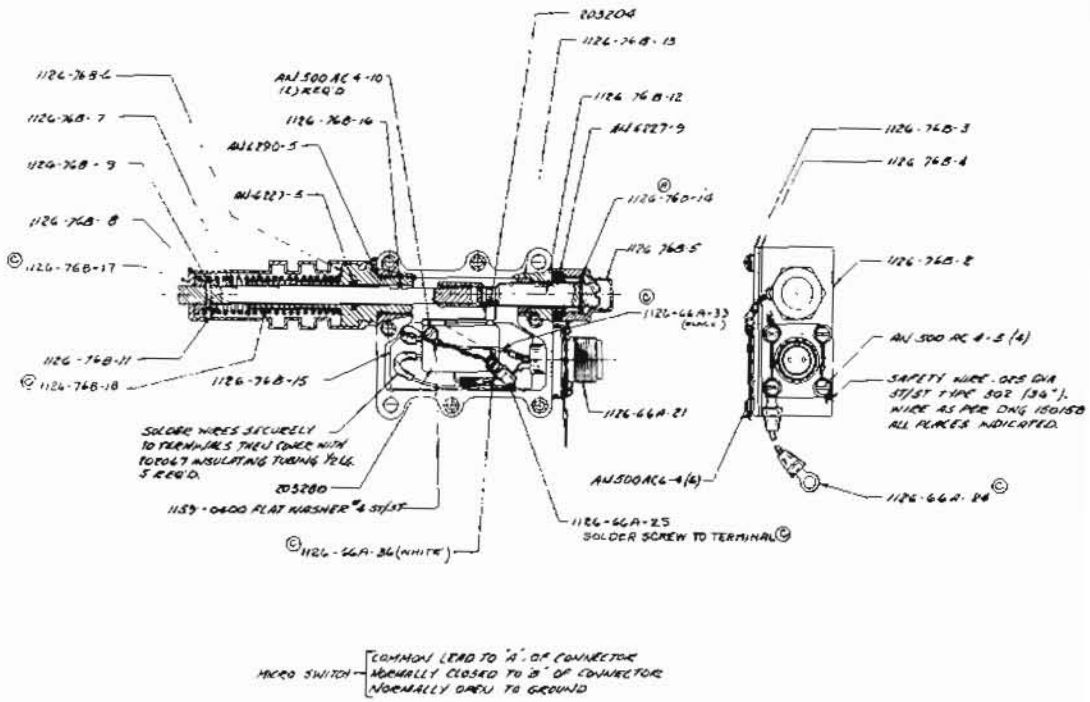

Figure 14 - Reaction Switch Assembly

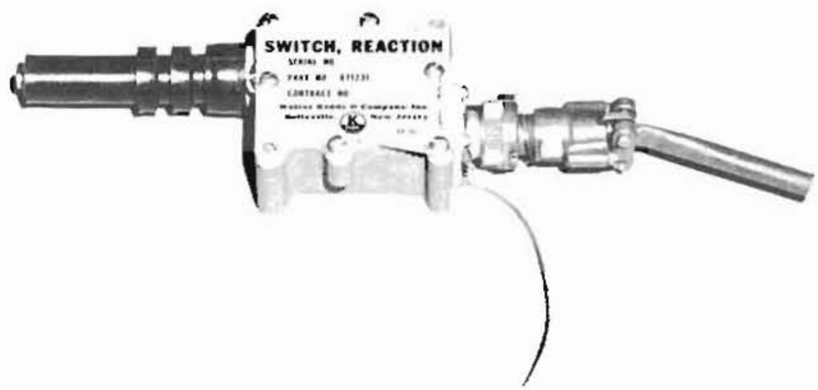

Figure 15 - Reaction Switch Assembly

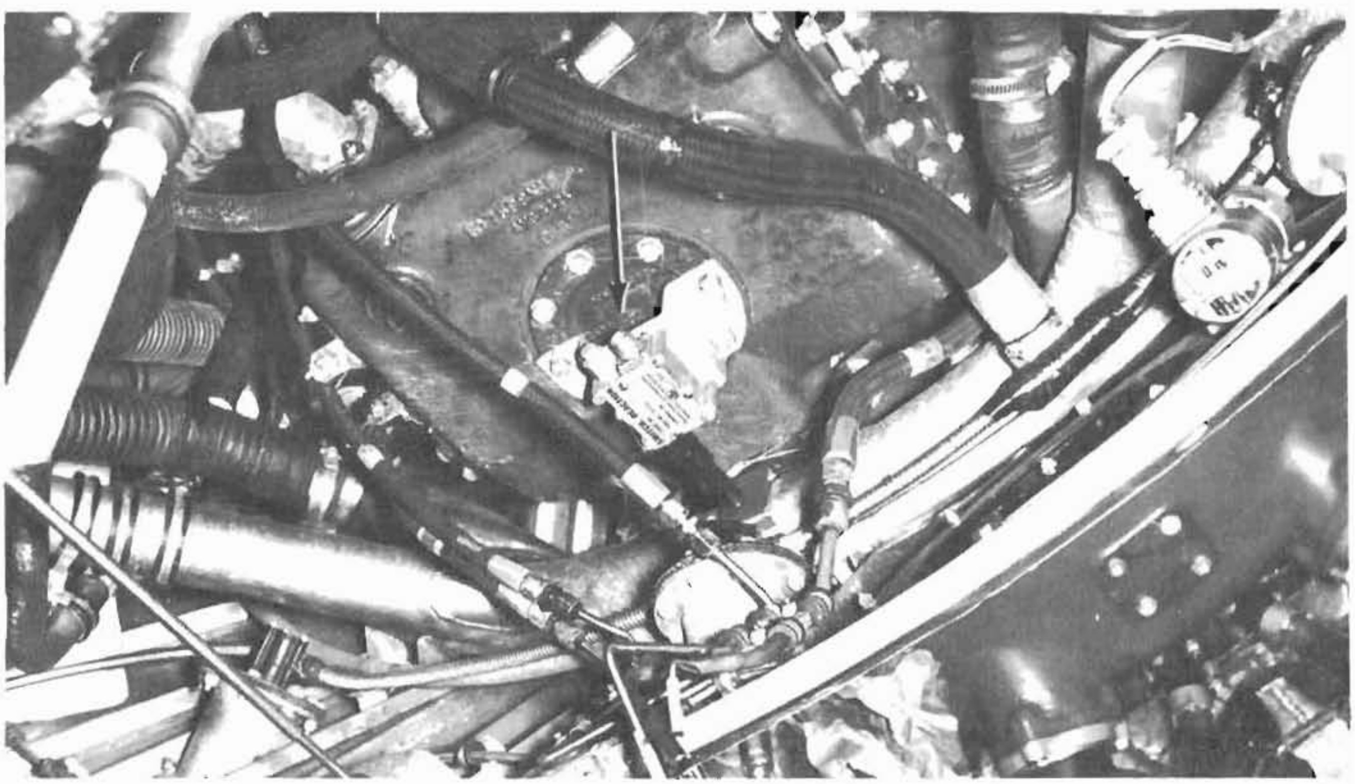

Figure 16 - Lower Reaction Switch (6 O'clock Position)

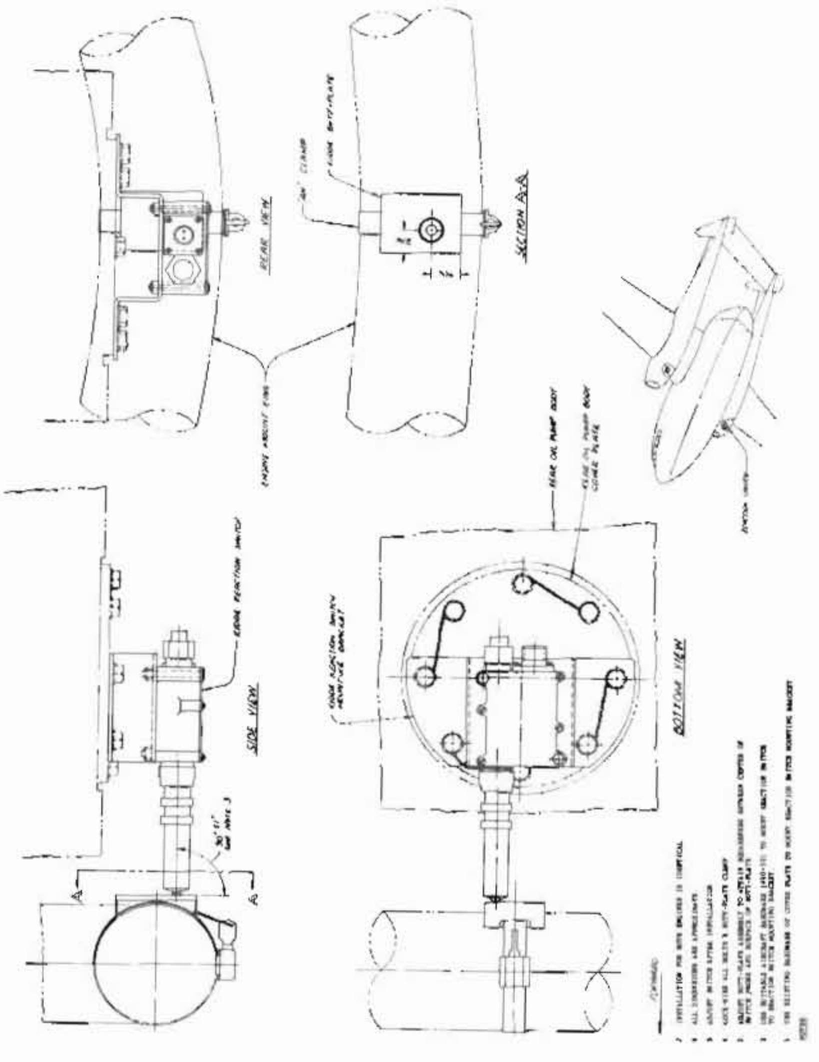

Figure 17 - Bottom Engine Reaction Switch Installation

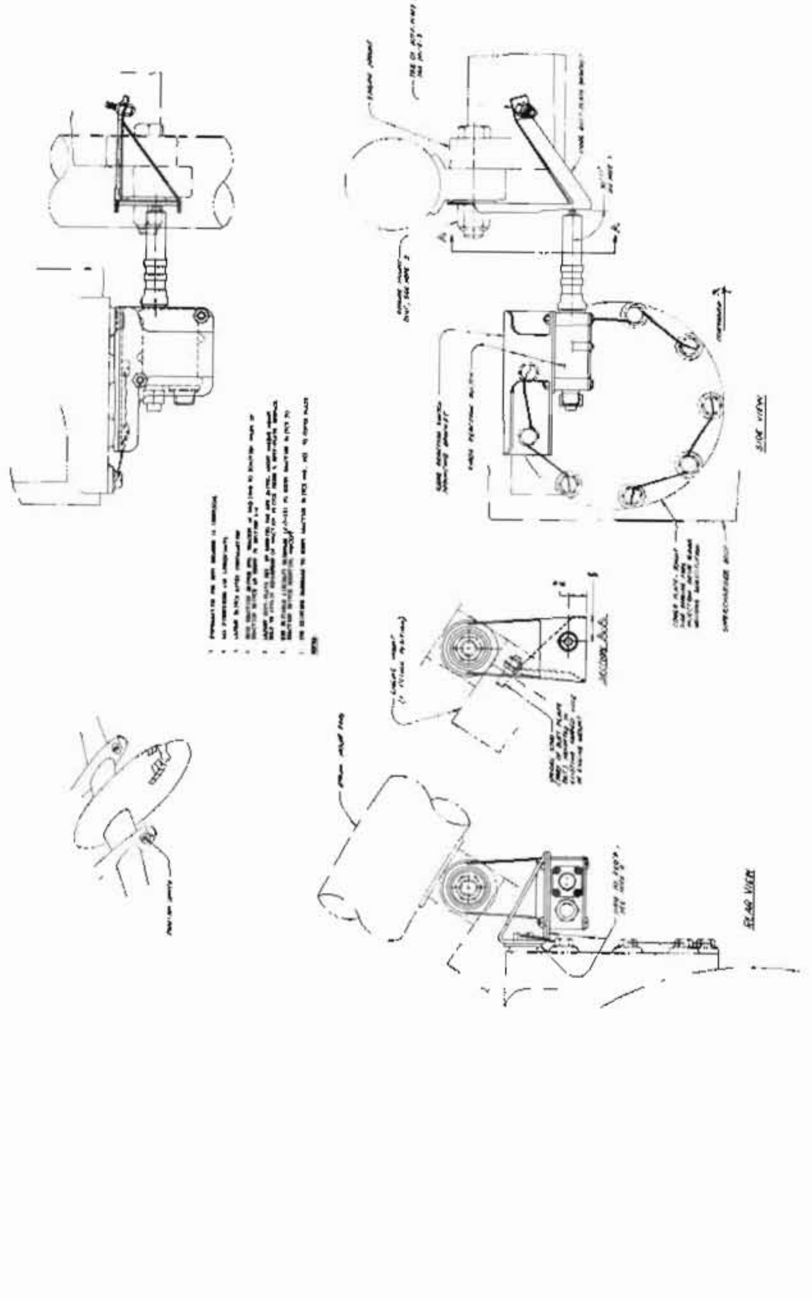

Figure 18 - Upper Engine Reaction Switch Installation

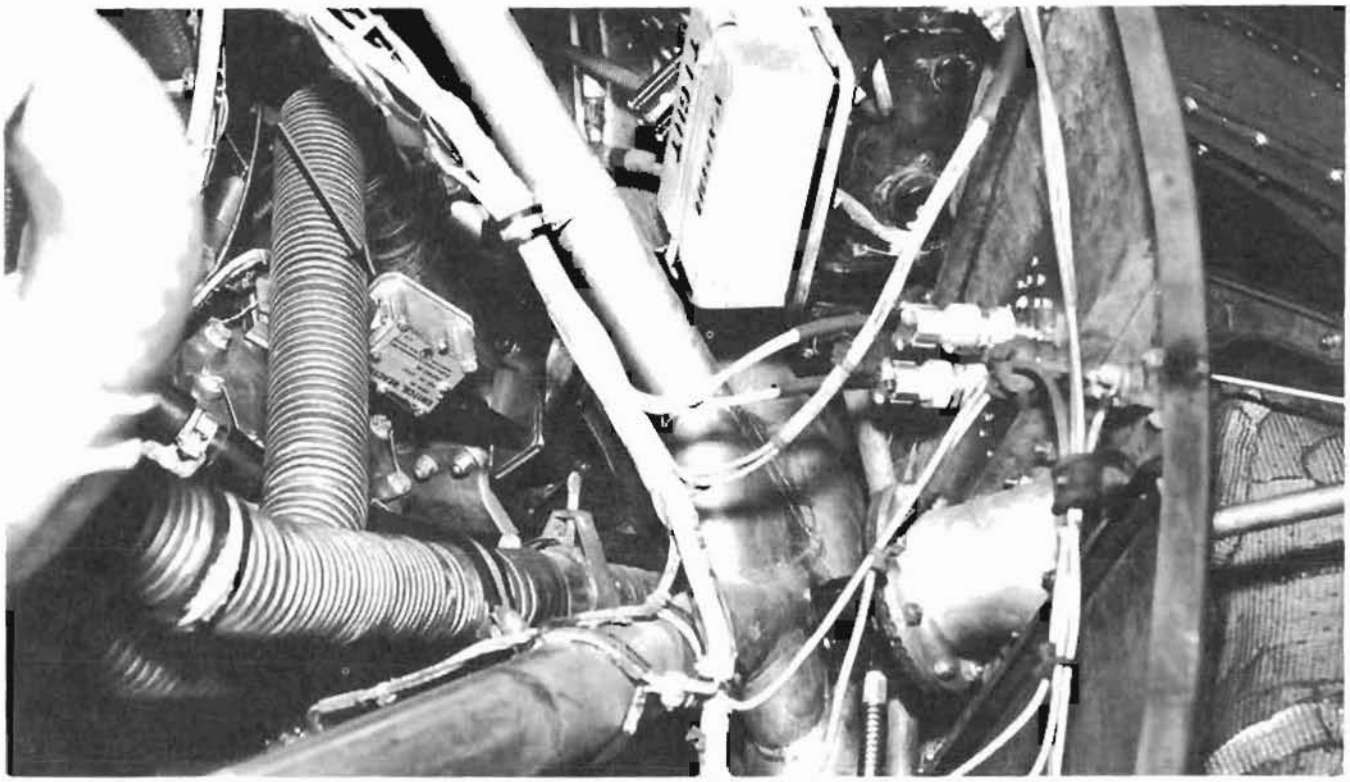

Figure 19 - Upper Reaction Switch (1 O'clock Position)

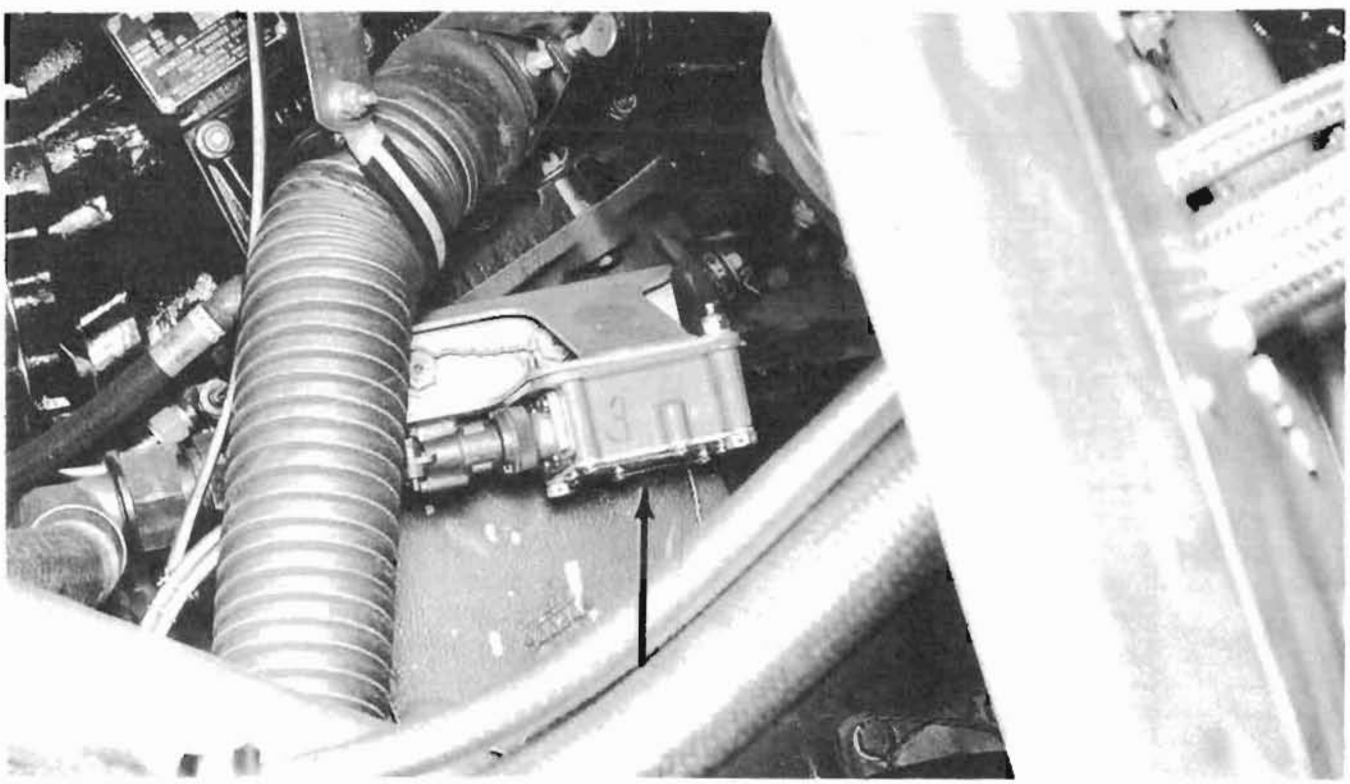

Figure 20 - Upper Reaction Switch Showing Mounting Bracket

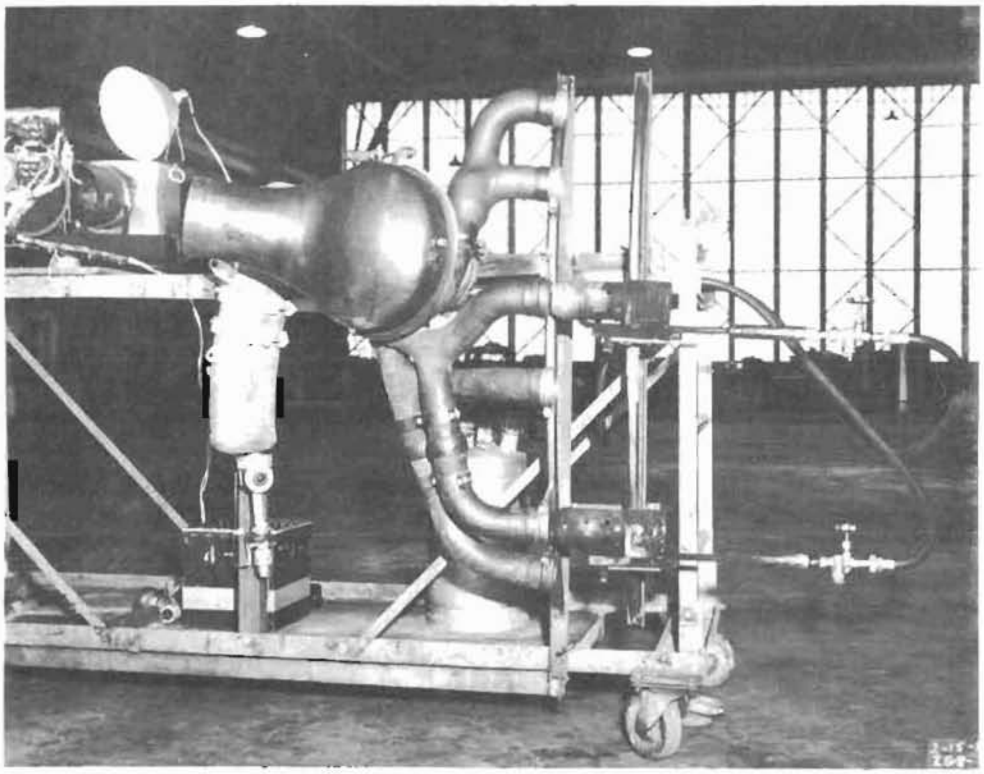

Figure 21 - No. 1 Turbine Section Exhaust System Mockup

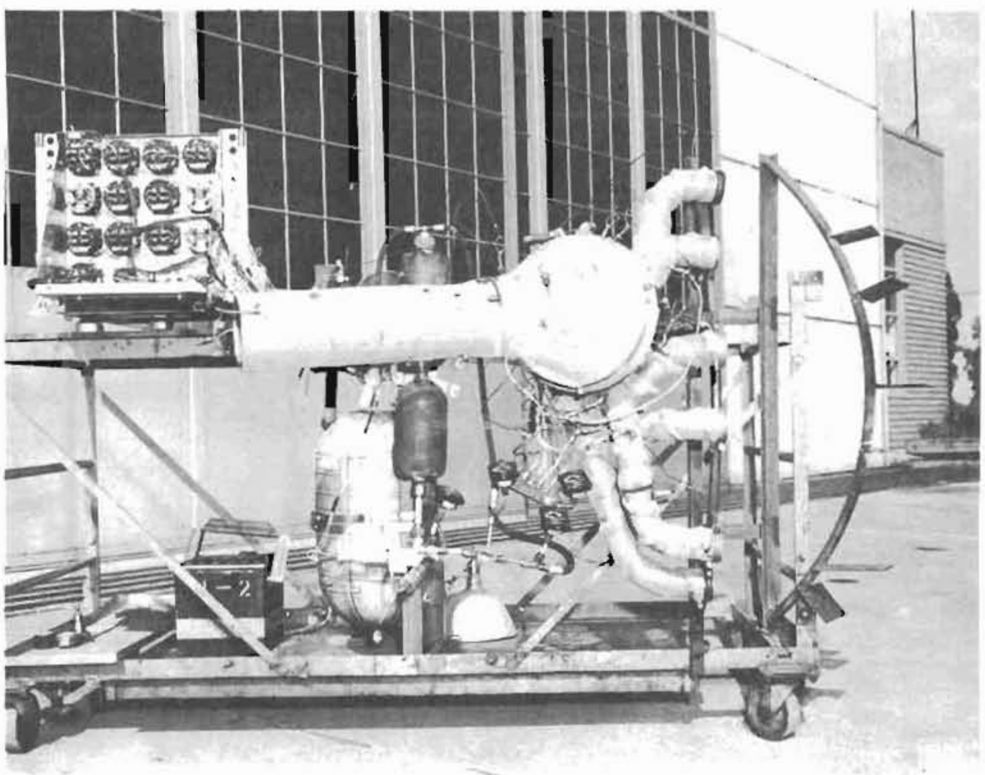

Figure 22 - No. 1 Turbine Section Exhaust System Mockup (Modified)

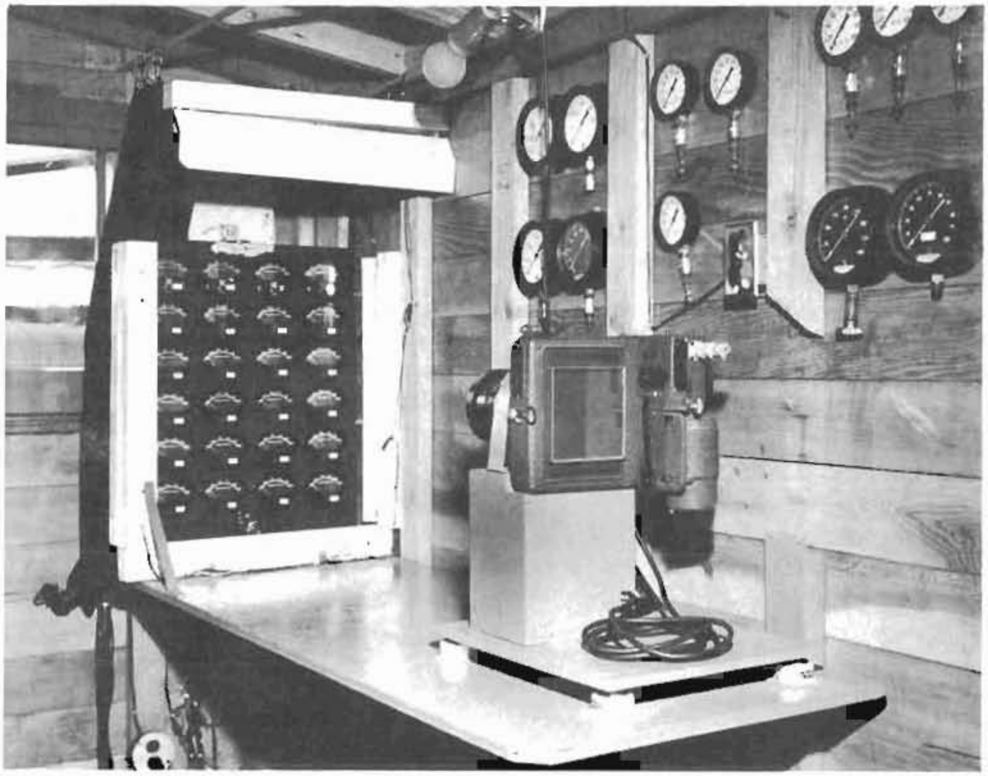

Figure 23 - Pyrometer Board in Control Building at Walter Kidde Test Grounds

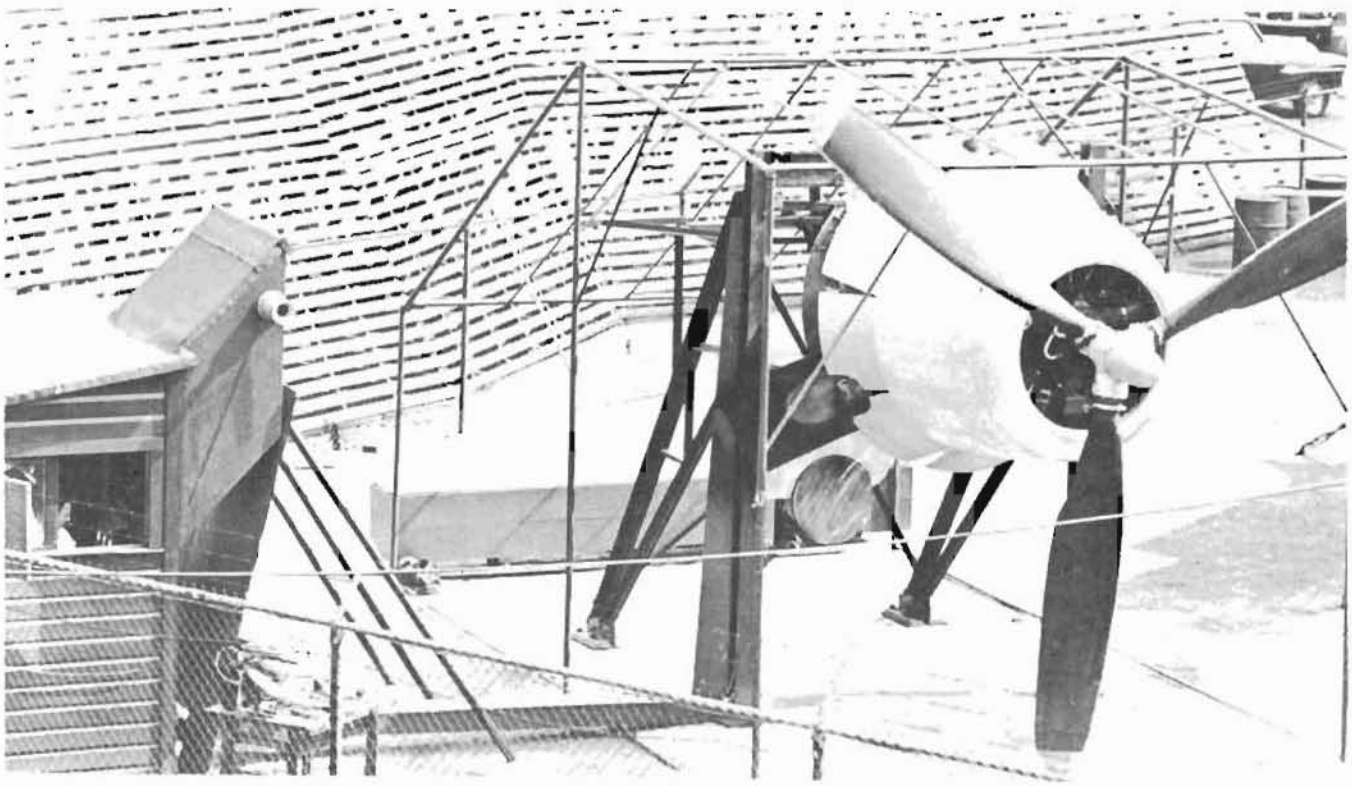

Figure 24 - R-3350 Test Package and Control Building at Walter Kidde Test Grounds

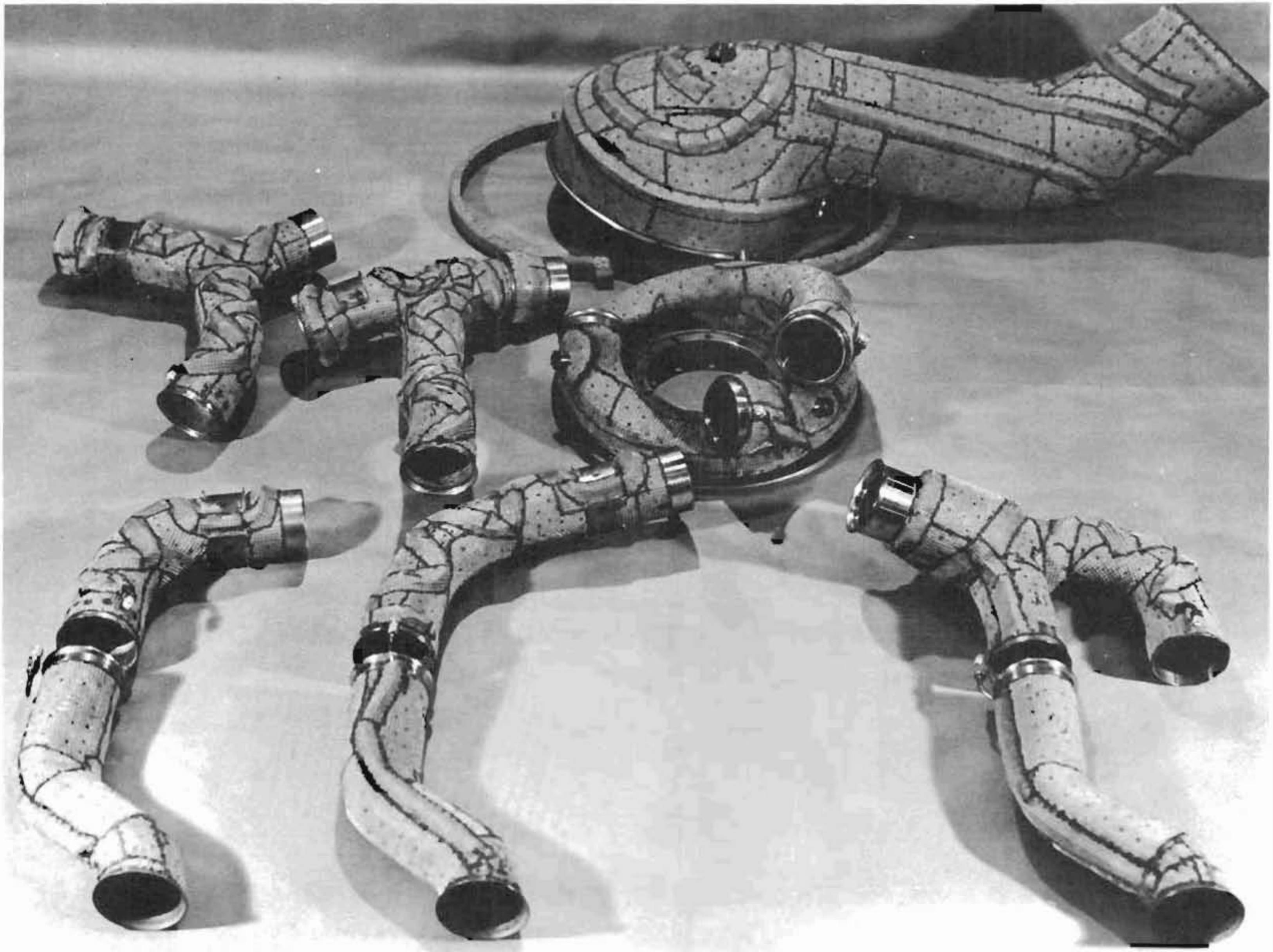

Figure 25 - Modified Exhaust System for one Power Recovery Turbine

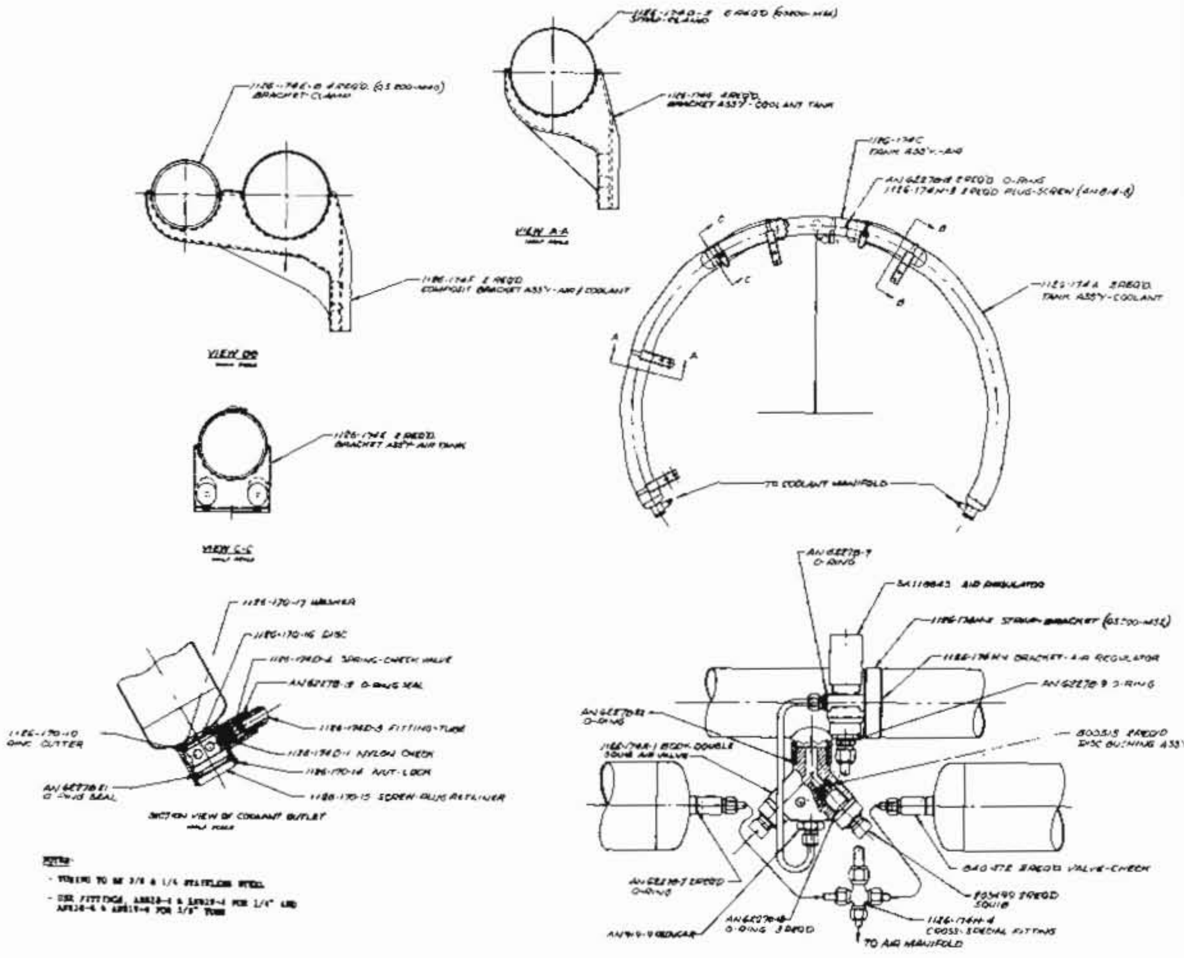

Figure 26 - Manifold Coolant Tank System