NONDESTRUCTIVE TESTING FOR SPACE APPLICATION

FEASIBILITY AND PRELIMINARY DESIGN STUDY

PHASE I REPORT

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Abstract

The need for development of nondestructive testing technology for in-space use is clearly defined by a thorough examination of current and proposed future space programs. Its use, and selection of the best method/methods of inspection considering ultrasonics, eddy current and radiography are also discussed. These are related to proposed in-space fabrication, repair and other functional requirements such as medical, preventative maintenance and scientific research aid. A preliminary design concept of an integrated ultrasonic - eddy current instrument with a detachable radiography unit is presented. Results of this study phase have indicated this prototype breadboard hardware, utilizing "off-the-shelf" equipment to be only the first step in what must be a parallel effort to other in-space fabrication studies and developments.

The space environments compatability and the required human engineering aspects are defined with a test plan for their evaluation. The demonstration test for the prototype unit is outlined, based upon use of a Hamilton Standard test subject suited in an NASA owned Apollo Block II suit in the 8 foot environmental chamber at NASA-MSC.

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1.0 INTRODUCTION

With the Gemini program nearing completion, and the lunar oriented Apollo program already under way, studies for programs considering longer life duration in orbit such as the "Manned Orbiting Laboratory" (MOL) are either in process or complete. These studies have resulted in several conceptual designs for extended duration space stations, all of which will, by necessity, be considerably larger than any to date. Because of size most of these space stations will have to be at least partially fabricated in the terrestial space environment. In-space fabrication studies, design and actual construction of in-space fabrication tools, (electron beam welders, minimum reaction power tool, etc) and space tool operation development are also either in process or nearing completion. Initial feasibility use and experimentation of these tools in-space will be accomplished on the Apollo Applications program. The products of these initial in-space fabrication experiments will be returned to earth for fabrication quality evaluation and determination of experiment success. Although mandatory from both an astronaut safety and structural integrity standpoint, on-earth inspection will obviously not be possible with actual in-space fabrication. The necessary in-space inspection concept must therefore be thoroughly explored. Nondestructive testing tools and techniques for in-space applications must be developed. This development effort must parallel the development of fabrication tools for space use if any significant degree of success is to be expected in the development of large manned orbiting space stations. Unfortunately little effort has been concentrated in this area to date. In addition to fabrication inspection, in-space nondestructive testing, equipment will definitely be required to determine the extent of space damage, (i.e., meteorite collision, potential docking damage, potential damage resulting from hard lunar landings, etc) and to determine the quality of subsequent damage repair. Other immediate and realistic applications of nondestructive equipment in space programs include areas such as preventive maintenance "in-space", medical use, and scientific research aid.

This program, "Nondestructive Testing for Space Application, Feasibility and Preliminary Design Study", has been conceived through recognition of the above indicated requirements, (i.e. astronaut safety, space station structural integrity space damage analysis and repair, "in-space" preventive maintenance, in-space medical use, and scientific research aid). The overall objectives of this program are to study and evaluate present and future space station concepts, structure, fabrication techniques, and potential in-space problem for the purpose of defining as specifically as possible present and future requirements of nondestructive testing in the space environment. Also an objective of this program is the preliminary design and prototype construction of applicable nondestructive testing equipment oriented toward the in-space utilization as defined by the study phase of the program. The scope of the preliminary design and prototype hardware construction includes consideration of all the various problems encountered when considering in-space applications, such as human engineering and materials-space environment compatability. It is expected that the results of this program will sufficiently define flight hardware requirements and problem areas to enable design and actual construction of flight hardware.

This report covers a three month Phase I, "Fabrication Study and Preliminary Design" effort of the total program. The period of performance was from 1 July, 1966 to 1 October, 1966. All effort in this phase was oriented toward requirements of NDT in-space. The effort is reported under the following major categories:

- Literature Survey
  - Fabrication Technique Study
  - Damage, Repair, and Preventative Maintenance Study
  - Other Application (i.e. Medical, etc.)

- NDT Method Selection
  - Multiple Selection
  - Single Selection
  - Ultrasonic Inspection
  - Eddy Current Inspection
  - Radiographic Inspection

- Preliminary Design Concept
  - Component Analysis
  - Qualified Components
  - Redundancy
  - Ultrasonic - Eddy Current Integration and Simplification

- Preliminary Thermal Analysis
  - General
  - Ultrasonic-Eddy Current Unit
  - Radiographic Unit

- Preliminary Vibration Analysis
  - General
  - Hard, Soft, Mount Considerations

- Packaging - and Human Engineering
  - Single-Dual Unit Concept
  - Ultrasonic-Eddy Current Conceptual Design
  - Radiographic Conceptual Design

Probe Design
- Multiple Head Concept
- Finger Tip Concept

Concept and Design Evaluation Testing
- Ultrasonics
- Eddy Current
- Ultrasonic-Eddy Current Package
- Radiography
- Radioisotope Package Operation
- NASA-MSC Demonstration

It should again be emphasized that all above categories are "Space-Oriented".

2.0 TECHNICAL DISCUSSION

2.1 Literature Survey - In-Space NDT Requirements

Increases in space flight activity, mission duration and space vehicle size that have progressed from Mercury to Apollo programs all point to future activities which will involve orbiting space laboratories, lunar landings and bases, orbital launch facilities and eventually interplanetary travel. The space vehicles necessary for such missions soon exceed the imposed aerodynamic restrictions of vehicle launch. To overcome these restrictions, both NASA and the Air Force are currently developing in-space fabrication techniques for expandable sectional space structures.

While it is true that completion of specific mission objectives is a measure of success, astronaut safety is yet the most important single factor in the space endeavor. The high reliabilities that are presently established on earth prior to launch cannot be sacrificed by an in-space welded or brazed joint which is not inspected for quality and reliability. With these requirements it is imperative that techniques such as current nondestructive testing be adapted for in-space use to perform these inspections.

The use of nondestructive testing equipment is not restricted to inspection of primary construction. Potential hazards such as space damage (i.e. Meteorite penetration) and even accidental damage (i.e. mismatch docking damage) must be considered. NDT equipment would be invaluable in assessing the extent of such damage and the quality of "its" subsequent repair. Nondestructive inspection is required to provide the needed reliability assurance. Results of numerous research programs conducted in the past ten years have indicated that degradation and deterioration of some materials is to be expected in the space environment during long duration missions. Nondestructive inspection equipment will be invaluable in evaluation of this degradation as preventive maintenance. As previously indicated, the same NDT equipment can be extended for use in such areas as medical diagnoses and scientific research aid.

This literature search was performed to define specific requirements of in-space nondestructive testing and to establish associated guide lines. More specific equipment requirements have been established by considering proposed fabrication techniques and expected defects that the equipment must be capable of determining. The study has been categorized in four major areas: (1) primary construction (2) damage and repair and (3) preventive maintenance, and (4) medical and other uses. In considering primary construction, in-space fabrication techniques were reviewed coupled with material investigations, proposed space station configurations, structures, and joint design. The interdependence of these areas and their effects on in-space nondestructive testing equipment are clearly demonstrated in Figure 1. Potential damage modes were investigated and the use of in-space NDT equipment for accurate damage analysis is demonstrated. The utilization of fabrication techniques in conjunction with NDT for repair operations are also included. Effects of the space environment on degredation of materials and use of NDT to assess

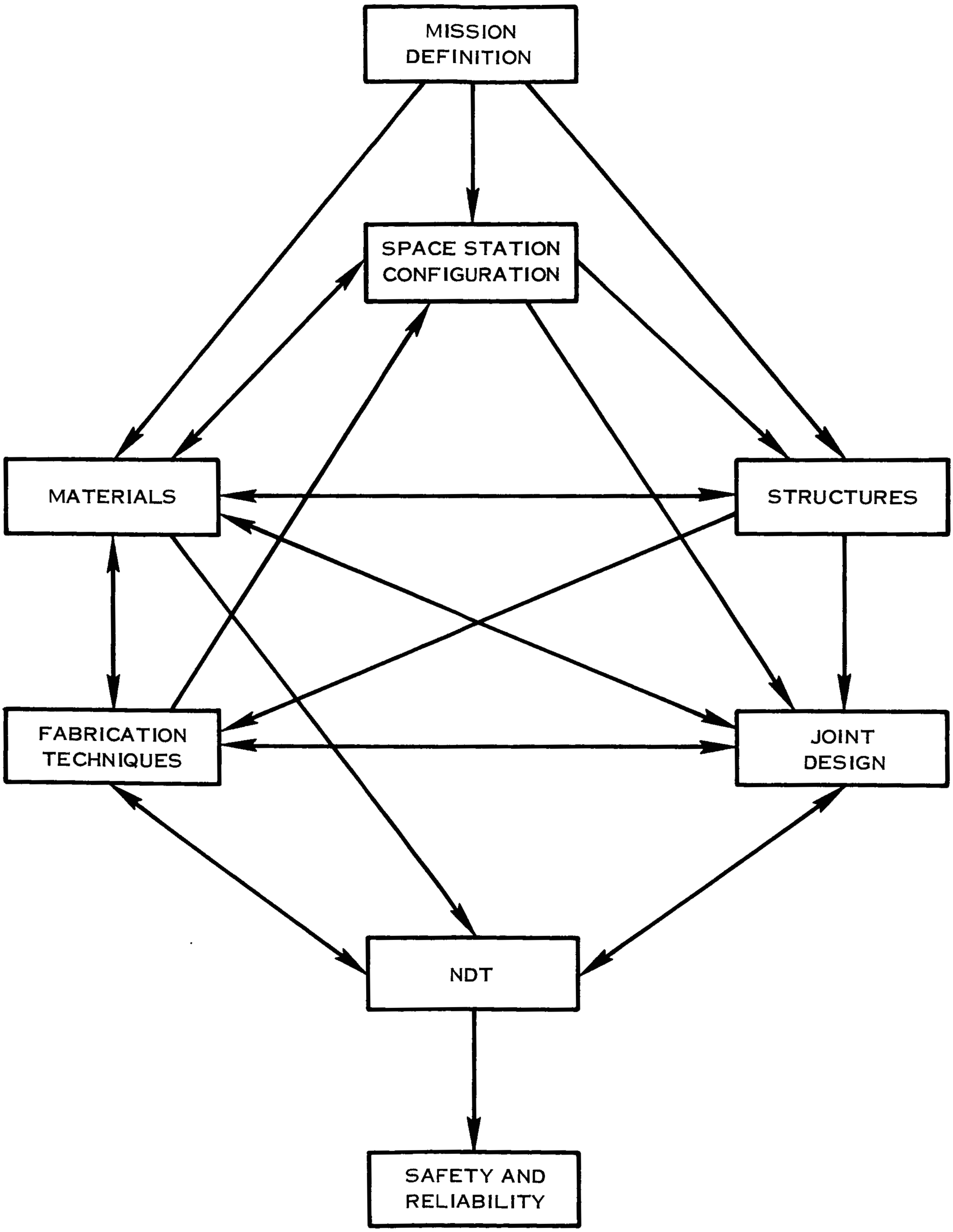


FIGURE 1. INTERRELATIONSHIP OF VARIOUS ASPECTS OF SPACE EXPLORATION

the extent of these effects is considered.

2.1.1 Primary Space Construction

Proposed Space Station Configurations

The broad mission definition of continued exploration of space is used as a starting point for primary space station construction. Design of future space stations must be considered, with few exceptions, as still conceptual. There designs are expected to be modified as the effects of the space environment on specific missions, physiological, psychological, and the biological requirements of man-in-space are more clearly defined. A program oriented toward such definition is the Manned Orbiting Laboratory (MOL). The prime mission of MOL is to investigate the behavior and capabilities of man-in-space for extended duration flights. The results of MOL experiments, specifically the effects of zero gravity on man for extended time periods, will strongly influence future space station designs.

Two configurations for large first generation space stations are considered. These are the cylinder and the spoked toroid (or hexagon). If mission requirements do not include a need for artificial gravity, the cylinder configuration will undoubtedly be selected. The toroid (or hexagon) configuration will be considered only if it is necessary to provide artificial gravity. This will be accomplished by rotation of the toroid at a speed sufficient to create the necessary tangential force which will create the sensation of gravity.

"In-orbit" considerations impose only minor geometric restrictions on non-rotating space stations. Geometry will be dictated by compatability with the launch vehicle which restricts shape to a cylinder. The first MOL, the only space station in the hardware stage, will be a cylinder with a 10 feet diameter and a 41 feet length which will conform to the Titan 3C launch vehicle. The primary cylinder will be of rigid wall construction. Should additional volume be needed for experimentation or hanger area, an expandable structure may be used. Work done by the USAF during the past 4-5 years has shown that expandable structures are currently within the state-of-the-art capabilities of industry.

Primary construction (in-space fabrication) on this type of space station is expected to be very limited. Possible in-space fabrication and subsequent in-space NDT includes attachment of solar cell panels, antenna structures, and probe booms to the exterior of the cylinder. Experiments will consist of construction and inspection of large structures that cannot be launched in final configuration.

As earlier stated, if it is deemed necessary to provide artificial gravity, the space station must be rotated. Human factors studies have indicate that major work areas must be more than 60 feet from the center of rotation. During a study by NASA - Langley Research Center, configurations considered included cross, rim, flywheel, tumbling cylinder, spinning cylinder, in-plane modules and axial modules. Based upon inherent stability, size and usable space, the configuration considered optimum was the flywheel, (spoked toroid or hexagon). A totally expandable flywheel shaped space station was considered but subsequently discarded due to; (1) equipment would have to be stored in the hub during launch and moved to the rim after inflation to provide the necessary spin stability, and (2) insufficient micrometeoroid protection could be provided.

A partially rigid-partially "inflatable" hexagon configuration was initially considered in a study of rotating space station configurations, and an applicable systems analysis conducted by North American Aviation for NASA-LRC. The final design was an all rigid hexagon configuration with center hub and spokes shown in Figure 2. The six sides of the hexagon can be hinged and folded into a cylinder where the axes are parallel to provide compatibility with launch. The sides would automatically be deployed once in orbit. The hub has facilities for a zero gravity laboratory and for re-entry vehicle docking. The spokes are telescoping walkways or laboratory areas.

The feasibility of a semi-rigid "expandable" (telescoping) structure has been demonstrated by the Martin Marietta Corp. in a contract with the USAF. Each section of the hexagon is expected to have its own power supply and environmental control system. The sections will be separated by air locks. In the event damage occurs to any individual section, the damaged section can be isolated from the remaining station until repairs are completed, inspected and qualified for use. In-space assembly and subsequent inspection will be more extensive in this configuration than that of the cylinder.

One of the more critical problem areas of large space stations of long duration is that of leakage. Hermetic sealing of space station interfaces is a recognized necessity. Methods of joining these interfaces have already been studied by NASA and the USAF. The six corners of the hexagon require hermetic seals as will the telescoping interfaces of the spokes. Hermetic sealing will also apply to the attachment of spokes for the rim. These joints are structurally loaded members and will require subsequent in-space NDT to assure astronaut safety.

To provide additional space "in-space", the Air Force has proposed the use of non-rigid and/or semirigid expandable structures. It is expected that some in-space fabrication and subsequent NDT inspection will be required to successfully accomplish this goal. The current NASA approach to additional space station volume is to utilize spent stage fuel tanks. An airlock is presently being developed to couple the spent S-4B stage to an Apollo spacecraft. Little primary construction is expected on the S-4B stage with the availability of the Apollo hardware. Extensive construction inside the fuel tank is however foreseen. Such construction again indicates the need for in-space fabrication and subsequent inspection. Linking several S-4B stages together to form a large complex space station is also under consideration.

Two configurations have been proposed that require extensive in-orbit fabrication and subsequent inspection. They are referred to as the "extended dumbell" and a modification known as the "Pseudo - Geogravitational Vehicle". The extensive fabrication and inspection

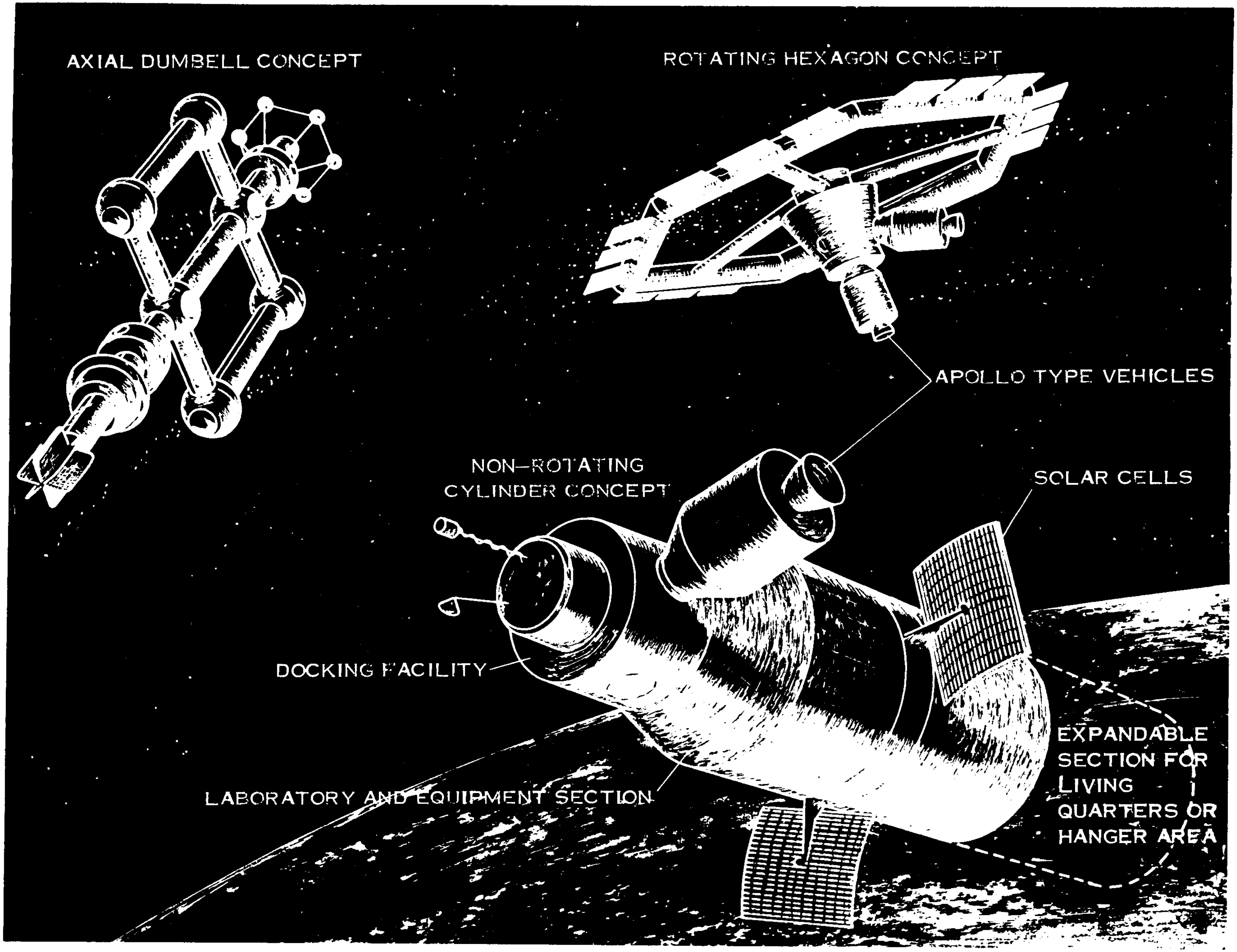


FIG. 2 VARIOUS POSSIBLE VEHICLE CONFIGURATIONS

are due primarily to the large size and complexity of these proposed concepts.

Applicable References: 2, 28, 25, 41, 42, 44, 47, 48, 49, 50

Proposed "In-Space" Structure Types

Space station configurations are influenced primarily by mission requirements, however materials and fabrication techniques also have an effect. The ultimate configuration will indirectly affect the required nondestructive testing through structures, joint design and fabrication techniques as shown in Figure 5. The structures concepts most often encountered in primary orbital construction are walls, tubing duct work, and external appendicies, (i.e. antenna supports, solar cell panels, radiators, etc.).

Walls of a space stations have the same broad functional requirements of any typical wall in that it must contain the desired environment while protecting the occupants and contents from an undesirable environment. This containment function in-space has more rigid requirements in that it must be hermetically sealed. Any such structure fabricated in-space must also be inspected in-space if the required seals and subsequent safety are to be maintained. The above requirement has been demonstrated by the high resupply cost of environmental fluids to large, long duration space stations. Although there is no gravitational loading in orbit, structural requirements are imposed by internal pressurization, thermal cycling, possible artificial gravity, dynamic aspects of a manned space station, launch loads and on-earth assembly. Protection must be provided in the space environment from meteoroid, solar radiation and charged particle radiation. The walls are also to be used for radiant heat dissipation to the near infinite heat sink of deep space and for absorption of heat energy due to solar radiation.

The simplest structure consists of a one layer sheet but imposes extreme weight penalties when considering meteoroid protection and structural loads (buckling) during launch. The addition of stringers to thin materials has been proposed, but this would solve the buckling problem only.

Hypervelocity particle impact studies have shown that a double wall construction is approximately 3 times more effective than an equivalent single wall structure for meteoroid protection. The proposed "Whipple bumper" is a two layer structure with an outer layer 1/12 the original single wall thickness and the inner layer 1/4 the original thickness. The outer layer produces the effect of fragmenting the impacting meteoroid and spreading out the debris over a larger area as it penetrates and strikes the inner wall. The meteoroid protection afforded by the "Whipple bumper has completely eliminated the single sheet wall structure concept. The structural requirements are met by using honeycomb, corrugation, stringer or truss configuration. Honeycomb is not considered an effective "Whipple bumper" due to the channeling effect of the cell structure. An additional third layer must be utilized for the outer meteoroid wall. The space between walls are to be filled with insulation material which will aid in thermal control. Radiation protection requirements are considered minor when compared to structural and meteoroid protection requirements of

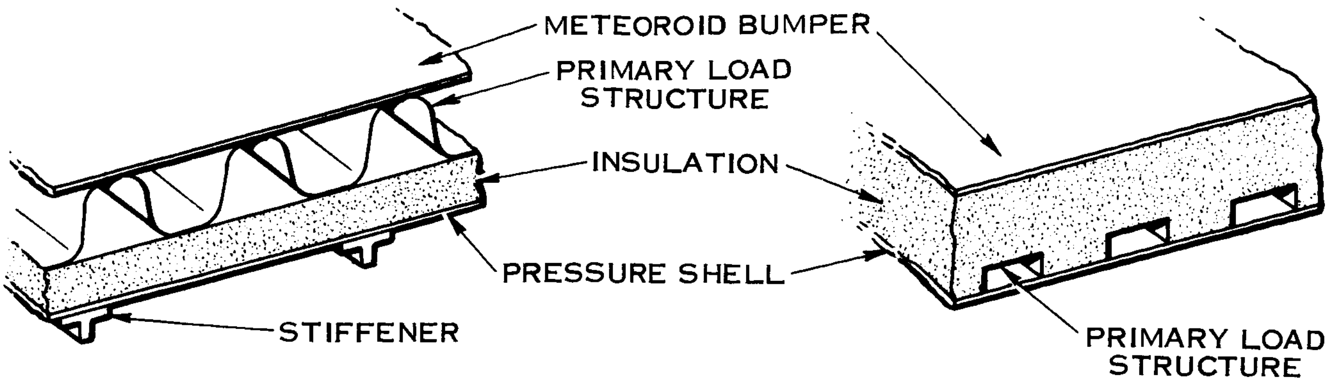


FIGURE 3 METEOROID PROTECTED WALL STRUCTURES

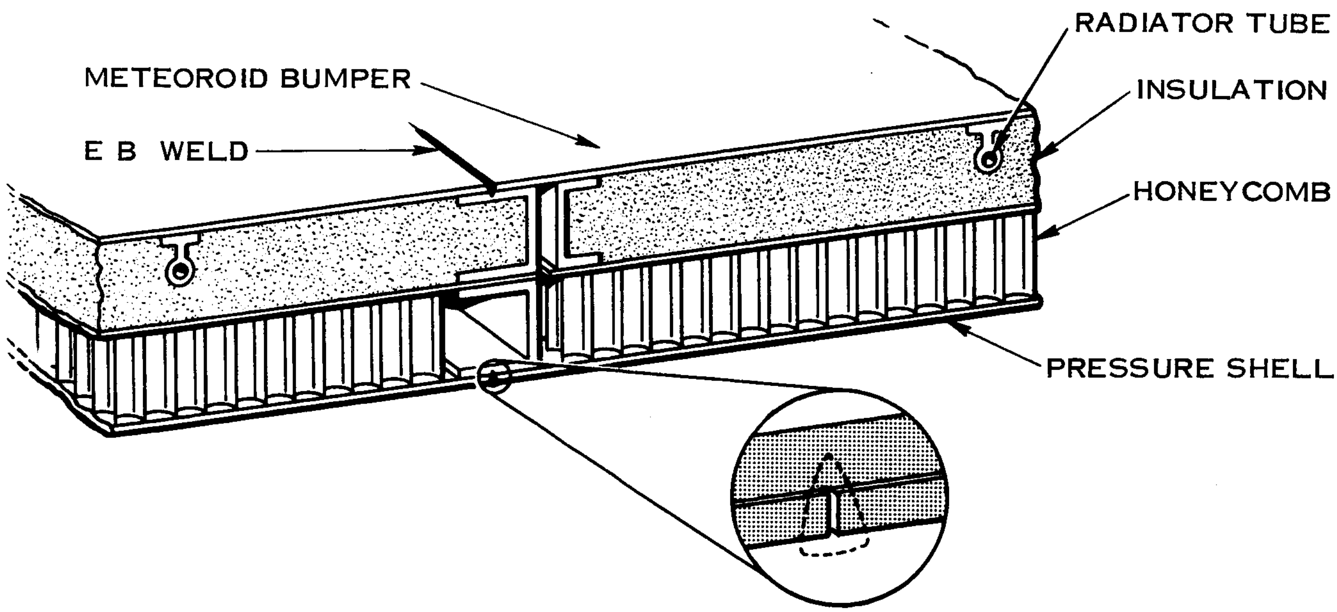
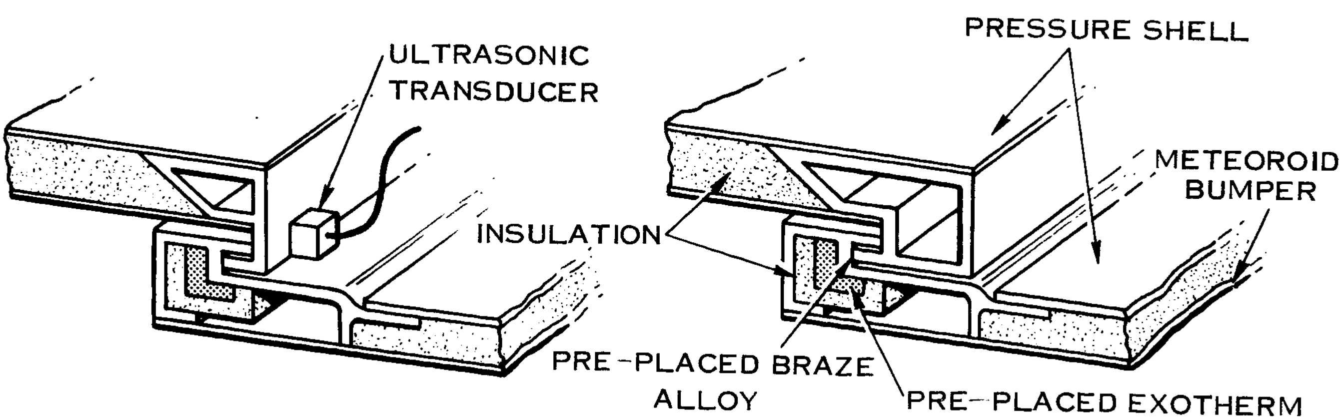


FIGURE 4 E B WELDED SELF ALIGNED WALL JOINT



A — MODIFIED JOINT FOR ULTRASONIC INSPECTION

B — HERMETIC SEAL FOR TELESCOPING SECTION JOINT

FIGURE 5

a space station in low (500 nm.) earth orbit and inclined at less than 30^{o}. Low radiation levels at these altitudes are the result of incident charged particle capture by the magnetic field of the earth at higher altitudes (The Van Allen Belts). However, when considering interplanetary travel, protection from the intense radiation of solar flares becomes a prime design consideration for a significant portion of the space station wall structure. (Protection of the entire space station leads to excessive weight penalties). Thermal stresses due to heat flux variations in orbit will have a significant effect on wall structure design, and hence upon in-space fabrication techniques and subsequent inspection. Allowances must be made for thermal expansion and contraction of the outer layer. A "semi-floating" outer wall is common in most concepts of wall structures. The three structures shown in Figures 3 and 4 have the same common characteristics: (1) internal pressure shell (2) reinforcements to prevent buckling, (3) meteoroid bumper (4) non-rigid support between inner and outer layers and (5) thermal insulation. Techniques for damage repair of these structures is currently under study, however in-space nondestructive test techniques must also be developed to assess the reliability of the repair to assure astronaut safety.

Tubing and duct work systems proposed for space stations are essentially the same as for earth use. However, closed environmental system imposes a much more stringent requirement because of sealing aspects. Tubing and duct work occur primarily in environmental control systems, propulsion systems and power generating systems. Tube joints are expected to occur more frequently as space stations become more complex. These joints may be brazed or welded depending on the specific configuration. If this work is accomplished in-space as will be necessary in many instances, the joint must also be inspected in-space if any quality assurance level is to be expected. Space stations such as the extended axial dumbell, which have atomic power stations and environmental fluid storage tanks located romote to the main living quarters, will require extensive tube and duck work joints and attendant high reliability.

The structure of external appendices is expected to be similar to counterparts on earth. In non-rotating space stations, load requirements will essentially be non-existent except for on-earth assembly and transportation. If used on a rotating station, the artificial gravity will not exceed one "g" due to human considerations, and design requirements are not expected to exceed those on earth. However the possibility of damage on lift off (heavy vibrational loads) is a definite factor, therefore a means of detecting potential damage such as a nondestructive test instrument must be provided.

Applicable references: 8, 44, 14, 13, 2, 12, 32.

Materials for "In-Space" Use

Although a large variety of materials are utilized in present day spacecraft, aluminum and titanium are the major structural materials. The Mercury and Gemini pressure vessels were of commercially pure titanium (AMS 4901) skins stiffened with titanium stringers or corrugations of welded construction. The pressure vessel of the Apollo command module is aluminum honeycomb of 2014 and 5052 alloys. Most secondary construction is also of titanium or aluminum alloys. Magnesium (HK 31A) is used extensively in the Gemini adapter module due to weight considerations but is not considered a principal material for man occupied sections due to a possible explosion hazzard from meteoroid impact. As this alloy is a thorium bearing alloy a potential astronaut communication problem exists in that the thorium could possibly produce radiation and hence interfere with communication. Stainless steel (PH15-7 Mo) is used as a structural support material for the ablative heat shield on the Apollo command module. The service module of Apollo is aluminum honeycomb.

Investigation of secondary structures, power systems and propulsion systems reveals a myrid of high temperature, high strength or corrosion resistant materials depending on the specific applications.

Due to its strength to weight ratio, and state-of-the-art fabrication, aluminum is presently the principal material being considered. Other materials that will also occur frequently are titanium, stainless steel and magnesium. It is expected that as the state-of-the-art of materials development increases, other high strength-to-weight ratio materials such as beryllium and fiber reinforced materials will see extensive use in space oriented systems.

A review of material thicknesses to be encountered in in-space fabrication indicated that approximately 90% of the cases would have wall thicknesses less than .125". The limited number of heavier sections are used in structural reinforcements and radiation protection.

Applicable references: 2, 14, 13, 24, 6, 23.

Joint Designs Proposed

Joint design requirements can be divided into three areas - structural, hermetic seal and attachment. The specific joint design or configuration for hermetic seals or structural requirements that are to be used for in-space fabrication and assembly will depend on space station design, wall structure, material, fabrication technique and inspection technique. In attachment operations where the joint strength is not of prime consideration, the design guide will be simplicity.

For first generation space station, the primary use of in-space fabrication facilities will be hermetic sealing. As stated earlier, because of both safety and atmosphere replenishment costs, these in-space fabrications must be inspected. Since this will occur essentially in rotating stations, the joints will also be structurally loaded. Figure 5B represents a conceptual joint that is contemplated for use in the hexagon configuration studied by NAA. The triple layer honeycomb wall structure has been modified to provide self-aligning facilities for the meteoroid bumper and the pressure shell. A deficiency of this design is the middle layer which is not joined. The stiffness of honeycomb is such that both skins can be used to carry the load of internal cabin pressure. It will be necessary to modify this concept to include either a method of joining the middle layer or a transfer of the middle layer load to the pressure shell. Similar type modifications will be needed for the two other wall structures shown in Figure 3. Electron beam welding appears to be the ideal joining method for this type of joint.

The sealing method for the telescoping joints in the expandable structure developed by Martin Marietta is an internal bladder and an O ring. A metallurgical bond is, however, preferred for a better hermetic seal. Figure 5B illustrates the Martin proposed joint design plus a preplaced placement of braze alloy and exothermic heat source. The joint is further modified in Figure 5A to provide capabilities for ultrasonic inspection. The pre-engineered aspect of the exothermic brazing process readily lends itself to a joint of this nature.

Attachment type joints could be mechanical fastened, E.B. welded or exothermically brazed. The use of organic adhesives, for external applications on long duration space stations,are limited due to deterioration in the space environment. Possible damage to the solar cell panels prohibits extensive astronaut activity in their immediate vicinity, however should such damage occur, a readily available in-space non-destructive test instrument would be invaluable.

Applicable references: 8, 44, 14, 13, 2, 12, 32, 42, 44, 41.

<u>In-Space Fabrication Techniques</u>

An evaluation of joining systems for in-space fabrication was performed by Hughes Aircraft Co. for NASA-MSC. The following systems were evaluated:

1. Electron beam welding
2. Resistance welding
3. Thermochemical brazing
4. Adhesive bonding
5. Solid state joining
6. Gas fusion welding
7. Arc welding
8. Focused sunlight welding
9. Laser welding

The systems are listed in decreasing order of probable in-space success. The criteria for evaluation was materials, design philosophies, logistics of currently contemplated missions,and human factors. The two systems selected by Hughes with the highest potential for adaptation to in-space fabrication are electron beam and resistance welding. The program is continuing with feasibility demonstrations and reliability tests in simulated space environment.

Although the study recognized the potential of resistance welding for space use, there is no current hardware development. While the system has several advantages, there are several major drawbacks. Short electrical pulse times that are characteristic of resistance welding result in a minimum of total power expenditure consistant, with good efficiency. The fusion zone during lap welding is completely surrounded by solid metal which reduces the problems of zero gravity and hard vacuum. Main disadvantages of resistance welding are the clamping forces necessary to provide contact and the inherent restriction to only lap type weld joints. The system is very adaptable for welding of intricate electronic systems but not hermetic seals or the structural members of large space stations.

Narmco Research has been contracted by the Air Force Materials Lab of WPAFB for development of exothermic brazing in vacuum. The object of the program was to design, fabricate and evaluate vacuum bonded joints of stainless steel, titanium, aluminum, and magnesium alloys using an exothermic heat source. A demonstration space station module was assembled and brazed remotely in a vacuum. The program consisted of evaluation of exothermic heating systems, application of exothermic heat to various base metal alloys, joint design and evaluation module design, fabrication, and evaluation. The advantages of the system are: (1) short heating times (5 to 45 seconds from ignition of exotherm to solidification of braze alloy) (2) complete remote control capabilities (3) light weight joints with relatively high strength (4) good hermetic sealing, and (5) totally pre-engineered for a minimum of in-space work. Principle disadvantages of the system are (1) close tolerances are needed for braze joints (2) the potentially dangerous reaction of the exotherm in a pure oxygen environment, and (3) poor brazeability of aluminum and magnesium alloys. Lap shear tensile specimens of 347 stainless steel and A-110AT titanium exothermically brazed had strengths comparable to ordinary vacuum brazed specimens. Exposure of titanium to the exothermic reaction causes near complete loss of ductility (a decrease from 20% to 1% elongations). This problem could be minimized by using a thin metallic interface between the exotherm and titanium.

Types of joints considered in the program were the joining of structural members, sealing of openings, and the rigidification of locating devices. Preliminary design was based on ease of fabrication and joint strength but modifications were made for brazing and exothermic heating. The nine joints designed, brazed and evaluated are listed below:

Type	Description
I	Butt joint, thin walled tube
II	Butt joint, thick walled tube
III	Door hinge
IV	Telescoping joint, thin walled tube
V	Ball joint
VI	Swivel joint
VII	Tee joint, tube to tube
VIII	60^{o} butt joint, tube to tube
IX	Door seal

All joints were successfully brazed with load carrying capabilities greater than the base material.

The space station module fabricated was very similar to the NAA hexagon concept shown in Figure 2. The module consisted of six type 8 joints, three type 7 joints and three type 4 joints. Assembly of the module included placement of the braze alloy, exotherm heat source and ignitor wires. Brazing was done in a vacuum of 3×10^{-6} torr. The total pre-engineered aspect of exothermic brazing is illustrated by the fact that the only operation necessary for brazing,once the module was in the vacuum chamber,was the operation of a switch. All joints were successfully brazed except one where ignition failure occurred. Due to the success of the program the Air Force feels exothermic brazing is now ready for tests in space test beds such as the MOL.

Independent development by Narmco has lead to a commercially available permanent tube connector under the trade name of "Pyrobraze". The totally prepackaged unit is shown in Figure 6. Modifications necessary for space use have been made by Narmco and the unit has been successfully demonstrated in a simulated space environment.

In work recently completed for the Air Force but as yet not yet published, Narmco has adapted the exothermic heat source for adhesive bonding in-space. The system uses a single-component thermoplastic as the adhesive that is heated to bonding temperatures by the exothermic reaction. The principle advantage of this system over the two-component systems developed by NCR is its capabilities of bonding to substrates that are at low temperatures. Figure 7 is the Narmco system adapted to a mounting bolt.

The National Cash Register Co. recently completed a program for the Air Force on the development of a capsular adhesive system that has potential use in a device for attaching an astronaut to an extravehicular activity (EVA) work site. Design goals were an adhesive for use in space that would bond to a variety of substrates with a strength of 100 psi and within 10 seconds of applying an activation force of 2-5 pounds. Two systems were developed with very promising results but more effort is required to evaluate in-space storage life. The encapsulated system is attached to the center of a pad as shown in Figure 8. Six pads can be stored in a dispenser unit which remains mechanically attached to the pad and becomes an integral part of the astronaut-to-work site attachment complex. The system has also been proposed for structural bonding hermetic sealing, meteoroid damage repair and astronaut space suit repair kits.

Although not directly applicable, a fabrication technique that has already been in orbit (Gemini 11) is the space power tool designed and developed jointly by Martin Marietta and Black and Decker. Unfortunately the experiment was cancelled. The power tool is an impact type minimum reaction wrench capable of torques of 400 inch pounds, operated by self-contained Ni-Cd batteries.

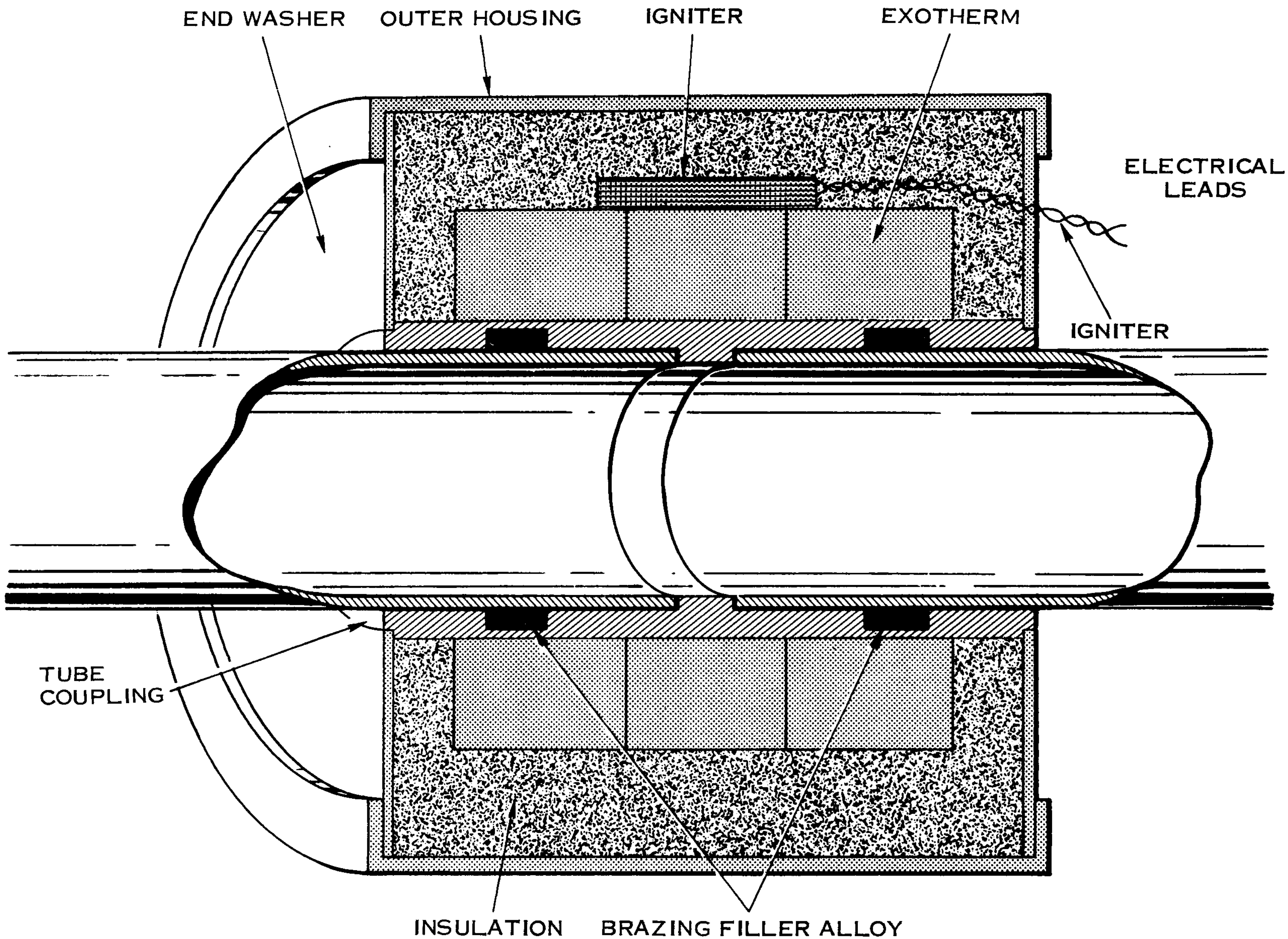


FIGURE 6 CUTAWAY VIEW OF PYROBRAZE TUBE COUPLING UNIT

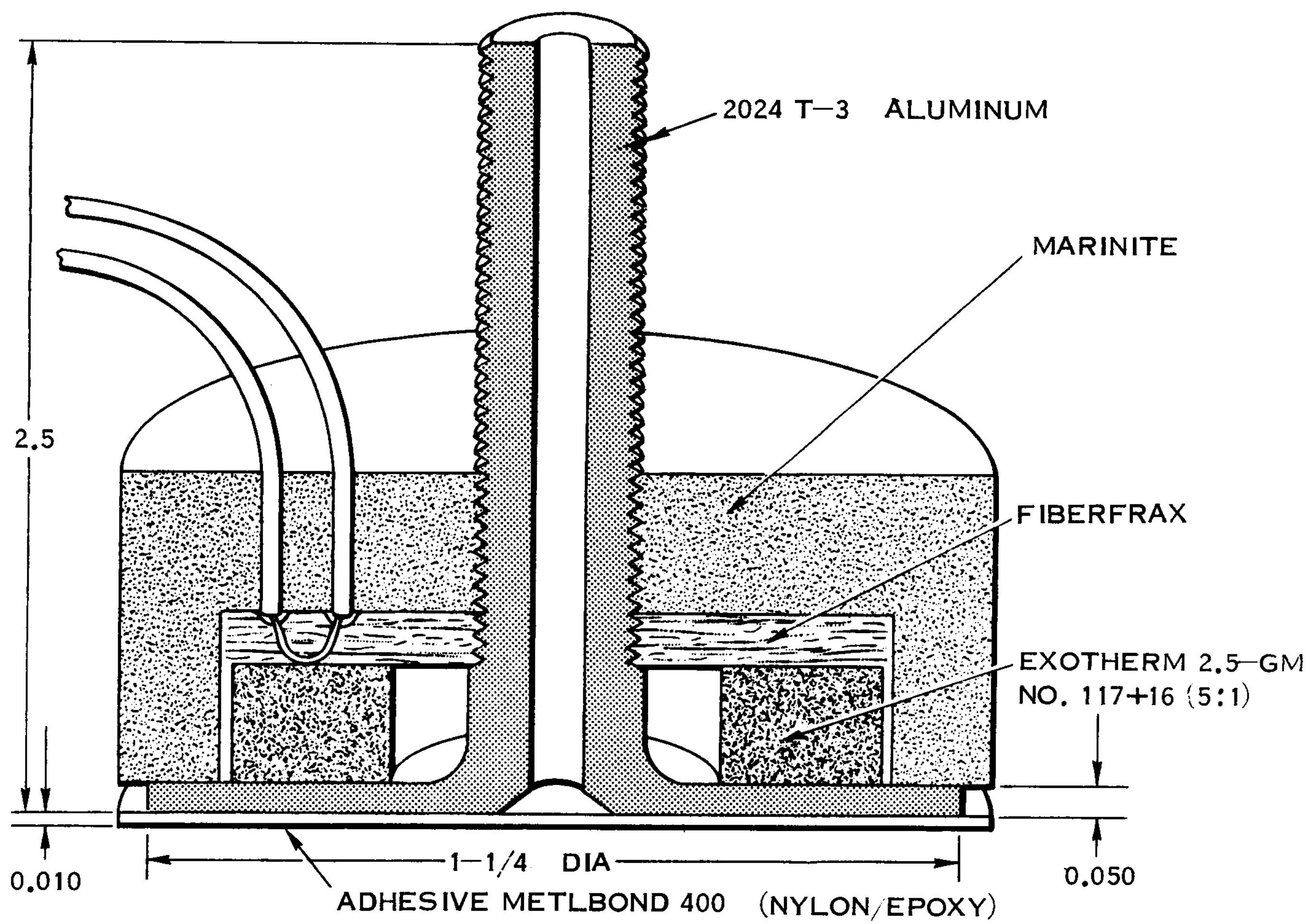


FIGURE 7 EXOTHERM/ADHESIVE PAD PACKAGE FOR IN–SPACE USE

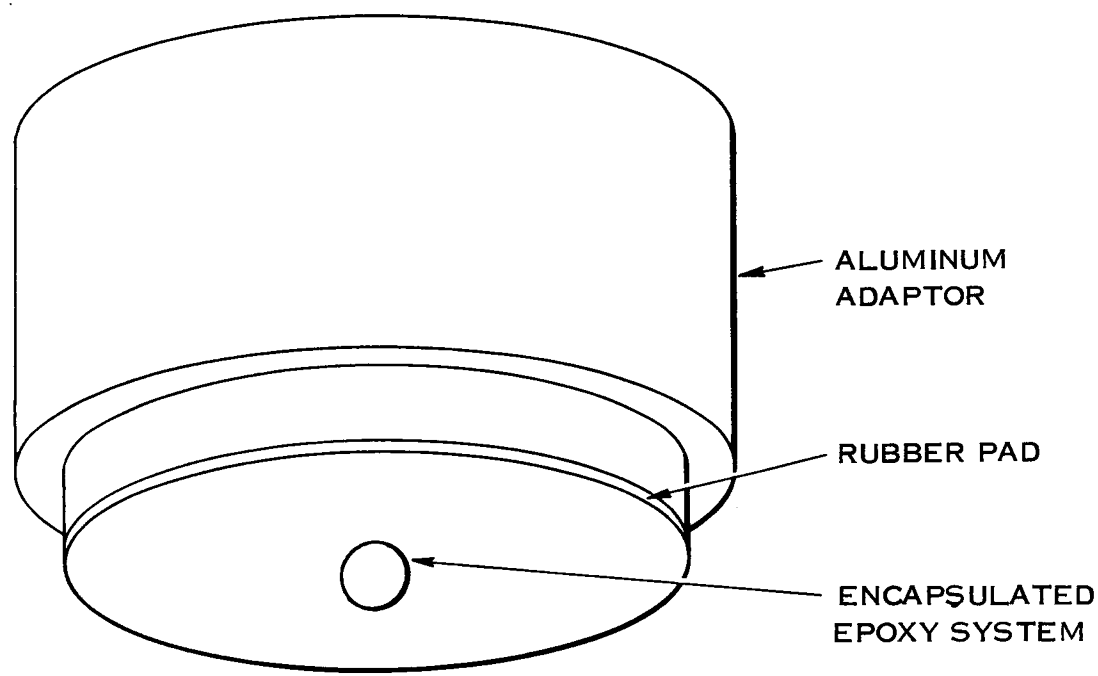


FIGURE 8. PROPOSED ADHESIVE SYSTEM FOR IN-SPACE USE

The most extensively studied and developed system for in-space fabrication is electron beam welding. A 2 1/4 year study was conducted by Hamilton Standard Division of United Aircraft Corp. for the USAF to determine design guidelines for electron beam welding equipment and techniques for variable environment (in-space) use. This study was completed in February, 1965. Hamilton Standard was then contracted by NASA Manned Spacecraft Center in June, 1965 to construct a hand held E.B. gun. This specific gun was successfully demonstrated at a simulated altitude of 73 miles in Hamilton Standard's man rated space chamber in September, 1966. The gun has a rated power of 1.5 KW and an acceleration voltage of 15 KV. This is capable of penetrating approximately 1/4 inch aluminum at a welding speed of 15 inches per minute.

Westinghouse has been contracted by NASA Manned Space Flight Center to design and build a self-contained electron beam welding machine. The equipment is designed to operate from its own battery pack. This equipment is scheduled for delivery to NASA in the latter part of 1967.

The high energy of the electron beam and the small fusion zone provide excellent efficiency. E.B. welded butt joints have strengths equivalent to base metal due to the small heat effected zone. Although the metallurgical bond resulting produces true hermetic seals, subsequent nondestructive testing for safety purposes is still required.

Applicable references: 1, 2, 3, 4, 10, 11, 21, 22.

Anticipated In-Space Defects

Nondestructive inspection instruments to be used for the detection of defects depend on (1) geometry of hardware (2) material, (3) geometry of defect and (4) the position of defect in that geometry. Hardware geometry has been illustrated in the sections on structures and joint design. With only minor modifications, these structures and joints can be inspected with near conventional equipment. Materials in space stations have had extensive use on earth and should impose no additional restrictions on NDT equipment in-space. The geometry of defects is not expected to be substantially different from on-earth defects. Basic equipment capabilities for in-space NDT are met by commercially available equipment, although this equipment must be redesigned to cope with the human engineering factors in the environment of space, and the vibrational problems encountered during launch into space.

Major characteristics of the space environment expected to effect joining operations are the hard vacuum, zero gravity and temperature extremes. These characteristics should have limited effects on electron beam welding. Currently commercial E.B. equipment operates in a vacuum of 10^{-4} torr or better. A vacuum of 10^{-8} to 10^{-10} torr is not expected to adversely effect the weldment. The main effect expected is possible changes in surface energies when absorbed gases are removed. This effect will be small because the high temperature of E.B. welding has a normal tendency to remove the absorbed gases during the welding process. The hard vacuum will increase sublimation and vaporization. This could cause porosity in extremely "hard" vacuums, and possibly a spewing of the molten metal if gas evolution is rapid. Zero gravity effects of the space environment are expected to have a negligible,if any,effect on EB in-space welding. This has been demonstrated at Hamilton Standard by inverted E.B. welding (i.e. directing the beam upward to the workpiece rather than the conventional downward direction). Workpiece temperature variations however will effect optimum weld parameters. If the workpiece temperature has significantly changed between E.B. parameter determination and actual welding, an over or under power situation will occur affecting the weld quality. This situation is possible due to effects of heat flux variations imposed on an orbiting vehicle.

A common defect found in welding processes is porosity. This is expecially true in the case of E.B. welding of joints where total penetration is not required, (i.e. lap joints). Vacuum effect of the space environment may increase this porosity potential. Weld bead cracking, longitudinal and transverse, can occur if weld parameters are not optimized. Beam diameter of the E.B. welding process is sufficiently small to make base metal "fit-up", and tracing of the weld joint a potential problem. Improper fit-up or a joint "miss" (which is highly possible) would obviously result in a totally unbonded area. Defects in in-space E.B. weldments are in general expected to be the same as on-earth weldments (i.e. cold shuts, porosity, bead cracking, lack of penetration, etc.). Initially, until space effects on E.B. welding and human factors (EVA) are better understood, it is expected that defects in-space will be more numerous. It is essential that a means such as nondestructive testing in-space be provided to locate and assess the effect of these defects.

Structural braze joints are expected to be mainly lap joints with close tolerances. The strength of a braze joint is derived from large shear areas. Characteristic on-earth brazing problems are the lack of flow and the lack of wetting. Flow is controlled by capillary action and surface wetting characteristics in the joint and is not expected to be effected by zero gravity conditions of the space environment. Hard in-space vacuum is expected to have a greater effect on brazing than on welding due to the strong dependence of brazing on the wetting phenomenum and surface energy effects. This is the result of removal of surface oxides and contaminants (of which the wetting action and braze flow are a function of) by the hard in-space vacuum. The ultimate result of brazing in the hard vacuum of space then may be excessive flow and a deficiency of braze alloy in the joint. Depending on braze alloy used sublimation and vaporization may also occur causing porosity. Close tolerances required of a braze joint will cause most of the defects to appear to be two dimensional (as cracks). Braze joints, as in the case of most in-space fabrication, will require nondestructive testing to achieve the necessary reliability confidence level. Ultrasonic techniques can best reveal subsurface crack-like defects. Eddy current techniques are also applicable with thin material cross section.

Structural adhesive bonds and seals are expected to have defects very similar to braze joints. Eddy current inspection techniques are not applicable to this type of inspection because of the low conductivity of the organic adhesives. Ultrasonic inspection then is the only reliable NDT method of establishing organic bond integrity.

2.1.2 Repair

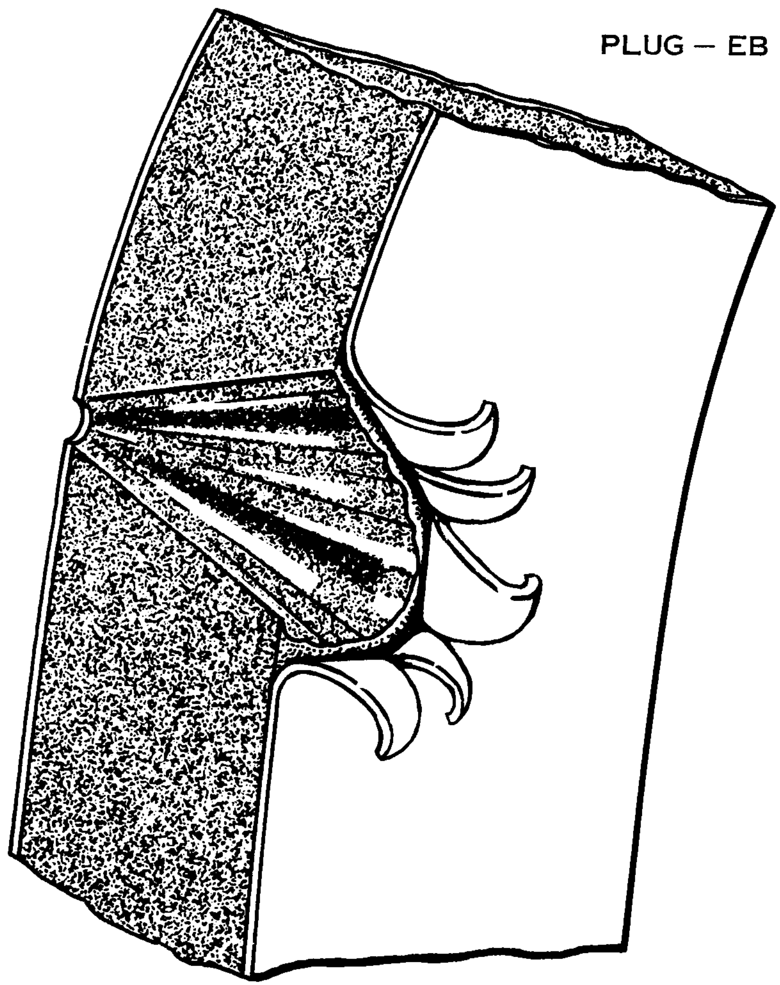
It is expected that accidents, such as meteoroid puncture or docking mishaps, will occur even though they are minimized by extensive on earth engineering and reliability measures. When an accident does occur however, on-board fabrication techniques can be used to make repairs. Nondestructive testing equipment can be used for accurate accessment of the damage and establish repair reliability.

Meteoroid Damage

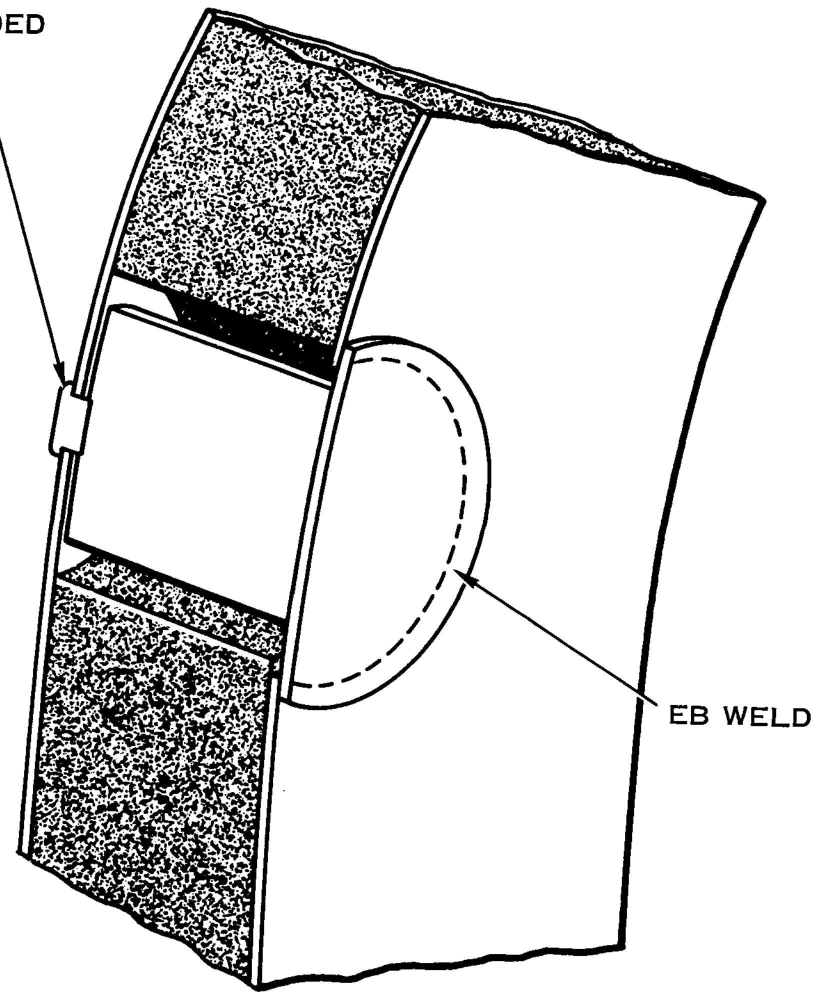
The effects of meteoroids on spacecraft are to be divided into two categories, erosion and puncture. Meteoroid environment in an earth orbit is an inverse log-log relationship between meteoroid mass and flux. The large number of micrometeoroids a spacecraft encounters causes a gradual erosion and degradation of the surface coating and material. The impact of "large" meteoroids (10 to 100 mils in diameter) can create emergency situations. Spacecraft wall structure design is based on a "trade off" between meteoroid protection and weight. Knowing the penetration characteristics of the structure, the meteoroid environment and the space mission, a probability of near zero punctures can be given. Even with the high probabilities of no puncture that manned spacecrafts are now designed to, "sufficient evidence exists at this time to consider meteoroid penetration, when and if it occurs, as a major emergency, rather than the minor inconvenience of locating the leaks." (ref. 8.) The ability of a spacecraft to complete its mission will be seriously jeopardized unless repairs are made and nondestructively inspected.

Consider a double wall, semi-monocoque structure that is filled with a low density foam insulation struck by a high velocity meteoroid. Impact with the outside wall causes a small (1 1/2 to 2 times meteoroid diameter) neat hole. The meteoroid and hole debris are broken into many small particles. As these particles hit the insulation they are generally stopped if it was initially a small meteoroid. However, if the meteoroid was large the results are very disasterous. Impact of the many small particles with the insulation material produces a pressure pulse that is almost an explosion. The pulse is transmitted to both the inner and outer walls, and produces an extremely destructive effect on the inner wall. It literally "rips" a hole in the inner wall that is several hundred times larger than the hole in the outer sheet. This inner wall hole is characterized by jagged petaling metal and large radial cracks. (See Figure 9.) The extent of the pressure pulse damage to the outside wall has not been thoroughly studied but it could be of the same magnitude as the inner wall. The next result of this type of construction is either no hole in the inner wall or an extremely large hole.

The immediate results of meteoroid penetration are (1) a pressure pulse in the interior of the cabin, (2) potential deoxidation of the cabin atmosphere, (3) combustion of cabin material and (4) meteoroid and cabin wall debris injected into the cabin. One of the most serious results are the radial cracks that occur in the inner wall (pressure shell). The shell will be stressed due to internal pressure and the cracks could initiate a catastrophic failure of the wall. Structural supports between the walls, such as stringers, honeycomb



METEOROID PUNCTURE IN DOUBLE WALL STRUCTURE



REPAIR OF METEOROID PUNCTURE

FIGURE 9

or corregution, may tend to minimize this effect somewhat. If complete penetration does not occur, the pressure pulse could cause a bulging of the inner wall, possible cracking and rupture of honeycomb or stringer bands.

Once penetration occurs and before repairs can be made, the puncture must be located and the extent of damage determined. The location of a 2 to 4 inch diameter hole in the inner wall will not be too difficult provided it is not hidden by equipment or secondary structures. If it is necessary to locate the puncture from the exterior surface visual inspection is probably the most effective method for large punctures. Small punctures that cause no ,or only small,effect on the pressure shell will be more difficult to detect and will create a problem of preventive maintenance rather than an emergency situation. Detection of these small punctures will have to be accomplished by nondestructive testing. Damage analysis of a large punctures will deal mainly with the extent of radial cracking. Any repair operation must eliminate these cracks or hermetically seal them and prevent further propogation. Eddy current NDT equipment can be used to determine the extent of these cracks. Bonds between the inner wall and reinforcement structures may be ruptured by the pressure pulse. This damage can readily be analyzed with ultrasonic NDT equipment in space.

Repair of meteoroid puncture damage involves 3 distinct problems. (1) hermetic seal of pressure shell, (2) structural reliability of repaired area and (3) prevention of crack propagation. Hermetic sealing will be essential since the station will be operating on the basis of no resupply. The large dynamic loads of launch will not have to be considered for repair but internal pressurization and possible artificial gravity produce substantial static loads which any repair operation must consider. Cracks can be sealed with organic material but a metallurgical bond is required to prevent propagation if the area of the crack is stressed. Again in-space NDT must be utilized to ascertain the achievement of a complete metallurgical bond in the repair operation. The repaired area compared to the total area of the spacecraft is considered small and the effect on thermal balance will generally be negligible. The low density insulation prevents convective currents of air trapped between the walls due to on-earth construction. The repairs in space eliminate the convective air problem and re-foaming should not be necessary.

Repair of meteoroid puncture has been suggested by two sources as involving a kit type of procedure. The damaged material in the area of puncture will be completely removed. A pre-cut and pre-contoured plate will be positioned over the opening and electron beam lap welded or exothermically brazed to the pressure shell. The advantages of E.B. sited earlier also apply to repair work. Also of consideration is the use of the E.B. gun as a cutting tool to remove damaged material. Exothermic brazing has the advantage of simplicity of operation. Since repair plates will be totally pre-engineered, the braze alloy and exothermic heat source can also be pre-placed on the plates resulting in a minimum in-space effort. Adhesive sealing can also be used but will not provide the permanency or reliability of a true fusion bond.

For mission continuation with reliability of the original structure, the repair operation will be critical and inspection will be required to establish the quality and reliability of the repaired joint.

Potential Cracking Damage

Fatigue cracking is expected to become an increasingly important design consideration as mission times become longer and spacecraft increase in size. The rotation of a space station to induce gravity will also have a definite and significant effect on fatigue considerations. If fatigue cracking does occur simple adhesive patching to prevent leakage will not be sufficient as crack propagation will occur and reopen the crack. Proper repair will involve reinforcement of the area to eliminate a design deficiency as well as sealing of the crack. If a preventive maintenance procedure is established, (as discussed below) the cracks can easily be detected with eddy current or ultrasonic inspection while repair is still feasible. Repair procedures can then seal the crack with an E.B. weld or exothermic braze patch. Structural reinforcements will be critical and NDT inspection of repairs will be mandatory if the mission is to continue. Although the astronaut may not be in any immediate danger without repair-inspection techniques, the cost of their development compared to the cost of "scrubbing" a mission is extremely small.

Structural Damages

Docking of space vehicles is still and will for some time be an intricate operation that requires precision in alignment and vector control. As space activities increase the frequency of docking, maneuvers will increase. The malfunction of a control rocket such as occurred on Gemini 8 and 11 or pilot error could cause sufficient damage to place the astronauts or mission in severe jeopardy. On-board fabrication equipment will be available for repair and NDT equipment can be used for damage analysis and inspection of the repair.

Lunar landings are expected to require the same intricate rocket control as dockings, except the operation will be on a larger scale. Damage of the structural support of a lunar module could prevent take-off unless repairs are made. During landing and launch from the moon's surface, rocket exhaust impingement on the surface are expected to set into motion lunar dust and debris that could cause damage to the module. Damage analysis, repair and repair inspection will again be essential.

Applicable References: 2, 8, 12, 26, 32, 44, 51.

2.1.3 Preventative Maintenance and Inspection

The characteristics of the space environment that contribute primarily to the deterioration and degradation of materials are vacuum, electromagnetic radiation, charged particle radiation and micrometeoroids. Although heat flux variations are important in the space environment they cannot be considered a characteristic of the environment alone. A preventive maintenance program to periodically nondestructively inspect for deterioration is a definite necessity for any extended space missions. Principle areas of concern are thermal control coatings, hermetic seals, and structural joints and materials.

Deterioration of thermal control coatings is the result of a change in the absorption and radiation characteristic (e/a ratio) or a coating thickness reduction due to errosion. Shorter wave length electromagnetic radiation, up to ultraviolet light, and charged particle radiation contain sufficient energy per quantum to initiate chemical reactions. This may change the chemistry of organic base thermal control coatings and hence change the e/a ratio. Other resultant effects are increase in strength and hardness, decrease in ductility plus variations in thermal conductivity and electrical resistivity. Micrometeoroids, charged particle radiation, and sublimation cause erosion or material loss in the space environment. It is conceivable that all of the above changes may be determined through nondestructive eddy current testings.

Continual nondestructive test surveillance will be required to insure material integrity and astronaut safety. In addition, data generated from this surveillance will be invaluable in improving and/or better selection of materials for the space environment.

Structural adhesives are utilized to bond the ablative heat shield to the vehicle substructure. Deterioration of this bond would seriously jeopardize astronaut safety during re-entry. An Apollo type re-entry vehicle docked at a space station for 6 to 8 weeks will be subjected to extremes of the space environment. Heat flux variations and charged particle radiation for these extended periods have strong potential to cause delamination of the heat shield. Nondestructive inspection to determine bond integrity in these cases is mandatory. Techniques currently under development will be capable of revealing these "debonded" areas. Cold welding is a potential problem in extreme vacuums. Failure of moving components; valves, solenoids, etc; may be due to either fracture or cold welding. Radiographic inspection of the interior of the unit would be a valuable aid for in-space "trouble shooting" and repair.

Long interplanetary missions will initially be launched from an earth orbit rather than by direct "lift-off" from earth. For safety and reliability, it may be necessary to inspect critical engine components (valves, tube joints) before final "in-orbit" launch. If failure occurred during launch from earth orbit, repair and inspection will definitely be required before the mission can be allowed to proceed. NDT equipment would be required for an "on-the'spot" failure analysis to enable the mission to continue and to aid future design.

During the separation and ignition of successive stages of a multi-stage rocket, the rocket plume could conceivably impinge on the foreward section of the spent stage. The effect could result in an effective thermal treatment and change the metallurgical properties resulting in a decrease in strength. In some cases these stages are being considered for use as auxiliary space stations, therefore, change in mechanical properties of the spent stage materials could jeopardize the structural integrity of the unit. A nondestructive evaluation of the material will definitely be required. The applicable NDT technique to determine this heat treat change would be eddy current inspection. This thermal change detection is common in on-earth inspection and requires no NDT state-of-the-art increase for in-space applications.

Applicable References: 5, 9, 32, 39, 40.

2.1.4 Other Applications

An NDT package containing radiographic, ultrasonic and eddy current equipment will have applications in areas other than in-space fabrication inspection. Increase of space activities increases the possibility of astronaut injury. On the spot diagnosis of possible bone fracture can readily be accomplished with the radiographic portion of the in-space NDT instrument. This radioisotope is currently being used for medical radiography purposes on earth. Ultrasonic techniques are finding increasing application in medical diagnosis and research. Current applications include determining blood flow, pulsation of arteries, brain symmetry, brain temperature, brain calcification, variations of bone calcium content, tumor location, tumor size, etc. Such information would be invaluable in the diagnosis of astronaut illness and monitoring biological effects of man on long duration space flights. This equipment differs from portable inspection equipment in read-out technique and in some cases a frequency measurement ability only. Basic equipment, such as the Sperry Reflectoscope, could provide meaningful data in this area, in-space, if standards are established and correlated with a slightly modified unit capable of including these techniques as an inherent function. Initial Lunar activities will principly involve geological exploration. Eddy current and radiographic data can be used for qualitative analysis of lunar materials. The low density of the surface and insufficient power supplies, will probably restrict the use of ultrasonic inspection for internal flaws. However, geological samples of dense materials can be examined for internal flaws and defects with ultrasonic equipment, for characterization purposes.
Applicable Reference: 60.

2.2 NDT Method Selection

2.2.1 Multiple NDT Selection

To adequately inspect joining methods considered for in-space fabrication, it becomes apparent that no one inspection method can possibly detect the required defects. For example, cracking in E.B. welds cannot readily be detected by radiography without proper source positioning and/or optimum crack orientation. On the other hand subsurface porosity and inclusions are readily located and identified. Employing ultrasonics, cracks are more easily found but subsurface porosity becomes difficult if not impossible to identify. Surface cracks in welds with crowns removed can easily be detected by eddy current methods provided surface finish is 250 rms or better, but difficult to detect with other NDT techniques.

In examining honeycomb panels, defects can be inherent in the fabrication process. Those which are detectable by ultrasonics are unbonds between the adhesive and face sheet, unbond between the adhesive and the honeycomb, and lack of adhesive material. Specific identification of the particular defect type is still difficult. Radiography is more suited to the detection of crushed core, lack of adhesive material porosity in the adhesive layer, improper cell direction and deformed honeycomb.

Evaluation of 2014 aluminum for changes in temper conditions from T6 to 0, can only be done accurately with eddy current. Defects which are inherent in tube brazing are shrinkage cracking, lack of wetting, joint porosity,low percentage of braze coverage, no capillary flow and excessive flow into the inside of the tubes. Of these defects, those which are most easily detected by radiography are joint porosity, excessive flow, and no capillary flow. Defects readily detected by ultrasonics are shrinkage cracking, lack of capillary flow, and lack of wetting.

It becomes evident that no single NDT technique will suffice for adequate in-space inspection, rather all three basic techniques (i.e. ultrasonics, eddy current, and radiography) must be incorporated to form the required in-space NDT package, if there is to be any assurance that reliable fabrication is completed in-space construction. Due to the serious consequences of a joint failure in space, both to astronaut safety and structural failure, it should be stressed that defects normally considered minor problems on earth, become major problems in space.

Equipment Review - Ultrasonics

Five ultrasonic equipment manufactures were reviewed for potential use of their portable ultrasonic units for in-space inspection. An evaluation of the instruments is summarized in Tables 1 and 2.

Each instrument was evaluated on the following characteristics:

1. Capability of performing defect detection in electron beam welds, adhesive bonded honeycomb, and brazed tubing.

2. Unit size

3. Weight

4. Power Supply

5. Physical Construction

6. Electrical Design

7. Types of components used

8. Availability of equipment

9. Overall capability for demonstration hardware

The manufacturers were evaluated for present or potential capabilities as related to this program in the following areas:

TABLE I . ULTRASONIC INSPECTION INSTRUMENT SURVEY

VENDOR	MODEL NO.	TEST CAPABILITY	WEIGHT	DIMENSIONS LG-W-HT	POWER SUPPLY	OPERATING CHARACTERISTICS FREQUENCY RANGE	CRT SIZE	ELECT. CONST.
Branson	Sonoray 301	Straight beam contact and immersion (with single or dual trans.) angle beam and surface wave.	16 lb including bat.	15"X9.5"X4.5"	Nickel-cadmium rechargeable 5 hrs-life	1.5m Hz to 8.0m Hz. Broad Band	2.5"X2.5"	Trans.
Budd	UT-700	Same as above	16 lb without chgr. 23 lb with charger	15"X9"X6"	Silver-cadmium rechargeable 4 hour-life	.5m Hz to 10m Hz Broad Band	3"X3"	Trans.
Krautkramer	USK-5M	Same as above	11 lb inc. battery	15"X4.5"X7"	Lead-acid rechargeable 10 hrs-life	.4m Hz to 12m Hz Broad Band	3.125"X 2.5"	Trans.
Magnaflux	———	Same as above	——— No specific info	15.5"X4.5"X6.5" (approx.)	Silver-cadmium rechargeable 8 hrs-life	1m Hz to 10m Hz Broad Band	Unknown	Trans.
Sperry-Automation	UCD Reflectoscope	Straight beam contact and immersion (with single or dual trans) angle beam and surface wave.	15 lb. Inc. charger Inc. battery	16"X8"X5.25"	Silver-cadmium rechargeable 8 hrs-life	1m Hz, 2.25m Hz & 5m Hz Tuned pulser/ receiver	2.5"X2.5"	Trans.

TABLE II . RATING POTENTIAL-ULTRASONIC UNITS
(1 = most favorable rating)

VENDOR		Sperry-Automation	Magnaflux	Krautkramer	Budd	Branson
MODEL NO.		U.C.D. Reflectoscope	——	USK-5M	UT-700	Sonoray 301
WEIGHT		2	- *	1	3	3
VOLUME		2	- *	1	4	3
POWER SUPPLY (Batteries Only)		1	2	3	5	4
COMPACTNESS	ELECT. DESIGN	1	- *	1	3	2
INTEGRATION	ELECT. DESIGN	1	- *	3 *	2	2
NO. OF INTERNAL PARTS	ELECT. DESIGN	1	- *	- *	2	-
EASE OF OPERATION		1	- *	- *	2	3
TECHNICAL ASSISTANCE		1	- *	4	2	3
MODIFICATION ABILITY		1	- *	3	2	2
UNIT AVAILABILITY		1	3	1	1	2
OVERALL CAPABILITY FOR DEMONSTRATION HARDWARE		1	3	1	2	2
POTENTIAL FOR FLIGHT HARDWARE FABRICATION		1	4	4	2	3
OPERATING TEMPERATURE RANGE		$0^{o}F-120^{o}F$	$0^{o}F-120^{o}F$		$10^{o}F-120^{o}F$	$-10^{o}F-150^{o}F$
VOL. - FT^3		.3889	*	.2734	.4687	.3958

* INFORMATION NOT SUPPLIED BY VENDOR

1. Technical assistance relating to their specific unit, during fabrication of the demonstration unit.

2. Conformance to NASA quality control requirements for flight hardware potential. Pertaining to parts procurements, typical requirements are: Source control documents, traceability, material identification, and NASA qualified and/or military standard parts.

   In the area of manufacturing, necessary requirements considered; process control documentation, in process inspection, and NASA qualified soldering procedures.

Equipment manufacturers applicable to this program were:

(1) Branson Instrument, (a subsidiary of Smith, Kline, and French Lab.)
Stamford, Conn.

(2) The Budd Instrument Division
The Budd Company
Phoenixville, Penn.

(3) Krautkramer
Stratford, Conn.

(4) Magnaflux Corp.
Chicago, Illinois

(5) Sperry Products Div.
Automation Industries
Danbury, Conn.

Due to the nature of this program and the factors to be considered in selection of an ultrasonic unit it was felt,that in addition to a literature survey,each vendor be contacted with regard to engineering personnel, manufacturing capability, and vendor facilities capabilities. Equipment demonstrations and internal construction were examined to determine quality level, and ease of adaptability to prototype flight hardware.

Results of the evaluation indicated the following:

1. Sperry and Budd exhibited small portable lightweight units most readily adaptable to prototype in-space hardware. These features included minimum number of controls, highest number of electrical components meeting Military Specifications and ease of operation.

2. In the area of potential capability to assist in flight hardware designs, Sperry Products appeared most suited. The other three American suppliers rated approximately equal while Krautkramer in Germany rated last. The Krautkramer rating stems only from the fact that unit design and manufacture are done in Germany, and a definite communication problem exists (prints, part numbers, etc. in German).

   a) Quality control is practiced by all vendors contacted only to the extend of supplying a reliable, commercial instrument which would be competitive with other manufacturers units. None of the vendors have established formal quality assurance systems equivalent to that required for NASA level fabrication.

   b) Procurement of parts is conducted on an economical basis only. Source control and procurement drawings are employed only on a limited basis, on critical parts. Military standard components are not usually used with the exception of resistors. In this area Sperry rated highest with the greatest percentage of components meeting Military Specifications.

   c) Due to the commercial nature of the manufacturers, none of the vendors possess in-house capability to perform NASA quality soldering.

Selection of the Sperry UCD was based on a number of evaluations oriented toward prototype space hardware. It is both compact and lightweight. Internal electrical construction and design appears above average. The power requirement-life consideration combination of the unit appears well chosen. Integration of the eddy current unit circuitry with the Sperry unit is possible with only minor modification. The UCD has the least number of internal parts consistent with the greatest number of components meeting Military Specifications. The minimum number of external controls and ease of operation of the UCD before human engineering for in-space use are desirable features. Sperry also indicated positive technical assistance for modifications and facilities if desired.

Eddy Current Equipment Selection

A total of five equipment manufacturers were reviewed for potential use of their portable eddy current devices. A summary of the instruments reviewed is illustrated in Tables 3 and 4.

Each device was evaluated on the basis of the following characteristics:

1. Ability to perform crack detection and temper variations detection. (range versatility)

2. Size

TABLE III . EDDY CURRENT INSTRUMENT SURVEY

VENDOR	MODEL NO.	TEST CAPABILITY	WEIGHT LBS.	DIMENSIONS L X H X W	POWER SUPPLY	OPERATING CHARACTERISTICS	METER READOUT SIZE	ELECTRICAL CONSTRUCTION
Automation Förster	Sigmatest "T", Type 2.067	Temper	4.8	8 3/4"X3 3/8"X6 1/2"	Batteries (2)"D" Cells	Range: 8 to 105% I.A.C.S	1 1/2"	Transistors
	Defect-ometer Type 2.154	Surface Cracks	19.8	13"X9"X6 1/4"	110 VAC	Variable Frequency	4"	Vacuum Tubes
Budd Company	Radac 150	Temper	5.5 Less case and 4 meters	4 1/2"X5 1/2"X 7 1/2" Less case	Battery Mallory #302478 9 VDC	Frequency = 64K Hz (5) Plug-in meters for changing range	3 1/2"	Transistors
Magnaflux Corp.	FM-120	Temper	8.0	9"X4 1/8"X6 1/2"	Batteries (3) "C" Cells	Range: 13 to 100% I.A.C.S.	1 1/2"	Transistors
	ED-510	Surface cracks & temper possible	22.5	13"X9"X8"	110 VAC	Variable frequency	4"	Vacuum tubes
	ED-520	Temper & Surface cracks	8.0	9"X4 1/8"X6 1/2"	Battery, re-chargeable	Variable frequency 50-200K Hz	2"	Transistors

TABLE III . EDDY CURRENT INSTRUMENT SURVEY (CONT.)

VENDOR	MODEL NO.	TEST CAPABILITY	WEIGHT LBS.	DIMENSIONS L X H X W	POWER SUPPLY	OPERATING CHARACTERISTICS	METER READOUT SIZE	ELECTRICAL CONSTRUCTION
Northwest Technical Industries (Nortec)	NDT-2	Temper & surface cracks	3.0	7"X4"X5"	Two batteries	Range: 100 to 100K Hz Frequency changed by replacing internal module	3 1/2"	Transistors
Uresco, Inc. (Rompas Ultrasonics)	Inductest FC 2001	Temper & surface cracks	6.0 w/batteries 8.0 w/110 VAC	6"X7"X10"	Either Batteries (3) 12 VDC or 110 VAC	Range: 45K Hz to 200 K Hz & 0.5 to 107% I.A.C.S.	4"	Transistors
	Inductest FC 300S	Temper & surface cracks	1.75	2"X3 1/2"X6 1/4"	Batteries (3) 9 VDC	Range: 50 to 150K Hz or 50 to 200K Hz	3 1/2"	Transistors

TABLE IV . RATING POTENTIAL-EDDY CURRENT UNITS
(1 = most favorable rating)

VENDOR		Uresco, Inc. (Rompas Ultrasonics)		Northwest Technical Industries (Nortec)	Magnaflux Corp.			Budd Company	Automation Forster	
MODEL NO.		Inductest FC 300S	Inductest FC 2001	NDT-2	ED-520	ED-510	FM-120	Radac 150	Defectometer Type 2.154	Sigmatest "T", type 2.067
WEIGHT		1	5	2	6	8	6	4	7	3
VOLUME		1	6	2	5	8	5	3	7	4
POWER SUPPLY (Batteries Only)		2	2	1	4	—	3	5	—	Data Not Submitted
COMPACTNESS	ELECTRICAL DESIGN	1	3	2	4	7	4	5	8	6
INTEGRATION	ELECTRICAL DESIGN	1	4	2	3	7	5	5	8	6
NO. OF INTERNAL PARTS	ELECTRICAL DESIGN	1	4	2	3	6	5	5	7	5
EASE OF OPERATION		1	4	5	2	4	3	6	7	3
TECHNICAL ASSISTANCE		1		1	2			3	4	
MODIFICATION ABILITY		1		2	3			4	Unknown	
UNIT AVAILABILITY		Stock	Stock	Stock	Not Before Jan-67	Stock	Stock	Stock	60 Days	Stock
OVERALL CAPABILITY FOR DEMONSTRATION HARDWARE		1	4	2	3	7	5	5	8	6
POTENTIAL FOR FLIGHT HARDWARE FABRICATION		1		1	1			2	3	

3. Weight

4. Power Supply

5. Physical Construction

6. Electrical Design

7. Probe Design

8. Method of readout display (ease of integration to CRT display)

9. Types of components used

10. Availability of equipment

The manufacturers were considered for their present or potential capabilities in the following areas:

1. Technical assistance regarding their specific unit during fabrication of demonstration unit, including ability to perform special modifications if necessary.

2. NASA level quality assurance.

    a. Parts Procurement

        (1) Source Control Documents

        (2) Traceability

        (3) Material Identification

        (4) MIL - Standard Parts

    b. Manufacturing

        (1) Process Control Documentation

        (2) In-Process Inspection

        (3) NASA Qualified Soldering

The equipment manufacturers considered for this program were:

Automation Forster, subsidiary of Automation Industries, Inc. - Ann Arbor, Michigan (American Sales Office) - Reutlingen, West Germany (Design and Manufacturing); Budd Instruments Division - The Budd Company - Phoenixville, Penna; Magnaflux Corporation - Chicago, Illinois; Northwest Technical Industries (NORTEC) - Division of Scientific Advances, Inc. - Richland, Washington; and Uresco, Inc. (Representing Rompas Ultrasonics) Downey, California.

In addition to a preliminary literature survey, evaluation of eddy current devices, manufacturing, and engineering facilities of all potential vendors, except one, were visited by engineering personnel. Equipment performance d demonstrations were observed and internal construction of potential instruments was examined. Manufacturers were asked to demonstrate the equipment on samples of 2014 Aluminum alloys in both the "T6" and "O" conditions and also to demonstrate on samples of stainless steel tubing.

Results of the evaluations revealed the following:

1. Two vendors, Uresco and Nortec, exhibited suitable features adaptable to in-space use. This was directly related to current available off-the-shelf equipment. Both of these suppliers were able to exhibit small compact instruments capable of performing both conductivity (temper variation) measurements and flaw detection.

2. In the area of potential assistance capability in flight hardware considerations, four of the five suppliers were rated about equal. The fifth supplier (Automation Forster) could not be equally evaluated because the design and manufacturing is done in West Germany. The four American-based suppliers all exhibited the following characteristics:

   a. Quality control is practiced on a limited level; comparable to most suppliers of industrial instrumentation. Only two vendors, Uresco and Budd are currently involved in a government contracts requiring military standard quality control of hardware. None of the vendors had established quality assurance systems equivalent to that required for NASA level fabrication. This, however, was to be expected as equipment of this type had never before been considered for aerospace flight use.

   b. Parts procurement is conducted by the four suppliers using only commercial level hardware. As a rule military standard components are not used. Source control and procurement drawings are used only to a limited extent by these vendors.

   c. Because of the commercial nature of the industry, none of the vendors possess in-house capability to perform NASA quality soldering.

Evaluation of available equipment, by comparison of instrument accuracy, was difficult to perform. This is so primarily because the only unifying standard available, in the eddy current field, is the International Annealed Copper Standard (I.A.C.S.). The marked differences in unit construction (type of readout, circuit design, external adjustments) do not allow direct unit comparison. The I.A.C.S. is a relative scale of conductivity measurement and does not directly relate to the ability to perform crack detection. All units surveyed had adequate resolution ability in determination of relative conductivity. The ability to do both flaw detection and temper detection, however, was currently available only in the Nortec and Uresco portable units. Significant in these units was the ability to integrate the complete system in the most compact of packages. Complete system performance (performance being a function of probe design, specimen material and shape and most important the electrical circuit design) however, enabled the selection process to be narrowed to the Nortec model NDT-2 and Uresco (Rompas) FC300S.

The ultimate selection of the Uresco Inductest FC300S over the Nortec NDT-2 was made of the basis of circuit design differences. This was, for the most part, related to the Hamilton Standard decision to recommend a dual ultrasonics and eddy current instrument design. (A further description of the reasons for going dual package over single package is contained elsewhere in this report). Basically, the Uresco unit allowed for ease of system integration with the ultrasonics unit.

Radiography - Equipment Selection

A total of three manufacturers of radiography equipment (both "X" and gamma ray) considered to be consistent with the objectives of this program were considered. In addition to product brochures, each manufacturer was contacted to obtain as thorough an evaluation of the applicable radiographic equipment as possible. The three manufacturers contacted were; Picker Division - Technical Operations Incorporated; Sperry Products Division - Automation Industries; and Budd Instrument Corporation. Specific points of evaluation considered included:

Test Capability
Weight
Dimension
Radiation Source
Operating Characteristics
Ease of Operation
Manufacturer Technical Assistance Available (if needed)
Ease of Modification
Unit Availability
Ultimate Flight Hardware Potential

The evaluation is itemized in tables 5 and specific ratings given each potentially applicable unit in table 6.

TABLE V . RADIOGRAPHIC INSPECTION UNIT SURVEY

VENDOR	MODEL NO.	TEST CAPABILITY	WEIGHT LBS	DIMENSIONS LG - WT - HT	POWER SUPPLY	OPERATING CHARACTERISTIC
Picker X-Ray	VISO 601	Directional	22	6 1/4" X 3 5/8" X 6 1/8"	Isotope Yb 169 100 Curies Max.	Gamma Radiation
Picker X-Ray	532	Panoramic	Container 23 Control 13	10" X 11 1/2" X 4 1/8" 7' Guide Tube	Isotope Ir 192 10 Curies Max.	Gamma Radiation
Sperry Products		Directional	50	Tube Head Dia. 6" Lg. 18" Power Pack 8" x 9" x 12"	Bat.	X-Ray
Budd	100	Panoramic	Container 37 Plus Control	7 1/2" X 4 1/2" X 9 1/2"	Isotope Ir 192 100 Curies Max.	Gamma Radiation

TABLE VI . RATING POTENTIAL - RADIOGRAPHIC UNITS

(1 = most favorable rating)

VENDOR		Budd	Sperry Products	Picker X-Ray	Picker X-Ray
MODEL NO.		100		532	VISO 601
WEIGHT		3	4	3	1
VOLUME		2	4	3	1
POWER SUPPLY		4	1	4	2
COMPACTNESS	ELECT. DESIGN	3	4	3	1
INTEGRATION	ELECT. DESIGN	3	4	2	1
NO. INTERNAL PARTS	ELECT. DESIGN	2	5	2	1
EASE OF OPERATION		3	4	3	1
TECHNICAL ASSISTANCE		2	2	2	2
MODIFICATION ABILITY		2	3	2	1
UNIT AVAILABILITY		1	1	1	3
OVERALL CAPABILITY FOR DEMONSTRATION HARDWARE		4	5	4	1
POTENTIAL FOR FLIGHT HARDWARE FABRICATION		3	5	3	1
VOLUME - Ft^3		.1855	.8182	.2743	.0804

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VOLUME		2	4	3	1
POWER SUPPLY		4	1	4	2
COMPACTNESS	ELECT. DESIGN	3	4	3	1
INTEGRATION	ELECT. DESIGN	3	4	2	1
NO. INTERNAL PARTS	ELECT. DESIGN	2	5	2	1
EASE OF OPERATION		3	4	3	1
TECHNICAL ASSISTANCE		2	2	2	2
MODIFICATION ABILITY		2	3	2	1
UNIT AVAILABILITY		1	1	1	3
OVERALL CAPABILITY FOR DEMONSTRATION HARDWARE		4	5	4	1
POTENTIAL FOR FLIGHT HARDWARE FABRICATION		3	5	3	1
VOLUME - Ft^3		.1855	.8182	.2743	.0804

Consistent with program objectives of developing radiographic techniques and equipment for earliest possible utilization in the space environment is the consideration of a low power, light weight, and commercially available isotope radiographic unit. Evaluation of available isotope radiographic inspection units of this nature has indicated that the VISO unit lends itself readily, with few modifications to in-space use. The unit currently marketed by the Picker Division of Technical Operations Incorporated, has a total weight of 22 pounds and overall dimension of 6 1/4" X 3 5/8" X 6 1/3". The isotope source utilized in the unit is the low energy (approx. 53 Kev) ytterbium 169 and with present design is AEC rated for a 100 currie source. Shielding for this source is the tungsten base alloy, Mallory 2000. The isotope produces gamma radiation with literally no backscatter. This, coupled with the low energy level, yields itself to optimum operator (astronaut) safety in isotope radiographic inspection. Although this source has seen only limited use, (because of high cost and short half life - 31 days) it is ideal for a program of this nature (i.e. thin, low absorptivity materials, optimum operator safety due to low energy peak, literally no backscatter, etc.). In addition, the relatively high photon emission of ytterbium - 169, when incorporating an aluminum collminator, allows a relatively short exposure time. The relatively short half life of this isotope for a program of this nature, is decidedly an asset in that the radiation hazard is minimized. Specific characteristic of this isotope are shown in table VII.

2.2.2 Single Method Selection

The inadvisability of utilizing a single NDT techniques to adequately determine the various defects expected in "in-space" structures, fabricated, or repaired "in-space", has already been documented (Section 2.2.) However because of possible weight, volume etc., restrictions on any specific mission it may be advisable to determine the most versatile NDT technique for in-space use. Results of this study program have indicated ultrasonics to be this most versatile NDT type. However, this study has further indicated that integration of the eddy current and ultrasonic units can be accomplished at minimal cost increase and with a fraction of a pound weight increase. Increased capability resulting from this integration would definitely justify its consideration as the "single method" recommendation resulting from this study. Specific defect types considered detectable by this unit include:

1. In Electron Beam Welds

   Subsurface cracking, porosity, lack of fusion, internal cracking, undercutting, and incomplete penetration.

2. In Adhesive Bonded Honeycomb

   Unbonds between the adhesive and face sheet, unbond between the adhesive and honeycomb, and lack of bond adhesive.

3. For Materials and Coatings

   Surface cracks, temper variation, and coating thickness measurements.

TABLE VII . SPECIFIC CHARACTERISTICS OF YTTERBIUM 169

Energy Range	50-60 KEV
Yield of Source	35m RHM per curie
Practical Range of Source Activity	Up to 75 curies for 2mm source
Useful life (3 half lives)	90 days
Half value level	
Fe	.030"
Al	.375"
Ref: 59	

4. In Brazed Tubing

Shrinkage cracking, lack of capillary flow, and lack of wetting.

Although a certain degree of skill is required in interpreting data, the increased versatility obtained with the ultrasonic - eddy current combination more than justifies the additional training. It should be stressed that cracking, which should be considered as the most critical detect, can most readily be detected utilizing ultrasonic techniques.

In reviewing the three methods of inspection, advantages and disadvantages are listed in Table 7.

2.2.3 Ultrasonic Inspection In-Space

In examining the electron beam welding process, major defects which can be occurred are; cracking, porosity, lack of fusion, incomplete penetration, and undercutting. These defects can normally be detected by ultrasonics utilizing proper ultrasonic techniques and parameters. Due to the relatively thin sections and joint configurations in space applications, shear wave techniques appear to be most applicable. Angle beams in the range of 45^{o} to 80^{o}, for material cross-sections smaller than 1/2" are presently good "on-earth" commercial practice. Because of possible unfavorable defect orientation in welds, (parallel to the shear wave) inspection must be performed from several directions to achieve the required defect detection confidence level. Size of the defects which now become critical because of the very nature of in-space applications (i.e. astronauts safety), are of such a small magnitude that quartz, which is compatible with the space environment must be replaced by a more sensitive crystal, such as lithium-sulphate. To withstand the temperature extremes of space, crystals such as lithium-sulphate will have to be suitably encased and protected. Also, to detect the small defects expected, higher frequency transducers must be used.

Current program plans require transducers ranging from 5m Hz to 15m Hz. Considering ultrasonic inspection of T-joints, the joint cannot be approached with an angle probe from both sides, and inspection must be done from one side only. To complement angle beam inspection, in thicker sections, it is possible to inspect the T-joint from the top of the "T" using a longitudinal beam. Because of the thin sections the ultrasonic instrument will be required to have very good resolution.

In evaluating bonded honeycomb structures, defects, such as voids, delaminations, porosity, poor adhesion, unbond and crushed core exist if all fabricating parameters are not followed precisely. To detect these defects four ultrasonic techniques are considered; pulse-echo, through transmission, surface waves and single probe-double-crystal transducers. Pulse-echo techniques considered are based on the honeycombs characteristics

TABLE VIII . ADVANTAGES & DISADVANTAGES OF 3 MAJOR NDT TECHNIQUES

METHOD OF INSPECTION	ADVANTAGES	DISADVANTAGES
Eddy Current	Has capability of detecting surface cracks. Is capable of measuring thickness of coatings. Is capable of detecting changes in temper in aluminum alloys. Is safe to operate.	Surface finish must be at least 250 RMS. Is limited to surface inspection.
Radiography	Numerous materials and configurations can be inspected, including assemblies. A variety of defects can be detected . Permanent records of inspection are produced. Results can be interpreted with a high degree of certainty.	Possibility of radiation exposure to the astronaut. Fixed positioning with respect to the test object essential. Sensitivity will be low due to film developing processes in space. Two sides of the object to be inspected must be accessible.
Ultrasonics	Has capability of detecting a wide variety of types of defects. Has versitility to inspect a variety of configurations and materials. Is lightweight and compact with self contained power supply. Can inspect from either side of an object. Safe to operate.	Requires skill to operate unit, manipulate the transducer, and interpret. Requires a couplant compatable with space environments.

of absorbing sound energy. Theoretically a good bond will dissipate sound through the honeycomb while unbond will cause a ringing of energy in the thin face sheet. Transducers considered for this application will be larger in diameter than the one inch debond test standard.

Utilizing through-transmission techniques, at frequencies ranging from 1 to 5m Hz, the majority of the previously disclosed defects can be detected. It should be noted that some of the difficulties inherent to through-transmission techniques are: (1) transducers must be placed on both sides of the panel (2) transducer alignment opposite one another and (3) exact location of defect with respect to depth cannot be determined.

Bond inspection using surface waves is the result of energy transmitted from one transducer to another, in the form of surface waves, and each adhesive bond absorbs a small portion of the total energy. The technique is such that if the transducers are adequately coupled to the test piece, debonds will be indicated by an increase in received energy since good bonds tend to attenuate sound. This type of inspection requires good coupling and wave lengths that are larger than the thickness of the face sheet.

A double crystal, pitch-catch type probe may also be utilized to detect debond areas. With this technique the probe is applied to the face sheet of the test specimen over a good bond and a minimum of sound energy is reflected into the receiving crystal. When a debond occurs, a greater amount of the incident energy is reflected and can be recorded.

It should be noted that no one method of the four ultrasonic inspection techniques described can fulfill all the requirements necessary to accurately locate, and identify bonding defects. The applied technique will be dependent upon the particular panel configuration and joint geometry.

Inspection of thin wall, small diameter welded tubing necessitates higher frequency transducers and a precise method of maintaining intimate contact between transducer and the work piece. The most difficult problem is maintaining proper entrance angle because slight deviations from the entrance angle can reduce system sensitivity. Modes of inspection considered are shear wave and surface waves at frequency ranges from 5m Hz to 15m Hz.

An expected in-space fabrication is that of brazing tube joints. The defects encountered in brazed joints are very similar to those adhesive bonded (i.e. determining percentage or area of the void). Here the transducer alignment problem is essentially the same as that described for weld joints except that longitudinal wave techniques are utilized. The application again requires high resolution capability in the instrument. With reference to vacuum, temperature, and oxygen atmosphere transducer requirements, the most critical factor appears to be the temperature extremes. Lithium sulphate considered to be a prime transducer material because of its desirable resolving power and sensitivity, tends to decompose at 220°F. The transducer therefore must either be insulated, covered with an emissivity coating, or designed in such a manner as to allow the couplant liquid to control crystal temperature. Present transducers utilize expoxies to bond the internal components to form the finished unit. At approximately 200°F, many of these epoxies breakdown causing transducer malfunction. Transducer temperature control may rectify this situation.

The ultrasonic circuits and readout systems can be suitably repackaged in a sealed container with specially designed sealed controls to operate in the O_2 and hard vacuum atmospheres. Suitable thermal control coating and insulation used in the package can protect circuitry from the temperature extremes. Due to the broad range of defect detection capabilities and suitable adaptation to the space environment, ultrasonics is expected to be one of the prime methods of in space NDT.

2.2.4 Eddy Current Inspection In-Space

Eddy current methods appear to be suitable for three major types of anticipated defects (surface cracks, material temper variations and thermal control coating degradation). Surface cracks resulting from in-space welding techniques and from unforeseen overstressed areas can be easily detected by eddy current techniques. Material temper variation (heat damage to materials) may occur in close proximity to weldments during fabrication, repair, or as a result of rocket plumage from the course of normal maneuvering operations, or possibly as a mishap in docking maneuvers. Since most of the thermal control coatings are pigmented organic films, they may be subject to sublimation and change in inherent characteristics as a result of long term vacuum and in-space irradiation. Eddy current thickness and conductivity measurements can provide the basis for preventative maintenance. The eddy current technique has been long recognized in on-earth inspection for detecting all of these defects. Specific details of the techniques are found in the Nondestructive Testing Handbook. (Ref #55). Because of relatively simple circuitry, lightweight, and adaptability to space environment, eddy current must be considered one of the key methods for in-space application.

Experimentation of specific interest to this program has recently been performed on aluminum alloys 2014 and 2024 using eddy current techniques. Mr. W. D. Pummel, in his article "Characterization and Evaluation of 2014 Aluminum Alloy by Eddy Current Conductivity Techniques", (reference #56) discusses practical, theoretical, and experimental data relating the various heat treat tempers of this alloy to corresponding values in eddy current conductivity. The study made by Rummel is based upon changes in the form and particle size the alloy precipitate. The range of conductivity, for 2014, was determined to be as follows: (All results were based upon specimen tested at ambient conditions).

Condition	Conductivity % I.A.C.S.	Hardness Rockwell F
"O" (annealed)	48.0	57.0
T4 (solution heat treated)	32.7	82.0
T6 (solution heat treated and artificially aged)	38.0	106.0

It should be noted that a strength loss due to overaging may occur in the 2014-T6 Al alloy if the material is heated just above the normal age temperature for any length of time. This same overaging may readily take place in spacecraft structure due to rocket plume inpingement, internal fire damage or severe overheating of electrical spacecraft components. The increase of eddy current conductivity with an overaged structure is a direct function of the strength loss of the material.

Environmental factors associated with modifying the eddy current instruments for in-space use are individually discussed below:

1. Temperature Effects

   A variation in temperature, between the eddy current probe coil and the test specimen, may possibly be the largest source of error to an eddy current measurement. A change in temperature of 5°F is sufficient to change the electrical conductivity of pure metals by about 1%. A literature survey to date revealed little work done in the area of electrical conductivity changes over the -250°F to +250°F temperature range with aluminum alloys such as AA2014. To overcome the potential temperature problem, standard calibration blocks of known I.A.C.S. values are required with an in-space eddy current instrument. Samples should be of known inspace structural materials only. In flight hardware the calibration pieces may be made an integral part of the exposed surface of the instrument case, thereby being exposed to the same environment as that of the test specimen. Since the readings are based upon a relative scale, the changes in the calibration blocks would be expected to reflect the ambient temperature changes in the structures to be tested. to be tested.

2. Vacuum and O_2 Environment Effects

   Eddy current instruments can readily be packaged for use in either a 100% O_2 spacecraft cabin environment or the in-space vacuum environment. Controlling factors, in meeting these requirements, are basically proper selection of materials (components) and in maintaining critical component isolation from the environment by hermetic sealing techniques.

The demonstration unit concepted in this program and discussed later is being designed to function in a 100% oxygen atmosphere and a hard vacuum. All electrical switching will take place in the internally pressurized inert gas envelope in an effort to provide maximum safety when operating in a 100% O_2 environment at the least cost in the prototype stages of this program.

3. Electromagnetic Interference Effects

Electromagnetic Interference Control is a critical function which must be considered in the design of all space flight hardware.

The eddy current probe, by its very nature, is an RF radiator. This is the result of the probe coil operating at a 50 to 150 kilohertz frequencies. By proper cable shielding and special probe design the effects of this coil can be localized to only the area of the specimen under study.

In the design and manufacture of the demonstration unit every effort will be made to provide for EMI control. Radio frequency measurements of the demonstration unit will be conducted to assure compliance with the requirements of the manned vacuum chamber at Houston.

2.2.5 Radiography Inspection In-Space

Radiation sources most utilized in conventional on-earth nondestructive testing are "x" and gamma ray. These two sources also appear most applicable to ready conversion for "in-space" utilization. There are however, certain major advantages and disadvantages to either of these radiation sources when considering their use in the space environment.

The primary advantages of x-radiation are its ability to vary incident radiation allowing control of penetrating power, and the fact that x-ray equipment always "fails safe". By reducing incident radiation to a minimum, subject contrast can significantly be improved thereby allowing less interpretation to the operator. Because current x-ray equipment always "fails safe" (i.e. since a power source is required to produce x-radiation, a power failure results in a source failure and radiation ceases.) danger to the operator (the astronaut for space hardware) is minimized. Important also in this area is the safer storage of the radiation unit as there will be no radiation when the power source is turned off.

Certain disadvantages however do exist when space orienting conventional x-ray equipment. Currently there is no commercial off-the-shelf x-ray unit which is readily adaptable to the scope of this program. As shown by the earlier indicated method analysis, the lightest unit (still too heavy) applicable to this program is the 50 lbs Sperry unit. Though not impossible, an extremely light weight unit such as required for flight hardware could be designed, constructed and tested, with sufficient time and funding. The weight has a potential of increasing when considering the possible requirement of a separate coolant for the cathode. This is because x-rays are generated by a continuous electron flow creating a potential heat problem. This is, of course, dependent upon material, location, and thicknesses to be x-rayed (intensity requirements). Additional problem areas with x-ray for in-space use are the requirement of a high power source and scatter radiation caused by the continuous spectrum of x-radiation.

Utilization of isotopes for radiography has certain distinct advantages over x-ray, especially when considering in-space radiography. The most predominant advantage is that of the small "easily made portable" size of most isotope packages. The overall cost of isotope radiography is also generally considered less as no heavy power generator is required.

There are several disadvantages to isotope radiography however. Although extremely simple to operate (little chance for error or damage) isotope units are not "fail safe"; that is should a highly improbable failure occur and the radiographic unit shielding is damaged, the unit will continue to emit radiation until again shielded. Also since radiographic exposure time is dependent upon isotope strength, an isotope with a short half-life will require frequent exposure time increases and frequent isotope replacements. This last consideration does not realistically apply to in-space use where limited initial usage of the isotope is contemplated, regardless of half life (provided of course that the initial half life is of sufficient duration to render the isotope useful for the specific application).

Numerous isotopes have been characterized as capable of providing useable radiation but only a few are regularly used in industry and are commercially available. Some of the more common isotopes utilized for radiography include Radium, Cobalt-60, Cesium-137, Iridium-192, and Thulium-170. These sources have half-lives ranging from 70 days (Iridium-192) to 1620 years (Radium), and except for Thulium-170 have Kev energies too high for most materials utilized in structural space members. For example, aluminum alloys with their characteristically low cross section absorptivity will not require the high energies needed to penetrate 2 to 7 inches of steel. In addition most of the sections contemplated or already in use are thin (1/2" or less) indicating an even lower required Kev rating. Excessive radiation in these cases results in reduced film contrast which effectively places more reliance on operator (astronaut) interpretation.

As earlier indicated, a relatively new radioisotope, ytterbium-169, ideally suited to a program of this nature (lightweight, compact, low energy level, etc.) was decided upon. As characteristics of this isotope were previously discussed (Section 2.2.1) further mention will not be made here.

Currently available radiographic recording media consist of regular radiographic film, Polaroid film, zerox process, and electronic devices. Due to weight, size limitations, and dust contamination the zerox process is not satisfactory. Electronic devices are currently too massive and will require development to be made applicable. Regular radiographic film offers the best image quality relative to sensitivity and definition but film processing procedures are too complex to perform under space environment conditions. The remaining media, Polaroid film, appears to provide the best immediate recording media. The film may consist of a print, or a print and negative, and a single solution developing process. These are contained in a single envelope. Preliminary testing indicates that the film is useable after a two hour exposure in a vacuum of 2 X 10^{-4} torr. Visible developer pad leakage did not occur and sensitivity to x-rays was comparable to film which had not been exposed in the vacuum. Processing of the film is accomplished by conventional Polaroid techniques. No tests have as yet been conducted for film development in the vacuum environment. It is expected that protection will be needed to prevent vaporization of the developer. If protection is needed, a plastic envelope, hermetically sealed, may provide the necessary protection.

The nature of radiation absorption limits the process to inspection for defects which create a difference in material thickness and a resultant difference in the emerging radiation intensitites. Defects in this catagory are porosity, inclusions, shrinkage, lack of penetration, etc. More important are the defects which will not be detected such as cracks whose direction of propagation is not located parallel to the direction of the photon energy. Lack of fusion in weldments fits into the crack catagory and would not be detected unless favorably orientated.

Also it should be noted that the isotope chosen, ytterbium-169 is currently utilized as a conventional medical radiation source because of its soft radiation characteristics. This must be considered as a "bonus" for "in-space" use.

3.0 PRELIMINARY DESIGN CONCEPT

3.1 Qualified Components Review

To establish the degree of compliance of off the shelf equipment to space flight hardware qualifications, a thorough part by part review was conducted on the two most adaptable ultrasonic and eddy current instruments (Sperry "UCD", Budd UT 700, Uresco FC300S, and Nortec NDT-2). Preferred parts lists from NASA, Grumman (LEM PPL) and RCA were used as a basis for approval. The eddy current units contained less than 5% approved parts and the ultrasonic units contained between 30 and 35% approved parts. The approved parts from the eddy current units were an occasional transistor or diode. However, the commercial procurement method eliminated these parts as approved. The bulk of the approved parts in the ultrasonic units are Allen-Bradley carbon composition resistors. Most other initially approved parts were eliminated due to procurement methods. A complete list of parts for the selected units was sent to Apollo Parts Information Center (APIC) for screening. No new information was obtained due to the commercial procurement method of the equipment manufacturers.

Recommended substitution data was determined for the Sperry "UCD" and the Uresco FC300S units. Direct substitution of many parts is not possible. Most capacitors were not on QPL lists or were not classified by MIL or NASA specifications. Substitution of most potentiometers, inductors and connectors involves size problems and subsequent redesign of printed circuit boards. Substitution of high reliability semiconductors can be accomplished by procurement to Fairchild's FACT III series or Texas Instruments or Motorola's high -rel" series. A tabulation of component data is presented in Appendix A.

An area of specific concern is the cathode tay tube. The current available portable ultrasonic testers use cathode ray tube (CRT) means for display of measurements. This two dimensional display is required due to the nature of the instrument techniques involved in ultrasonic testing. From the stand point of future flight hardware, the CRT represents the weakest link in the entire system. A major effort, to be conducted during Phase II, is to evaluate other display means to develop design criteria for flight hardware.

Minimum requirements for high reliability flight components include:

1. Quality assurance and control of procurement having:

    a. Source control drawings and specifications

    b. Vendor source survey

    c. Material identification and traceability

2. Quality assurance and control of in-house receiving — receiving inspection (on 100% basis) to verify conformance to procurement specifications and source control documents.

3. Quality assurance and control of parts usage in-house.

   a. Segregation of components

   b. Control of received lots

   c. In-house traceability of parts ;

4. Quality and reliability tests.

   a. 100% x-ray inspection of all semiconductors.

   b. Power burn-in of critical components to cull out units susceptable to "infant mortality" (i.e. certain electronic components have higher failure rates during initial usage).

The above list represents the initial step towards flight type component verification only. In the final analysis however, the true test of the ability of components to function with high reliability is how these components function as an overall package. Environmental testing (vibration, shock, acceleration) of transistor, resistor type components, by the manufacturers, is generally to levels much lower and far less severe than a qualified flight hardware package containing such components must meet. Most "qualified-components" lists in existance today are based upon these components having passed hard environmental tests only as a discrete part of a total instrument package. For example, it has been demonstrated that quality controlled (including 100% x-ray inspection) transistors, guaranteed to only 20 g's sinusoidal vibration by the manufacturers, have successfully passed 60 g's sinusoidal and 1.5 g^2/cps random vibration when tested as part of a larger transducer package. It is the method of packaging, circuit board construction, wiring, soldering, potting, etc., that determines when small component parts can survive and function for flight requirements. Industrial electronic equipment cannot be made into space flight hardware by merely replacing internal resistors, transistors, capacitors, etc. It should be emphasized that designs for flight hardware must start from the "ground-up". Due consideration must be given to internal physical construction, mounting methods, wire routing and density of packaging. In addition, the optimum package is designed only when all the environments to be met are fully known. Once a package has been designed and controlled components built-in, then a formal qualification program must be instituted. Such a program would typically include a block of Design Verification Tests (DVT) run on production hardware. The purpose of DVT is to establish the necessary high degree of confidence that production hardware will pass in qualification tests. The DVT levels are higher than those of qualification and are intended to find the weak links in the system.

It is during DVT that potentially failure-prone components, solder or weld joints, and circuit design are culled out. A formal qualification program is then conducted to show conformance to actual program environmental and system requirements.

3.1.1 Ultrasonic-Eddy Current Integration and Simplification

The objective for this phase of the "Non-Destructive Testing For In-Space Use" program is to combine the two electronic units selected, into one package. This package will be capable of performing tests indicated in the test phase of this report. Size, weight and simplicity of operation are the main objectives. Two separate instrumentation units selected were the Rompas (URESCO) Inductest FC300S eddy current unit, and the Sperry UCD Reflectoscope, for ultrasonic tests. The following discussion describes the basic operation of each unit before modification, and the operation of the modified combined system. Block diagrams, as well as the schematics, are included and referred to in the discussion.

Eddy current (or electromagnetic induction) systems work on many different principles. The principle described is the amplitude indicating type with a one coil probe application. A difference in amplitude of the reference oscillator and the probe output is detected and indicated on a panel meter. Referring to block diagram I and schematics ES 21-1030 and ES 21-1031 will help in understanding the unit operation.

Block diagram I shows the basic blocks of an oscillator section, a probe, an indicator section and a power source.

The oscillator section of the Rompas Inductest produces a sine wave output frequency that has a range of 65 to 150 kilo-hertz. This frequency range is determined by the course and vernier frequency adjustment on the instrument front panel. When the test material is placed in the varying magnetic field (primary ac field) that is produced by the coil, eddy currents are induced in the test material. These eddy currents in turn produce an ac field (secondary ac field) in the test material that tends to oppose the primary field. This is reflected as a change of impedance which for this technique appears as an amplitude change in the signal. Again referring to block diagram I, the probe output and the reference signal are connected to the inputs of the differential amplifier. With the probe placed on the test material and located on a surface where there is no crack, the reference control of the indicator section is adjusted to give a meter or scale indication. The probe when moved to the area of a crack or change in conductivity, will pro duce a change in meter reading. Sensitivity controls gain in the differential amplifier and is adjusted for the required deflection in the meter between the normal area in the test material and area that contains a crack or change in conductivity. The system is powered by three 8.4 volt dry cell batteries with a life of 90 hours.

Ultrasonic Reflectoscope

The ultrasonic reflecting technique is a method used to detect discontinuities or boundaries of different ultrasonic properties. To better understand the operation of the Sperry "UCD" Reflectoscope, which was selected for this study, a brief description is presented along with the accompanying block diagram II and Sperry schematics 50D397 and 50D398. The operation starts with a synchronous pulse from the clock section contolling the pulser network. This pulse fires a silicon controlled rectifier in the pulser

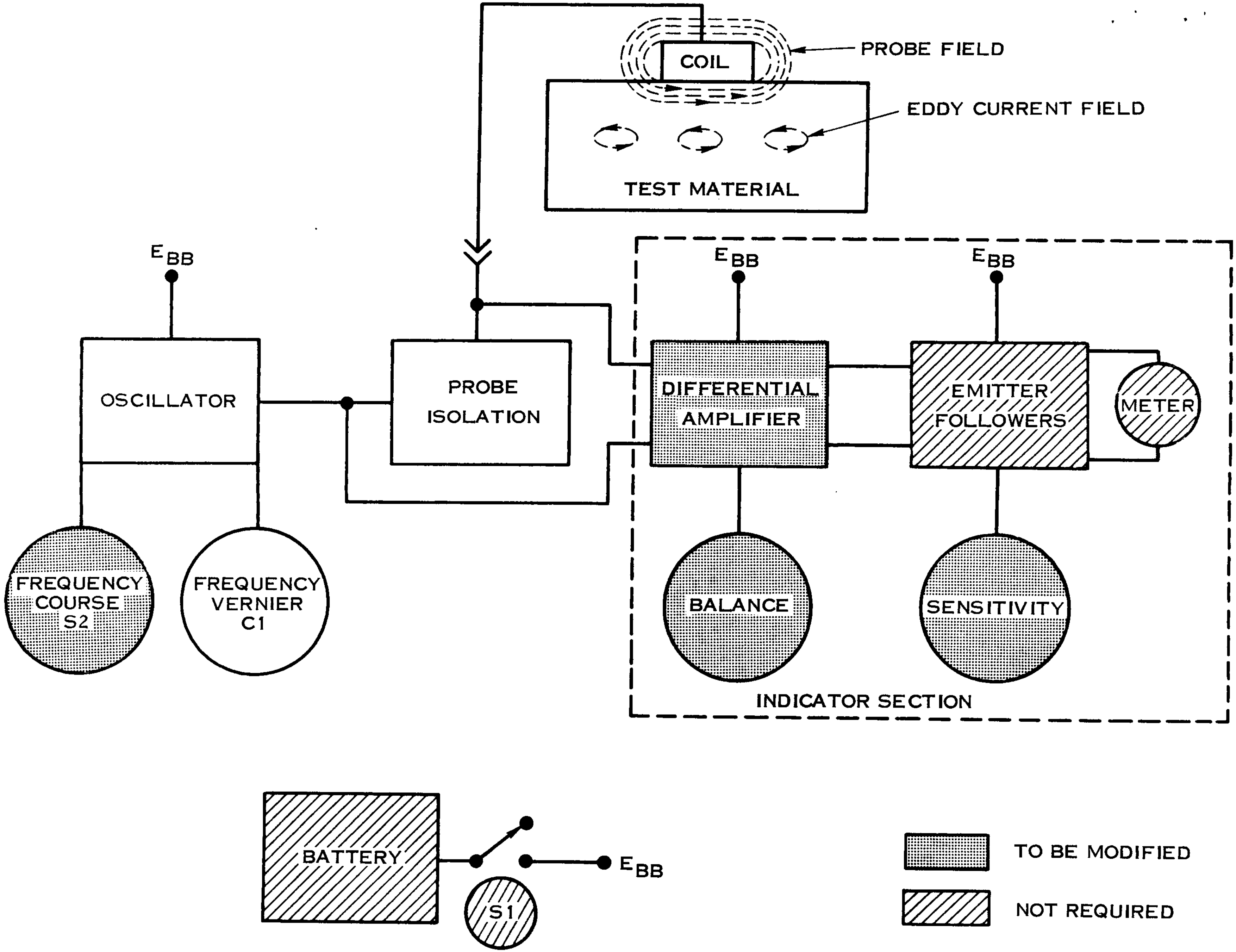


FIGURE 10 BLOCK DIAGRAM I EDDY CURRENT UNIT
ROMPAS INDUCTEST 'FC300S'

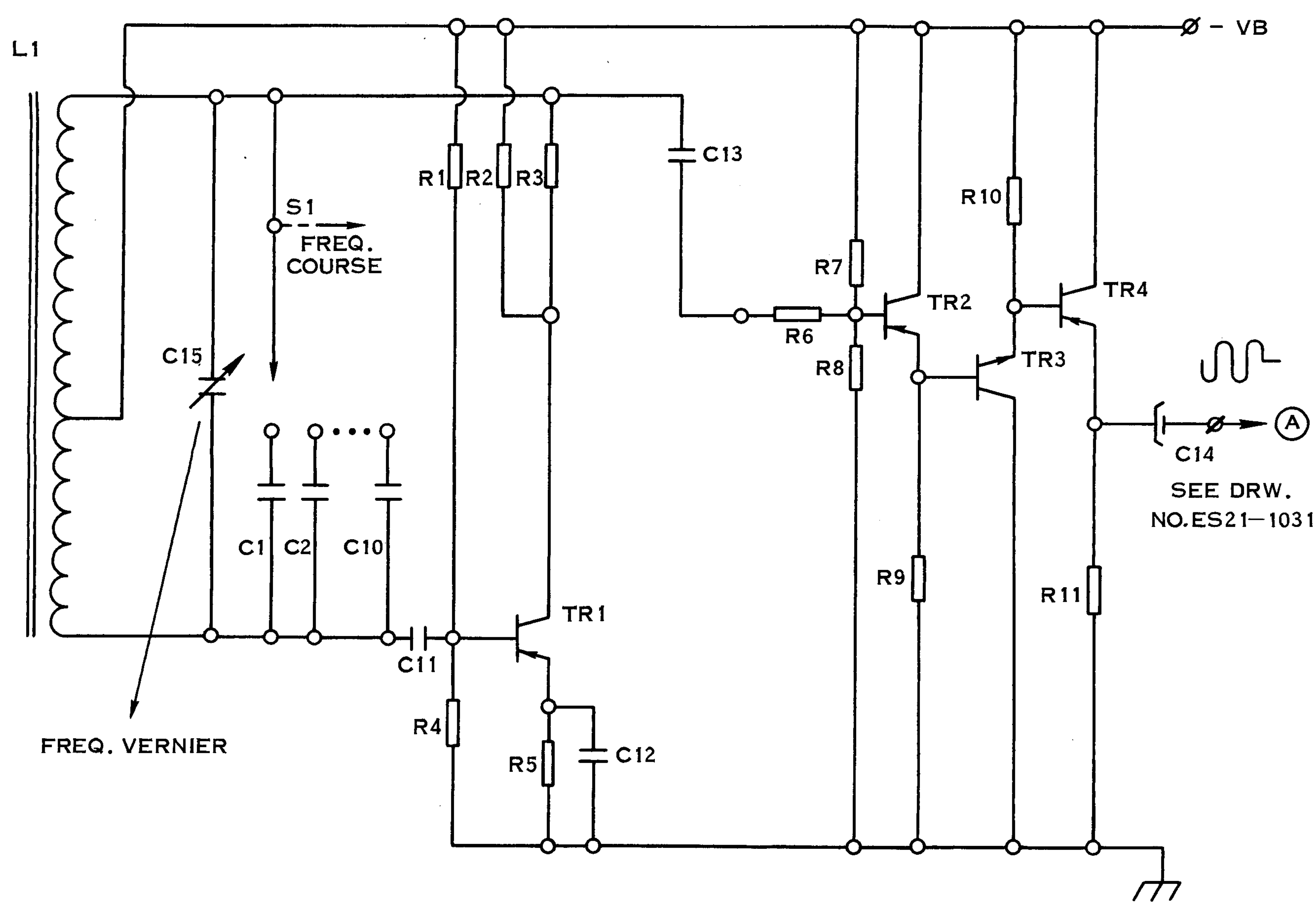


FIGURE 11 FREQUENCY OSCILLATOR
INDUCTEST FC300S DWG NO.ES21—1030
ROMPAS ULTRASONICS

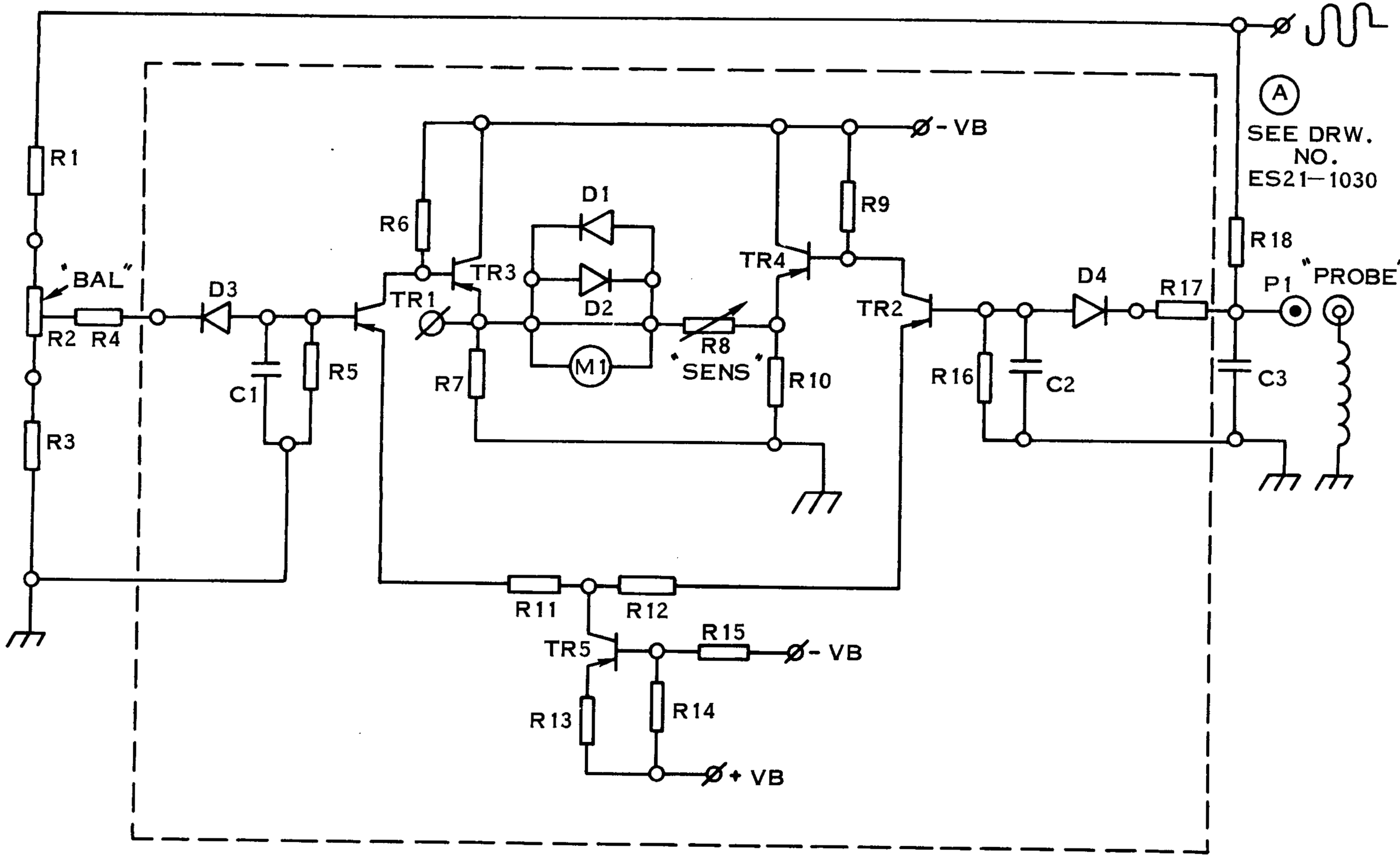


FIGURE 12 DIFFERENTIAL AMPLIFIER

INDUCTEST FC300S DWG NO. ES21—1031

ROMPAS ULTRASONICS

block which then discharges a capacitor through the pulser coil. The frequency selector switch of the pulser determines at which frequency the pulser will ring; these frequencies are 1, 2.25 and 5 megahertz. The output of the pulser is fed to the search unit and to the receiver section. For discussion purposes it is best to describe the operation with a single search unit and later explain the purposes of a two search unit method.

For single search unit operation the "NORMAL- THRU" switch (to the right of the pulser blocks) will be in the "NORMAL" position.

In this operation the transducer functions as both the transmitting and the receiving transducer. Shown on Diagram II is such a probe. This probe is connected to connector R on the main unit. This initial pulse is also present at the input of the receiver section. The duration of this pulse is 2 millionths of a second or less, and at the end of this period the crystal vibration decays rapidly to zero. The time that it takes for the initial pulse to return to zero essentially defines the resolution capability of the instrument. For thin space type materials, higher resolution is required than is available in this instrument. The initial pulse and the reflected pulses are amplified by tuned radio frequency amplifiers in the receiver section. The radio frequency section consists of 4 stages of tuned amplifiers with a frequency range of 1 to 5 megahertz. Frequencies up to 15 mhz are necessary for thin space type materials. The signals are detected and applied to the vertical deflection amplifier which drives the vertical plates of the cathode ray tube display.

The sweep section of the unit provides the horizontal drive for the horizontal deflection plates of the cathode ray tube. This is a typical sweep circuit normally found in standard oscilloscopes where a signal is generated and produces a horizontal trace on a cathode ray tube. The sweep length control for the Sperry unit is calibrated in inches with the range being 1 to 200 inches selected in 8 steps. This range is greater than needed. The system power for the Sperry unit is a 8.7 volt silver cadium battery, regulating section, and an inverter to produce the required operating D.C. voltages.

For through transmission technique, one search unit is connected to connector T, and acts as the transmitting transducer and the other search unit is connected to connector R and acts as the receiving transducer. With the "NORMAL-THRU" switch in the thru position, the pulser now drives the transmitting search unit and the reflected pulses are fed to receiving search unit. The remaining functions are the same as previously described.

Block diagram III represents the proposed mechanical and electrical modification of the Sperry UCD reflectoscope and the Rompas Inductest FC300S to combine both units into one package in order to perform the eddy current and ultrasonic type testing with maximum simplicity. The two units will now be referred to as "Ultrasonic" for the Sperry Reflectoscope and "Eddy Current" for the Rompas Inductest.

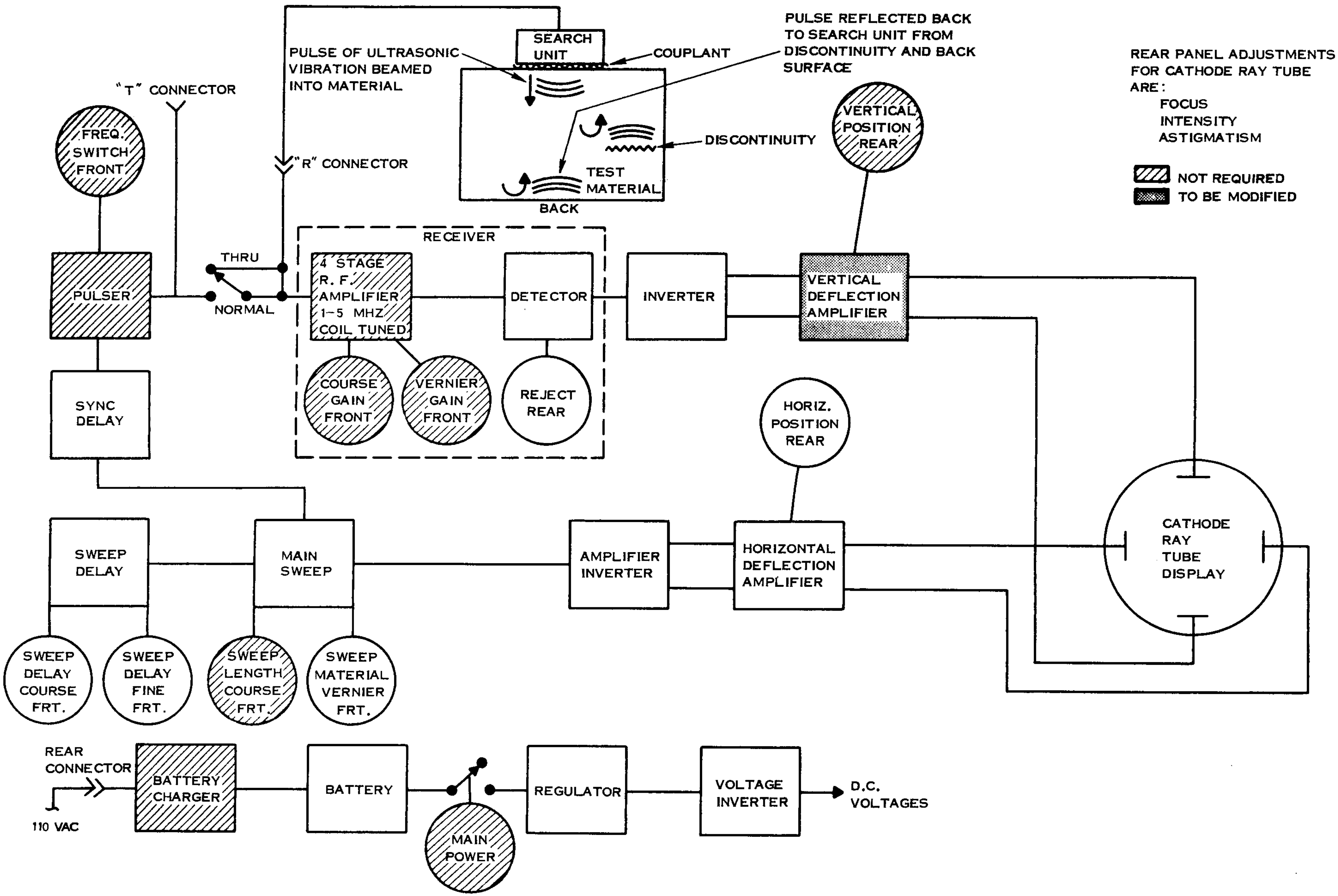
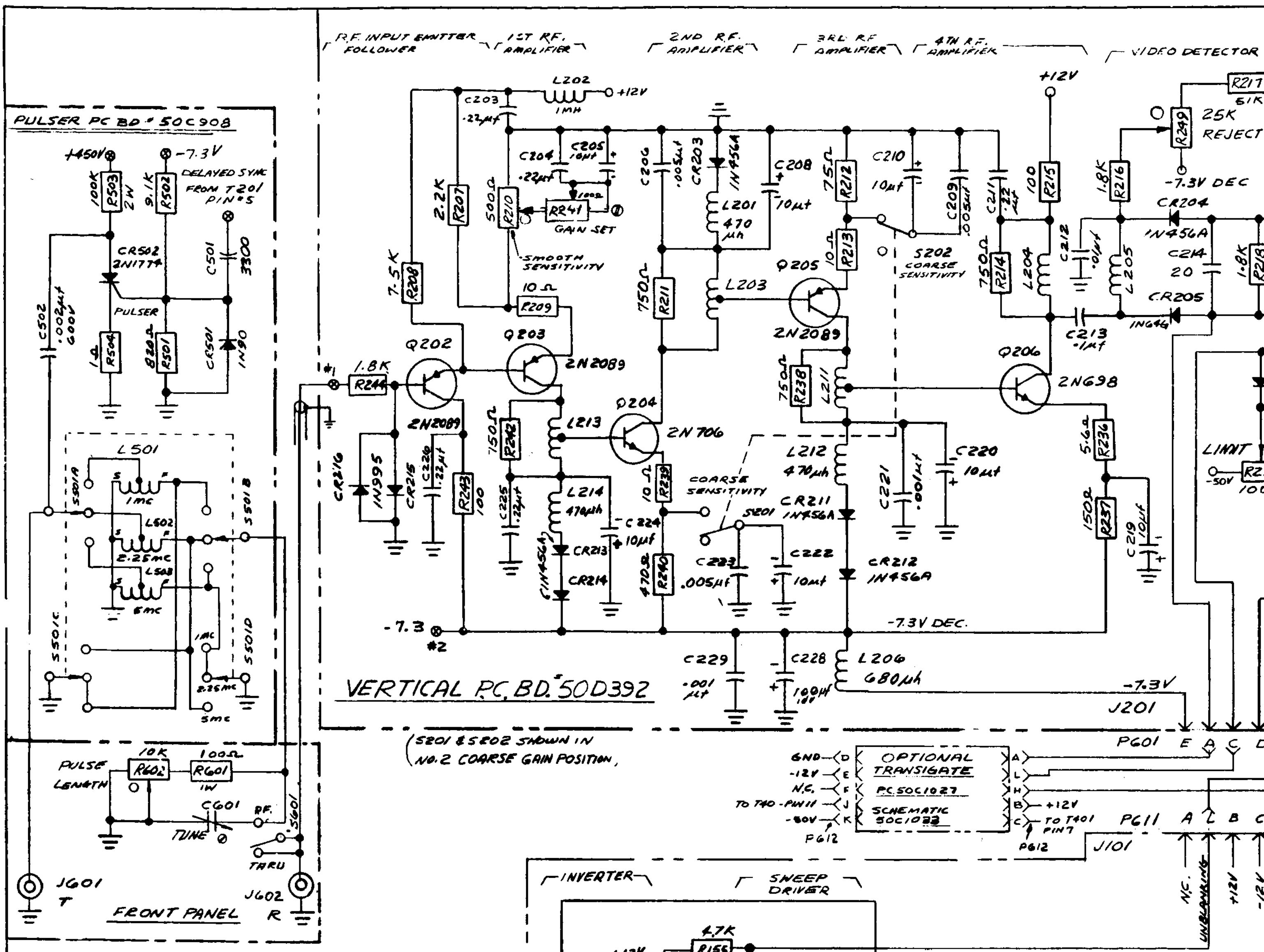


FIGURE 13 BLOCK DIAGRAM II ULTRASONICS UNIT SPERRY REFLECTOSCOPE UCD



NOTES:

1. UNLESS OTHERWISE SPECIFIED
   a. ALL RESISTORS ARE ½W
   b. ALL CAPACITORS ARE PF
   c. ⊗ = INDICATES SOLDER TERM.
   d. "600" SERIES INDICATES FRONT PANEL

LAST NO. USED	
R253	R160
C231	CR115
L214	C117
V208	Q115
Q218	J101
CR217	R604
R504	C601
C502	S603
CR503	J602
L503	V601
S501	

NO'S NOT USED			
Q111	R157		
Q112	R159		
	R152		
CR113			
Q110			
R107	R247		
	CR503		
	C218		
R153	L207		
R154			
R156			

FIGURE 14.

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Sperry Products
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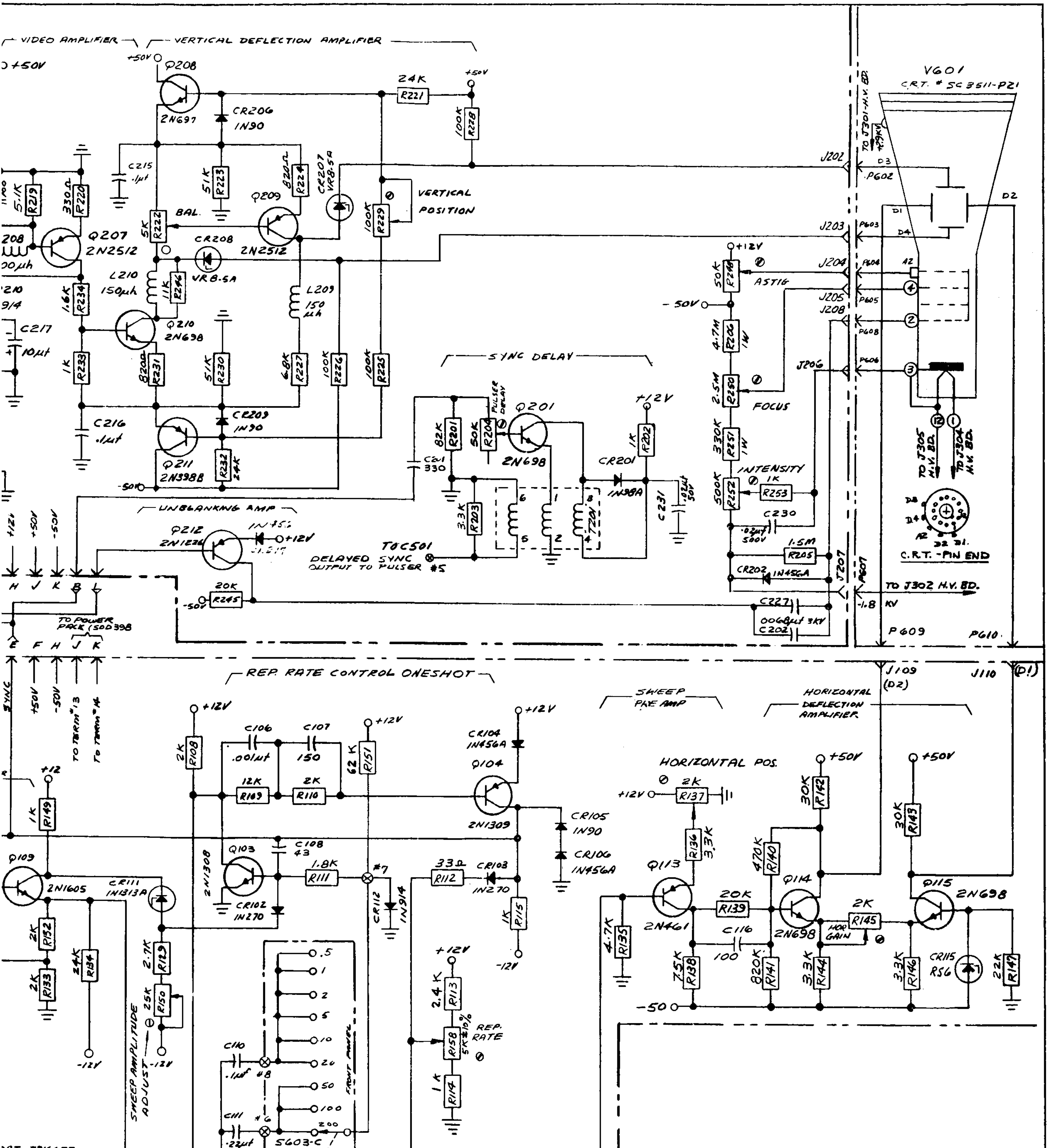

ITEM NO.	DWG. OR PART NO.	NAME	QUAN REQ.

BILL OF MATERIAL AUG 1 8 1965

50C901 — REF. ASS'Y. NO.

SPERRY PRODUCTS
DIVISION OF AUTOMATION INDUSTRIES INC.
Danbury, Connecticut, U.S.A.

U.C.D. VERTICAL AND HORIZONTAL SCHEMATIC

50D397

Geo Chabor 9-11-64

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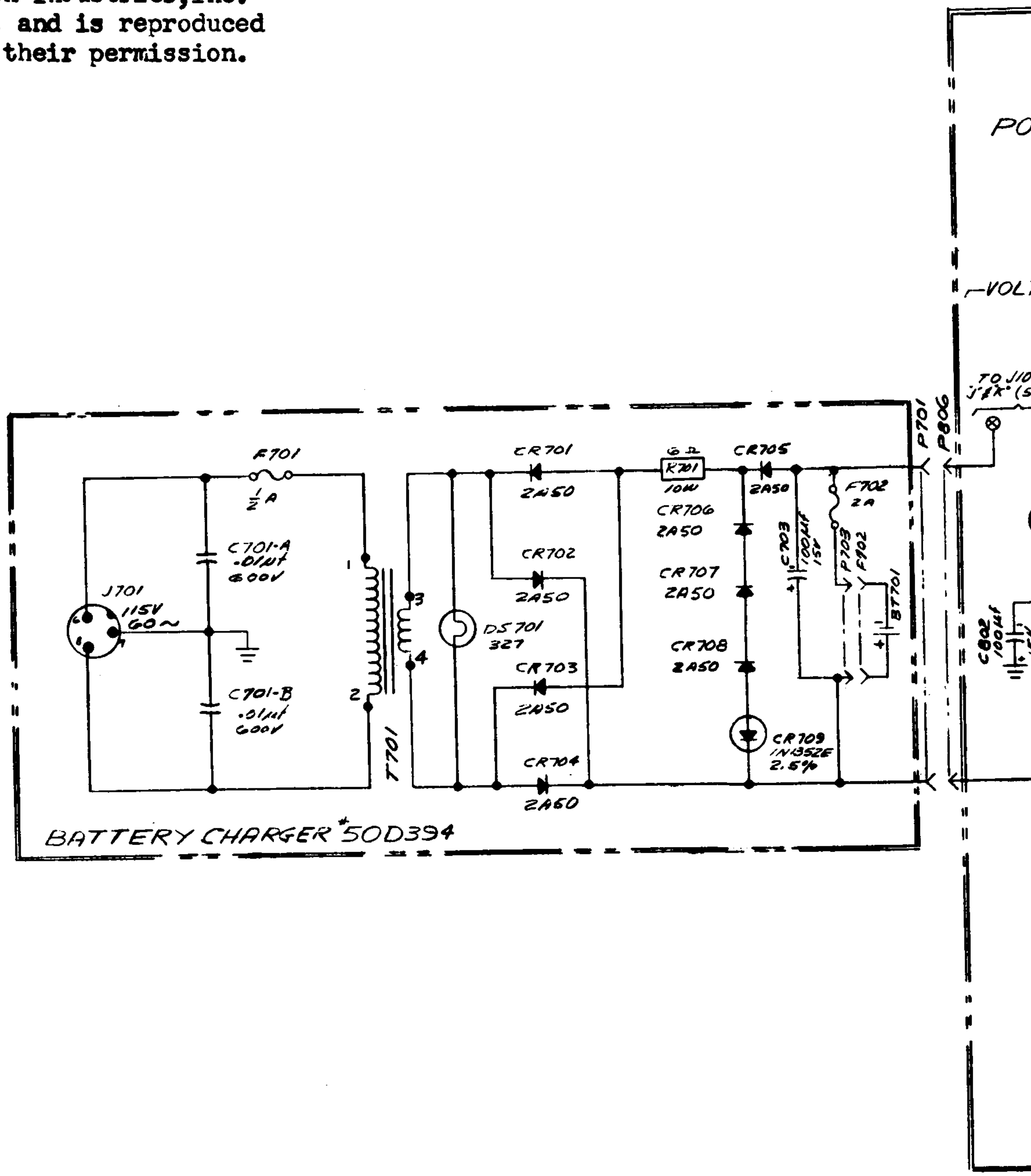
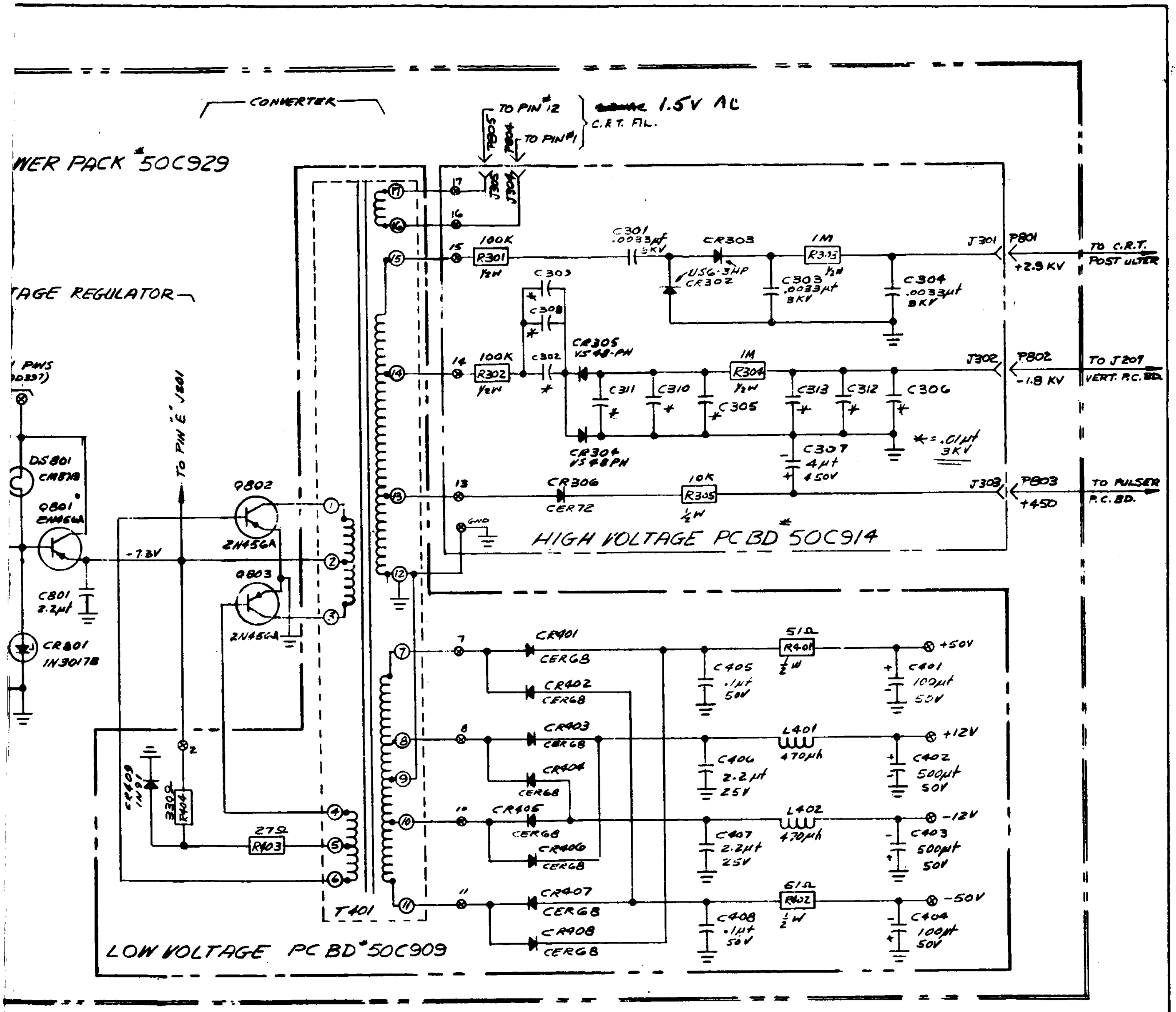


NOTE
1. ⊗ = INDICATES SOLDER TERMINALS ON PC. BD.

FIGURE 15.



ITEM NO.	DWG. OR PART NO.	NAME	QUAN REQ.
		BILL OF MATERIAL 7 8 1965	

SCALE 50C901
REF. ASS'Y. NO.
EXCEPT AS NOTED
DEC. ±
FRAC. ±
ANGLES ±
MACH. ✓
MAT. OR P. SPEC.
FINISH
CHG. REF.
TIMES USED

SPERRY PRODUCTS
DIVISION OF AUTOMATION INDUSTRIES, INC.
Danbury, Connecticut, U.S.A.

POWER SUPPLY SCHEMATIC
(UCD)

SPERRY 50D398

Geo Chaaber 3-19-64
DRAWN BY
CHECKED BY
APPROVED BY
MFG CHECK BY
DATE

50D398

The pulser section of block diagram II consists of a high energy pulse which rings a tuned circuit to produce the required operating frequency to drive the crystal probe of the ultrasonic section. Modification is required to increase the present upper frequency range to 15 megahertz (presently 5 megahertz) and to eliminate the frequency selector controls. This modification would eliminate the tuned coils presently used and therefore reduce one possibility of electromagnetic interference.

The Radio Frequency receiver section of the ultrasonic unit also has a frequency response limitation of 5 megahertz. Replacing this section with an integrated circuit type amplifier that has a frequency response of at least 15 megahertz would decrease the physical size of the package and eliminate some additional coils and components which are a potential radiation problem. The high frequency response is needed for the higher resolution. Separate display indicators are presently used for the Ultrasonic and Eddy Current systems. Modification of the Ultrasonic vertical deflection amplifier to accept signals from either the Eddy Current or Ultrasonic units would eliminate this duplication of indicating devices. Such a modification is shown on block diagram III. This includes the differential amplifier block and sections S1D and S1E of switch S1. Included within the differential amplifier are the sensitivity and balance controls. The sensitivity control is primarily used for the eddy current unit to adjust the level of the differential amplifier. The Eddy Current indication will be a vertical shift of the horizontal trace on the cathode ray tube. Zero positioning of the Ultrasonic and balancing of the Eddy Current unit is performed by the one "Balance" control on the front of the modified system. The display for the Ultrasonic portion of the system will be a series of vertical pulses, (start, discontinuity and back) along the horizontal axis.

Evaluation of the power supply requirements for a combination ultrasonics/eddy current package has been completed. Results of this study indicate that the entire NDT demonstration unit may be powered by the same battery as is currently supplied with the Sperry unit. This study has been directed only towards the demonstration unit requirements. It is expected that the design for flight hardware will not require as much power as the current available equipment.

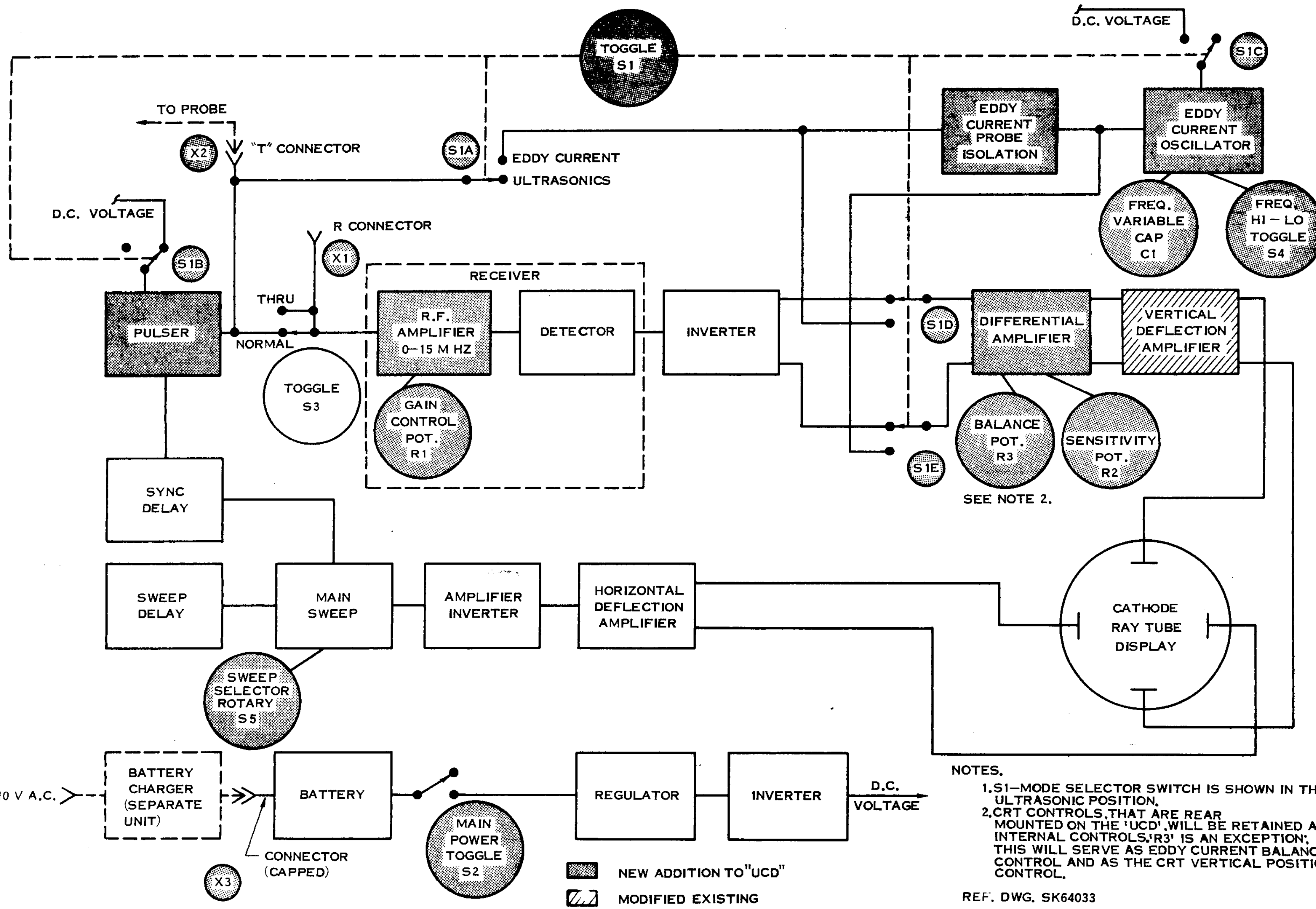
The battery supplied with the Sperry UCD Reflectoscope is a rechargeable type. A 110 VAC charger is built into the Sperry unit. The specification of this battery is as follows:

Type: Yardney 8XYS5, silver cadmium

Voltage Output: 8.7 volts DC at 1 ampere current drain

Life: 5 ampere hours

The "UCD" draws 670 milliamperes of current at 8.7 volts DC.



NOTES.

1. S1–MODE SELECTOR SWITCH IS SHOWN IN THE ULTRASONIC POSITION.
2. CRT CONTROLS, THAT ARE REAR MOUNTED ON THE 'UCD', WILL BE RETAINED AS INTERNAL CONTROLS. 'R3' IS AN EXCEPTION; THIS WILL SERVE AS EDDY CURRENT BALANCE CONTROL AND AS THE CRT VERTICAL POSITION CONTROL.

REF. DWG. SK64033

FIGURE 16 BLOCK DIAGRAM III IN–SPACE NONDESTRUCTIVE TESTING ULTRASONIC–EDDY CURRENT UNIT

The Uresco (Rompas) Inductest FC300S uses (3) 9 volt DC (nominal) mercury batteries wired in parallel. The specification for these batteries is as follows:

Type: Mallory TR146X

Voltage Output: 8.4 volts

Life: 350 milliampere - hours

This unit draws 37 milliamperes (at any operating frequency) at 8.4 volts DC. The FC300S is capable of being operated at 8.7 volts DC with no decrease in performance.

Both the "UCD" and the FC300S may be powered by the 8.7 volt battery. The expected current drain of the two combined units will not exceed 707 milliamperes.

The battery life, before recharging, would be:

$$\frac{5 \text{ ampere-hours}}{0.7 \text{ ampere}} = 7.1 \text{ hours}$$

An operating life of 7.1 hours before charging is more than adequate for the purposes of the demonstration. For flight hardware, however, it may be advisable to consider a longer-life battery. In addition, evaluation of hermetically sealed batteries (similar to the Yardney battery now used in the EMU Back Pack) would be required. In the demonstration unit the battery will remain with the electronics package in the sealed container. The built-in charger unit, now in the Sperry unit, will be removed and used as an outside device when ground charging is required. The use of a built-in charger would be excess and redundant in the breadboard demonstration unit. Further evaluation of the requirements for a built-in charger will be made for the design of flight hardware.

A review of the new configuration control functions are as follows:

Central Function

I. - Switch S1 - "Eddy Current - Ultrasonics" Toggle

S1A - Selects the proper excitation to the probe

S1B - Controls the operation of the pulser for the reflectoscope section.

S1C - Selects power for Eddy Current oscillator

S1D and S1E Select the display, Eddy Current or Ultrasonic

II. S2 - "Master power" (toggle) Power for the complete system. May possibly be combined with VI.

III. S3 - "Normal-Thru" (toggle) Selects the probe application for the ultrasonic

IV. S4 - "Hi-Low" (toggle) The frequency selector for the Eddy Current oscillator

V. S5 - "Sweep" Sweep selector for the scope, 4 Positions, .5, 1.0, 5.0 and 10 inches

VI. R1 - "Gain" (Pot.) Sensitivity for the receiver section. (ultrasonic)

VII. R2 - "Sensitivity" (Potentiometer) Gain control for the Vertical deflection Amplifier (Eddy Current)

VIII. R3 - "Balance" (Potentiometer) Balance control in the Vertical Amplifier. (ultrasonics and eddy current)

IX. C1 - "Vernier" (capacitor-rotary) Vernier frequency control for Eddy current oscillator.

All controls that will not require adjustment while the system is in operation in the chamber will be included in the system but will be made internal controls. These include, the horizontal positioning, focus, intensity, astigmatism, and the reject. The vertical positioning rear control will be made a fron control (R3 above) and will serve as the vertical position and the Balance Control (for Eddy Current Unit).

3.2 Preliminary Thermal Analysis

3.2.1 Thermal Protection Considerations

Thermal analysis of the NDT package in the space environment can be divided into two areas; heat transfer within the package, and heat exchange of the package surfaces with its environment. Although transfer of heat within the package can occur by radiation, conduction, or convection, the only means by which the package can exchange heat with its environment is by radiation. Thus, for passive temperature control of the package, that is without an integral environmental control system, excess heat must be eliminated by radiation to space. This exchange of radiant heat transfer between the package and its environment must be controlled, at least passively, to maintain the unit and component temperatures within tolerable limits. It has been found that for low internal power dissipation, such as an NDT package, the surface temperature of the package, as determined by the net radiant heat exchange with the space environment, is of primary importance in regulating the internal component temperatures. For this reason a preliminary space radiation analysis has been conducted to estimate expected surface temperature extremes and assess the possibility of passive temperature control of varying surface radiation properties for selected earth orbiting conditions.

The surface temperature of an object in orbit about the earth is determined by the heat balance between the surface of the object, and its environment, which includes the sun, earth and its atmosphere, any other nearby objects, such as space vehicles, and space itself which acts as a heat sink. Inherent to these radiant heat flux exchanges are several variables which significantly influence the surface temperature of an orbiting object. These variables, and their effects on radiant heat exchange, are summarized and discussed below:

Variable	Description
Orbit height	Distance from planet
Relative orientation	Position of surface relative to various radiant heat sources, direct and reflected
Orbital position	Angular location in the orbital path relative to radiant heat sources
Internal heat load	Direct heat transfer to surface from interior of surface
Thermal mass	Total heat capacity per unit area of surface
Surface finish	Absorptivity of radiant heat flux at solar wavelengths and emissivity of radiant heat flux at infrared wavelengths
Solar constant	Intensity of solar heat flux at orbit location
Albedo	Reflectivity to solar radiation

For an earth orbit the solar constant, S, and the earth albedo, a, can be assumed to be constant and for analytical work are normally taken as $S = 440$ Btu/hr-ft^2 and $Q = 0.35$. The orbital height, relative orientation, and orbital position are the most influencial variables on the surface temperature but also the most difficult to control or alter. These surface location variables influence the percentage of direct solar radiation, the direct thermal radiation from the earth or other orbiting objects, and reflected solar radiation, and are normally expressed analytically by geometry factors. For the NDT package the internal heat dissipation per unit area of surface is sufficiently small to have little effect on the surface temperature, but the thermal mass per unit area is of sufficient magnitude to level out temperature oscillations due to cyclic variations in the radiant heat flux from the environment. Surface finish can be altered to greatly influence the surface temperature by adjusting the percentages of incident heat flux absorbed and emitted by the surface, and is the only variable available for thermal control of the NDT package aside from a separate environmental control system. The expected influence of these variables on NDT package surface temperature have been estimated in a preliminary thermal analysis which is reviewed in the following paragraphs.

For simplicity and because of the multitude of situations the NDT unit was assumed to be an isolated orbiting object, with no influence from a nearby space vehicle or astronaut. The orbit was considered circular at a height of 105 nautical miles (120 miles) in a plane passing through the earth on sun centers. This results in the maximum percentages of incident solar and earth emitted radiant fluxes.

An energy balance of the surface of an orbiting object can be written as

$$\frac{dE_I}{dt} = q_{SR} + q_{E(SR)} + q_{E(TR)} + q_I - q_{TR}$$

where

$\frac{dE}{dt}$ = the rate of change of internal energy of the object or surface with time

$= MC_P \frac{dT}{dt}$

q_{SR} = direct solar radiation absorbed by surface

$= \alpha_S \mathcal{S} F_{SR} A \sim$ Btu/hr

$q_{E(SR)}$ = solar radiation reflected from earth and absorbed by surface

$= \alpha_S \mathcal{S} a F_{E(SR)} A \sim$ Btu/hr

$q_{E(TR)}$ = thermal radiation from earth absorbed by surface

$= E_t \, \varepsilon \, F_{E(TR)} A \sim$ Btu/hr

q_I = rate of internal heat transfer to surface - Btu/hr

q_{TR} = heat radiated from surface

$= \sigma \varepsilon A T^4$ - Btu/hr

t = time $\sim$ hr

T = temperature of surface - ^{o}R

A = surface area - ft^2

m = mass of object or surface - lbm

C_p = specific heat of object - Btu/lb - ^{o}R

a = reflectivity of earth in solar wavelength range

α_S = absorptivity in solar wavelength range

$\mathcal{E}$ = emissivity in infrared wavelength range

$\mathcal{S}$ = solar constant = 440 Btu/hr-ft^2

E_t = effective earth planetary emission = 66.4 Btu/hr-ft^2

σ = Stefan-Boltyman constant

= 0.1714×10^{-8} Btu/hr-ft^2-$^{o}R^4$

Substituting into the energy balance equation and rearranging terms gives

$$\frac{M}{A} - C_P \frac{dT}{dt} = \alpha_S \mathcal{S} \left(F_{SR} = a F_{E(SR)} \right) = \mathcal{E} E_t F_{E(TR)} + \frac{q_I}{A} - \sigma \mathcal{E} T^4$$

Note that the first two terms on the right represent the radiant energy from the space environment to the surface. The variation of these two environmental radiation flux terms were computed for two different orientations of a cylindrical surface in the assumed earth orbit and are shown in Figures 17 and 18 . These orientations were chosen because they represent the extremes in terms of the maximum and minimum contributions of the solar radiation flux. It is important to realize that the solar radiation flux is zero on the shade side of the earth orbit and the surface only receives earth emitted thermal radiation if it is facing the earth.

The maximum radiant flux of the two orientations was used to estimate the maximum expected surface termperature for various surface finishes by solving the energy balance equation without the thermal lag term, $\frac{M}{A} \; C_P \frac{dT}{dt}$ (i.e. instantanious temperature response). An internal heat transfer per unit area of 7 watts/sq-ft was conservatively used for these calculations.

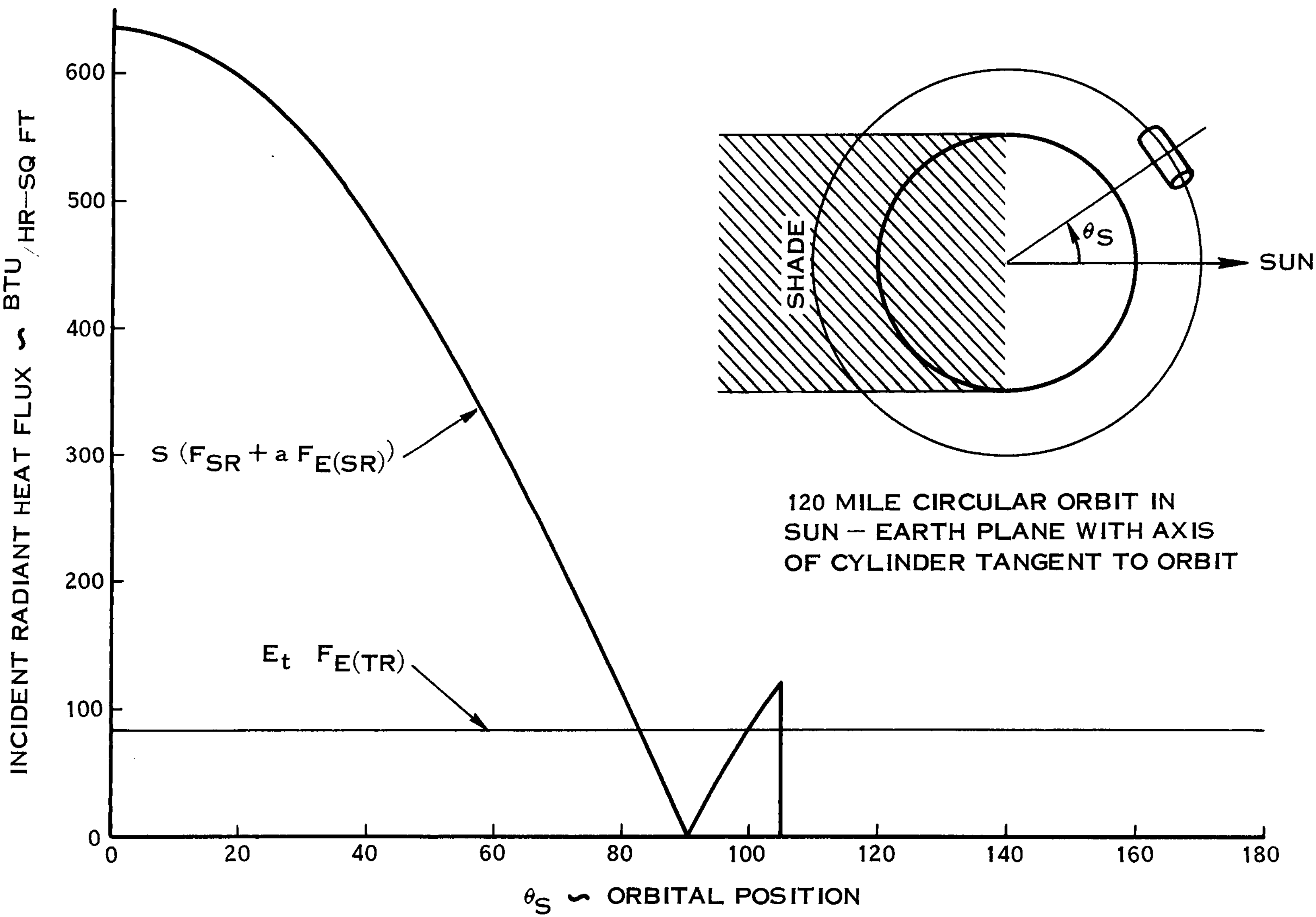


FIGURE 17 . RADIANT HEAT FLUX FROM SUN AND EARTH INCIDENT TO THE CONCAVE SURFACES OF AN EARTH ORBITING CYLINDER

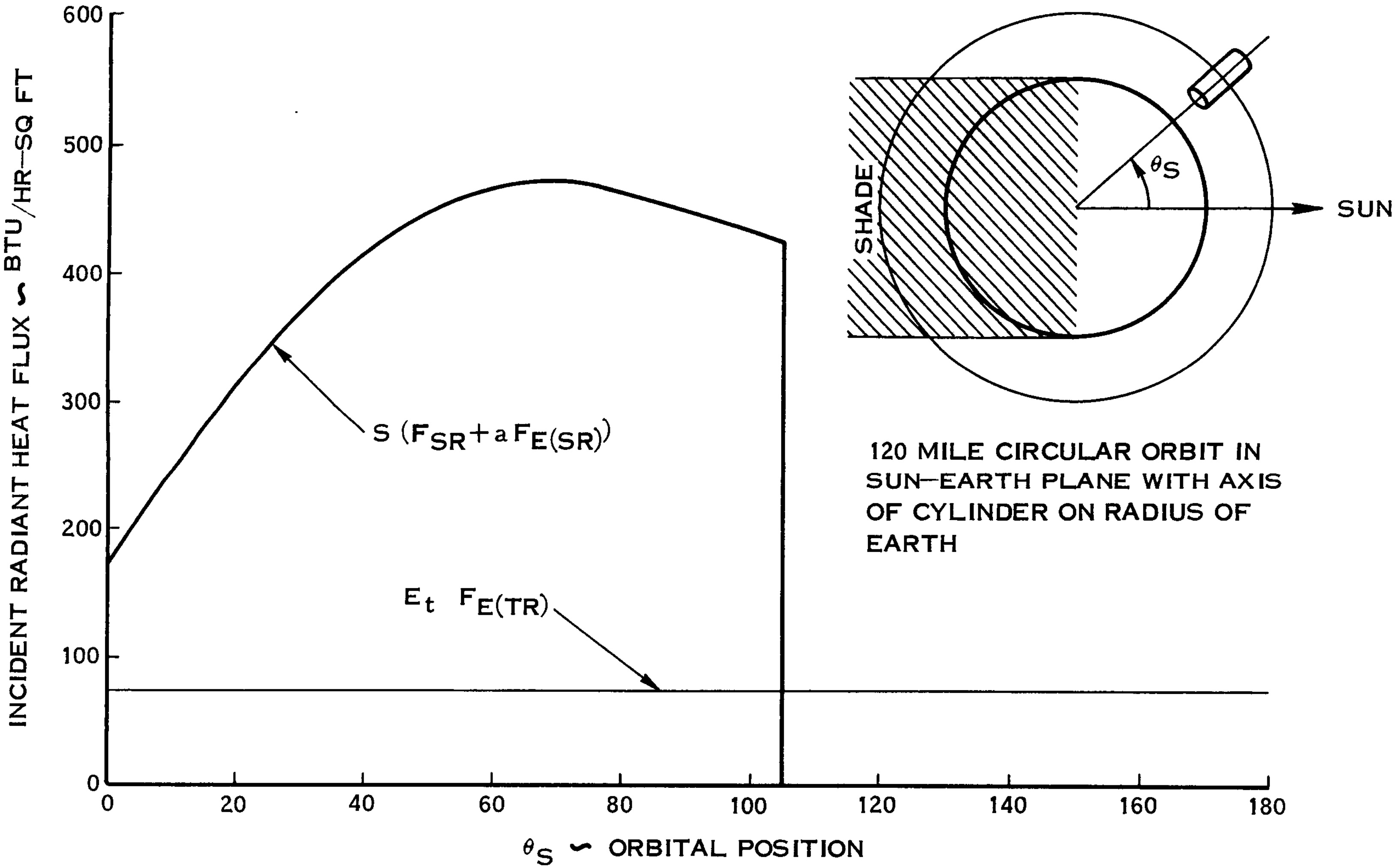


FIGURE 18 . RADIANT HEAT FLUX FROM SUN AND EARTH INCIDENT TO THE CONCAVE SURFACES OF AN EARTH ORBITING CYLINDER

Surface Finish	α_s	ϵ	Maximum Surface Temperature - ^oF
Anodized Aluminum	0.15	0.77	149
White (Z_nO) Epoxy Paint	0.25	0.88	180
Oxidized Stainless Steel	0.89	0.75	385
White (Z_nO) Pottasium Silicate Paint	0.159	0.925	138

These tabulated maximum surface temperatures indicate that by proper selection of the surface finish overheating of the NDT unit can be prevented. This is a conservative statement since the calculated temperatures are for the surface receiving the maximum radiant influx and do not account for temperature damping due to the thermal mass effect. To account for variations in the radiant influx over the surface and the thermal mass effect involves a very complicated analysis which will require a digital computer program.

With regard to the minimum expected surface temperature the radiant influx and internal heat transfer were taken as zero which permits analytically solving the energy balance equation while accounting for the thermal mass. This situation is similar to orbiting on the shade side of the earth with the surface facing deep space. The solution for the surface temperature as a function of time under the conditions just described can be written as

$$T(t) = \left[\left(\frac{1}{T_i} \right)^3 + \frac{3\sigma\epsilon t}{M/A\ C_p} \right]^{-1/3}$$

where T_i is some initial temperature taken at the instant the surface enters the earth's shade and all influx goes to zero. This expression shows that for a fixed period of time without radiant heat influx the change in surface temperature can be controlled by proper selection of the surface emissivity and the thermal mass of the surface. Using a conservatively estimated mass per unit area for the NDT unit surface of 3.36 lb/sq-ft, the specific heat for aluminum of 0.214 Btu/lb-^oR, a shade period of 150^o of orbit which is 36.7 minutes of each 88 minute orbit, and an initial temperature of 80^oF which is a typical operating level for electronic components; the variation of surface temperature with emissivity was computed. The results of these computations shown in Figure 19 illustrate that by providing the proper surface emissivity the minimum temperature of the NDT package can be controlled to reasonable levels for satisfactory operation of the various electronic components involved. These estimates represent greater temperature change than the components inside the package will experience since the effects of internal heat paths was not represented. If a higher initial temperature had been used in this analysis the resulting change in temperature during the shade period of an orbit would have been less.

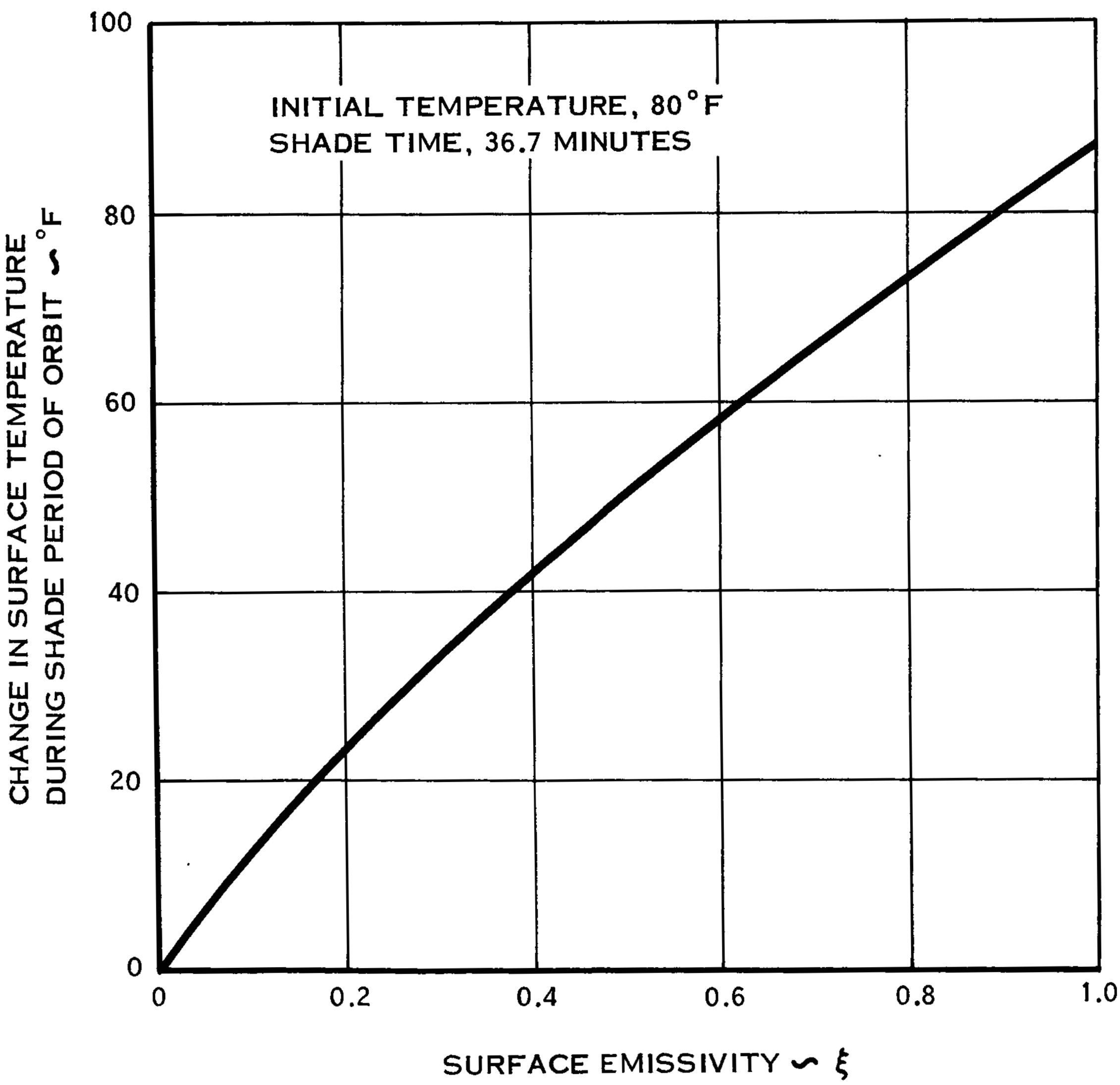


FIGURE 19 ESTIMATED MAXIMUM CHANGE IN NDT UNIT SURFACE TEMPERATURE WITH SURFACE EMISSIVITY

This preliminary radiation analysis shows in general what can be done to control the temperature of hardware in the space environment by providing selected surface finishes with proper radiation characteristics. However, because of the many variables inherent to the space radiation problem more exact analyses based on the expected environments of the NDT unit must be conducted before an optimized thermal design can be defined. These more exact analyses should be conducted for various expected conditions both outside the space vehicle and in storage. The analyses should account for radiations in the radiant flux over the NDT unit surfaces, and the thermal lag due to cyclic variations in the radiant influx.

The results of the preceeding preliminary analysis will be used in the following sections to discuss the specific thermal control considerations of the individual NDT unit components.

3.2.2 Probe Thermal Control Considerations

The probe temperature is important since the piezoelectric effect of the probe crystal can be influenced by temperature extremes. The temperature of the surfaces being inspected could also be of significant importance. These temperatures should be evaluated analytically such that operation in the space environment can be assured. This analysis may indicate that a protective cover or an integral electric heater may be required during certain phases of a space mission to properly regulate the probe temperature.

The fact that no heat is dissipated in the probe would tend to reduce the estimated maximum temperatures, and for an anodized aluminum surface a maximum temperature of $130^{o}F$ is not unlikely. In addition, the thermal mass effect of the probe will be considerably greater than that assumed in the preliminary analysis, tending to reduce the temperature drop during the shade phase of an earth orbit.

3.2.3 Ultrasonic-Eddy Current Package Thermal Considerations

This portion of the NDT unit will require the greatest thermal analysis effort since it involves electronic components and power supply that must be maintained within certain temperature limits during operation and storage. The preliminary analysis is typical of this portion of the unit and represents what temperature extreme can be expected with proper thermal control surfaces. However, specific anticipated environments should be evaluated by a more exact analysis. The objectives and scope of these recommended thermal analysis is discussed separately below for the demonstration package and actual space flight hardware.

a. Demonstration Package

The preliminary analysis did not include a heat transfer analysis of the interior of the NDT unit on the assumption that the package surface temperature controls the temperature of the components within. This is true if adequate direct conduction, radiation, and convection paths are provided to transfer heat generated by components to the package surface. For the demonstration package a heat transfer analysis will be conducted in the second phase of this study. Direct conduction paths will be utilized whenever possible to transfer the heat generated to the package surfaces, but it may be necessary to pressurize the package with a gas, such as nitrogen, to get proper heat dissipation. These analytical results and the test data which will be derived from the demonstration package should be of value in defining the internal heat dissipation effectiveness for specification of the surface temperature limits of space hardware.

b. Space Flight Hardware

NDT hardware capable of in space operation and storage will require considerably more analytical thermal design effort than the demonstration package. This thermal analysis should be twofold, namely; internal heat dissipation design and external surface temperature control through radiation analysis. It will be necessary to conduct detailed analyses of the unit in the following space environments and related conditions outlined below:

Storage Environment:

1.) Radiation influx

2.) Compartment temperature

3.) Compartment pressure

4.) Mounting structure

5.) Compartment thermal conditioning

Extravehicular Environment:

1.) Orientation of unit relative to sun, earth, and space vehicle

2.) Temperature of space vehicle

3.) Radiation properties of space vehicle

A detailed space radiation computer program capable of handling the above situations has been written and is currently in use for analyzing the Apollo program Portable Life Support System in all of the anticipated environments. Assuming that this study program will lead to the development of NDT space hardware this computer program would be available for the necessary space radiation analysis.

3.2.4 Radioisotope Package Thermal Consideration

There should be very few problems with the Radioisotope package as a result of exposure to the thermal effects of a space environment. The Radioisotope is quite insensitive to temperature extremes compared to the other components, and the very large thermal mass per unit area should produce a substantial damping effect on temperature fluctuations due to cyclic variations in the radiation influx. However, as part of the thermal design for space hardware the radioisotope package should be analyzed as outlined above for the Ultrasonic-Eddy Current package.

3.3 Vibration Testing Evaluation

In order to obtain meaningful data from most vibration tests, it is necessary to test hardware that at least closely resembles the final configuration. This ensures that response of components within the package are realistically related to that of the final configuration.

If this similarity is not maintained, values of resonant frequencies and transmissibilities obtained will vary to the extent of clouding the ultimate design requirements. Resonant frequencies and transmissibilities vary according to the means of attachment and/or mounting configuration employed. This fact alone present a strong case for testing assemblies on the same mounting configuration that will be employed in the vehicle. The "hard" or "soft" mount systems result in very different responses of the assembly and the components, and more often than not can make the difference between an item, component, or assembly passing or failing a vibration test. Depending on where the major resonances lie, a case can be made for "hard" or "soft" mounting a system. In most cases it may be said that "soft mounting" or "isolation mounting" a system provides some relief from higher loads when subjected to sinusoidal and/or random vibrations with the above considerations in mind. The initial equipment being proposed for use in NDT will differ substantially from that used in the final configuration and as such any development work or design verification testing performed will not provide realistic data for use towards the design of the final configuration. A glance at the vibration levels indicate that a measure of design from the groundup would be required in order to produce a piece of hardware that could withstand the vibration levels called out in the specifications. Transmissibilities obtained from sine scans can be combined with the required random levels to provide PSD (Power Spectral Density) plots of backup response and on this basis item, component and/or assembly resonance can be placed to provide the required device of isolation.

3.4. Demonstration Hardware Packaging

The objective of the packaging effort was to design a breadboard unit capable of being operated by an astronaut in both a vacuum and 100% oxygen environments as a means to demonstrating feasibility of in-space nondestructive testing. The design criteria can then be outlined as follows:

1. Design of compact package or packages.

2. Design for operation and handling, considering human factors requirements of a space-suited astronaut.

3. Design for maximum utilization of equipment with minimum prior specialized training.

4. Design for built-in versatility and minimum equipment redundancy.

In complying with the program objectives, (i.e. Feasibility and Preliminary Design Study for In-Space NDT) basic off-the-shelf NDT equipment was selected.

A study of thermal test requirements, discussed earlier in this report, (Sections 3.2) indicated specific orbital parameters, (orbits, storage conditions, proximity of other space vehicles to the NDT instrument, etc.) to be premature and beyond the scope of this specific program. In accordance with these findings, only thermal balance (thermal flux) problem areas and flight hardware requirements will be defined in this program. As actual flight hardware design is expected to change significantly, any further effort in this area would be meaningless!

Similar studies of launch vibrational tests were also conducted (reported earlier in section 3.3). Results of these studies indicated that to be of any value, these tests must be performed on actual flight hardware. Minor changes such as mounting bracket design changes would make any vibration test meaningless! As earlier pointed out flight hardware must be designed from the "ground-up". Actual In-Space NDT flight hardware will then be significantly different from the prototype demonstration package of this program. In addition, it is anticipated that clearer definition of the specific launch location of the flight hardware unit in the launch vehicle can be made at the time of actual flight hardware construction. In accordance with these findings, only problem areas, and anticipated flight hardware requirements will be defined in this program.

It should be noted that within the scope of this feasibility study every effort possible will be made to provide sufficient information in defining flight hardware requirements, that actual problem areas in flight hardware construction are minimized.

Careful evaluation has indicated the optimum, most versatile demonstration package to be in two combinable sections. Section 1 being a combination ultrasonics/eddy current instrument and section 2, the radiographic device as shown in Figure 20.

3.4.1 Combination Ultrasonics/Eddy Current Instrument

The decision to combine the Sperry "UCD" ultrasonic unit with the Uresco FC300S eddy current tester became quite apparent once the components and circuitry were studied. The total weight of the FC300S with case, batteries and meter, is approximately 1 3/4 pounds.

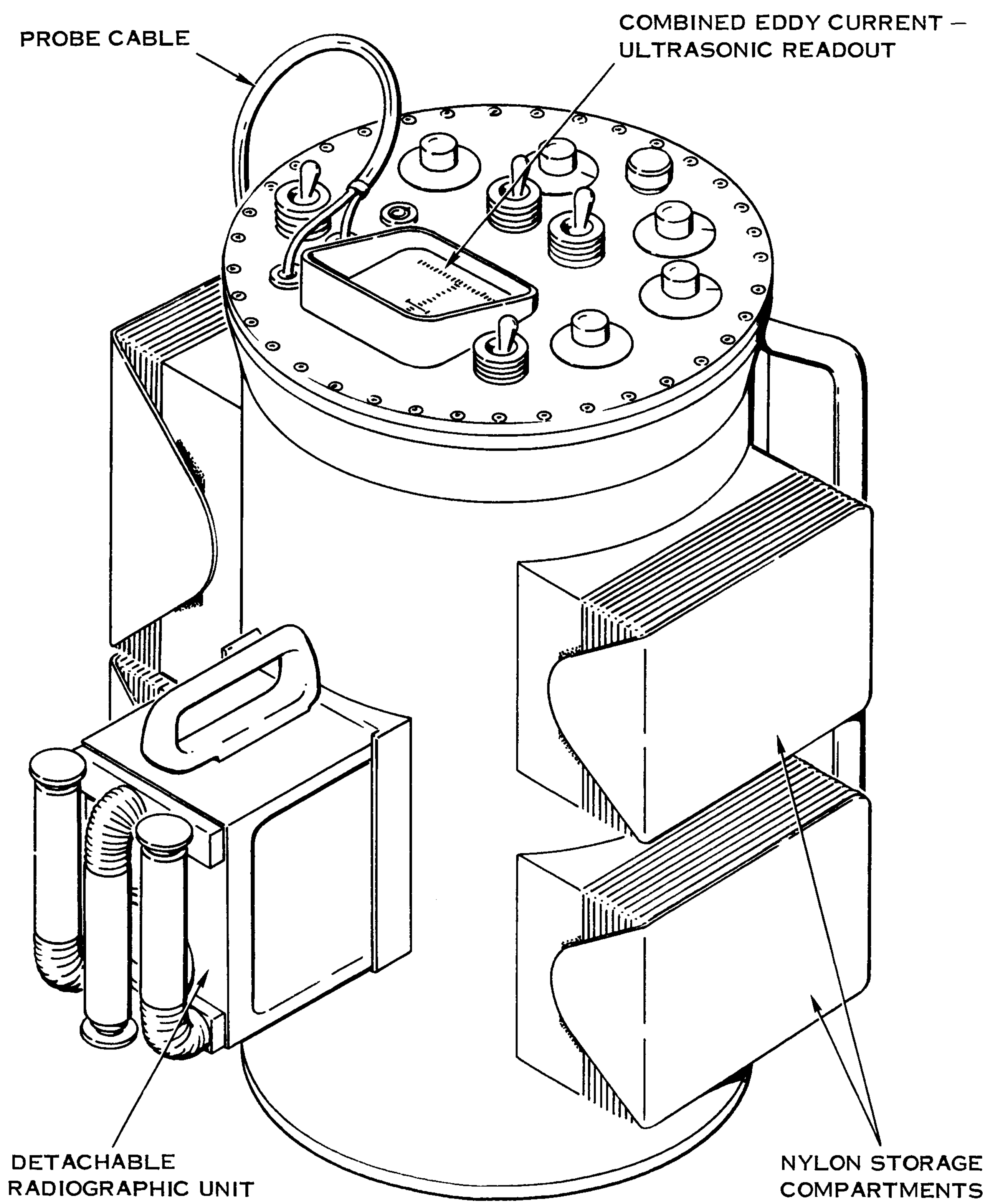


FIGURE 20 PROPOSED IN SPACE NDT UNIT – COMBINED

The unit volume is small and printed circuitry quite compact and modulized. With the CRT being an inherent part of the ultrasonics system, a unified method of displaying both ultrasonics and eddy current readouts is available. The present meter readout of the FC300S becomes redundant and a means of displaying eddy current outputs on the CRT is integrated into the package. By doing this,the differential amplifier section of the eddy current device also becomes redundant and is deleted. With a slight circuit modification, the vertical display deflection system of the "UCD" now serves two functions; that of the eddy current differential amplifier circuit and that of the vertical control of the CRT. Other advantages of the integration of the two units are:

1. Utilization of a single power supply. The batteries of the FC300S are eliminated.

2. Better utilization of external controls by the design of dual function controls. Refer to the package assembly drawing for switch functions. (Figure 21)

3. Weight saving in deletion of excess batteries, instrument case, meter, control knobs, cable connectors, and unnecessary circuit boards and components.

4. Ease of operation. One instrument package and readout as compared to learning two instruments.

5. Elimination of probe cable assemblies. The ability to use one major cable assembly (serving both ultrasonics and eddy current).

6. Smaller overall volume with one integrated package than with two separate units.

7. Deletion of the eddy current meter. Meter movements are always potential problems in flight hardware consideration. In the environments of vacuum and zero -g's, bearing lubrication and pointer drive mechanisms become susceptible to failure. In addition, launch stresses on meter movements are quite severe and damaging.

8. Small number of total parts improving equipment reliability.

Thus, for a slight addition in weight (in ounces) to the ultrasonic unit, and for a negligible increase in current drain (probably less than 30 millamps) the eddy current device is easily integrated into the demonstration package. The overall effect here is that no major cost decision has to be made between the two measurement processes. Both ultrasonic and eddy current testing may be performed by one package for almost the same cost (in dollars, weight, packaging, power, function, etc.) as a single ultrasonic device.

3.4.2 Isotope Radiograph Unit

The radiographic device is being packaged by itself for the following reasons:

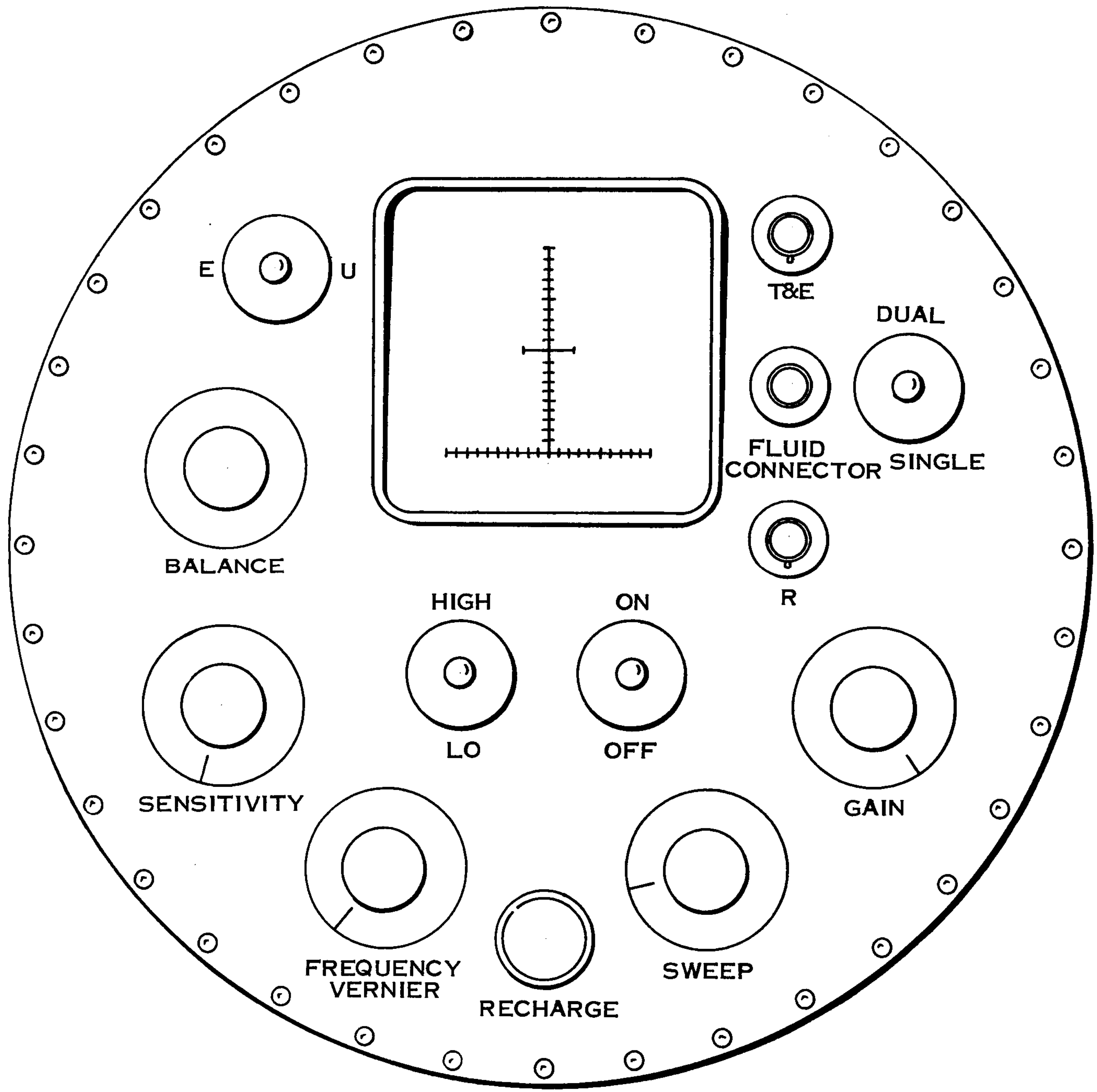

E
U
T&E
DUAL
FLUID
CONNECTOR
SINGLE
BALANCE
R
HIGH
ON
LO
OFF
SENSITIVITY
GAIN
FREQUENCY
VERNIER
SWEEP
RECHARGE

FIGURE 21

1. Depending on type of inspection required, whether fabrication or repair, the astronaut may select the method of testing required without being subjected to the bulk effect of the entire package.

2. Separation from the other instrumentation also provides for testing by two astronauts at different locations, at the same time, if needed. (This would be extremely valuable if urgent accessment of damage must be performed).

3. The radiographic camera, at present, requires no electronic circuitry and integration into the other package would only add considerably to cost, isolation, and weight distribution.

4. In combination with the safety considerations, such as storage and accidential exposure, there would exist a significant psychological barrier to performing the other inspections; the user having knowledge of the radioactive source.

5. The process for changing the radioactive source becomes less complicated in the self contained unit. The ultrasonic/eddy current device need not be subjected to excess handling abuse.

While not designed as an integral part of the ultrasonic/eddy current package, however, provision has been made to enable the radiographic unit to be carried externally, This removable attachment design is illustrated on concept drawing SK64034. (Figure 22) This would allow the astronaut to bring the camera and accessories to a work area, in case both techniques are required.

The radiographic unit drawing No. SK64035 (Figure 22) illustrates the adjustable leg arrangement designed for this package. The legs shown will serve a dual function:

1. They will provide for positioning of the radioactive source to the target material.

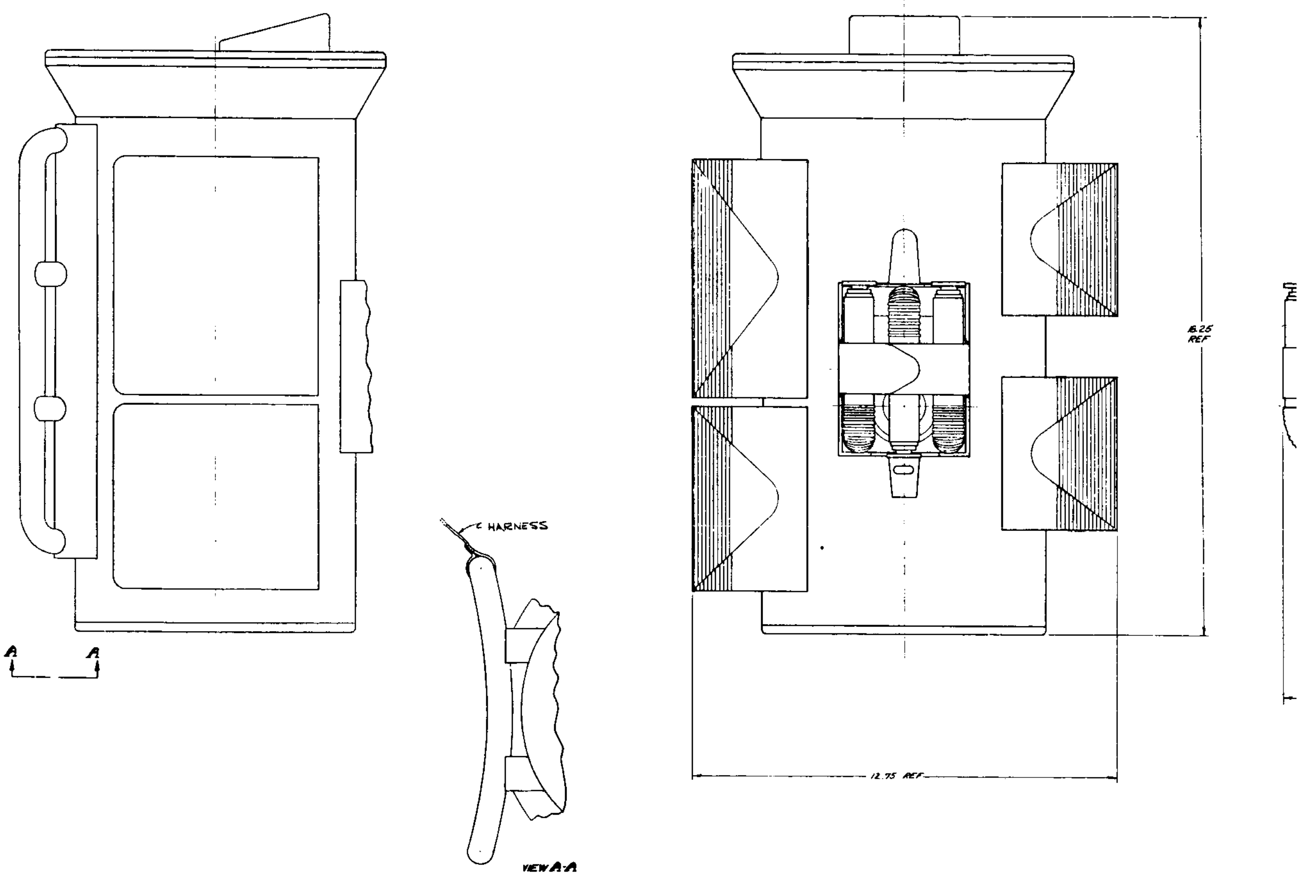
2. They will stabilize and attach the unit to the spacecraft wall. Methods for attachment of the legs to the test article are under study. At this time, adhesive bonded "Velcro" padding appears to be the "best" choice.

3.4.3 Concept Drawings - Human Engineering

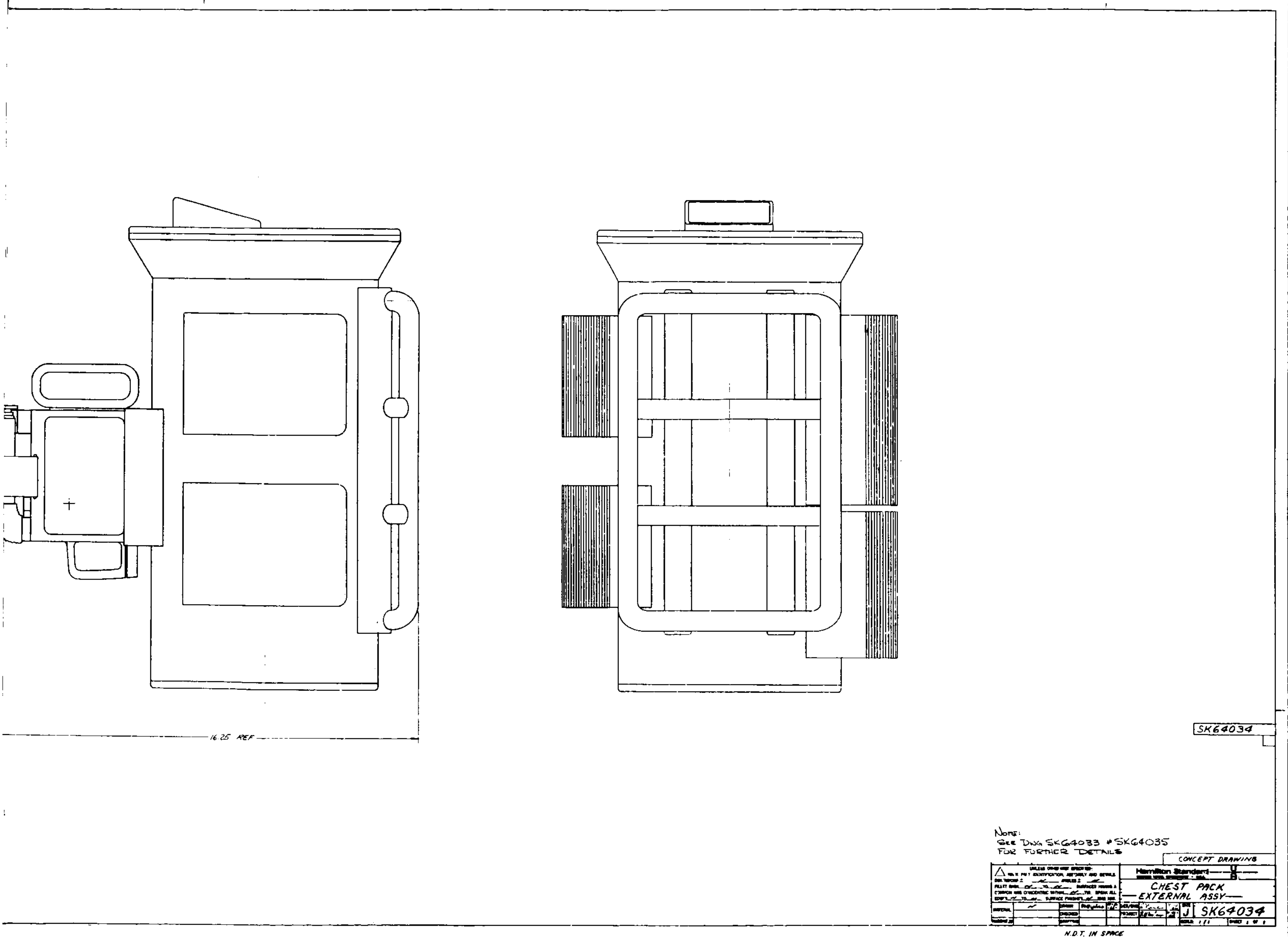
A. Drawing SK64033 (Figure 23)

This drawing illustrates the front panel configuration and internal assembly of the ultrasonic/eddy current chest pack.

The front control panel has been human factors engineered for ease of operation by a space-suited astronaut. The guidelines used here are those developed, through extensive studies, by the Hamilton Standard Human Factors group under the direction of Dr. Vail. Taken into consideration are such factors as:

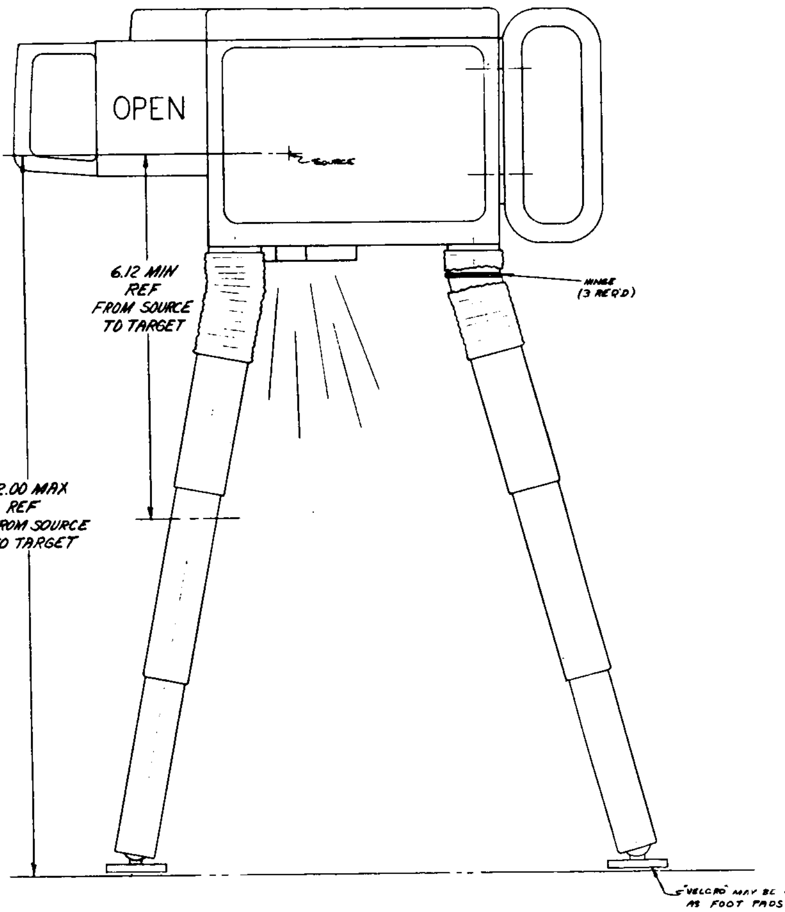


C HARNESS
VIEW A-A
A A
16.25 REF
12.75 REF

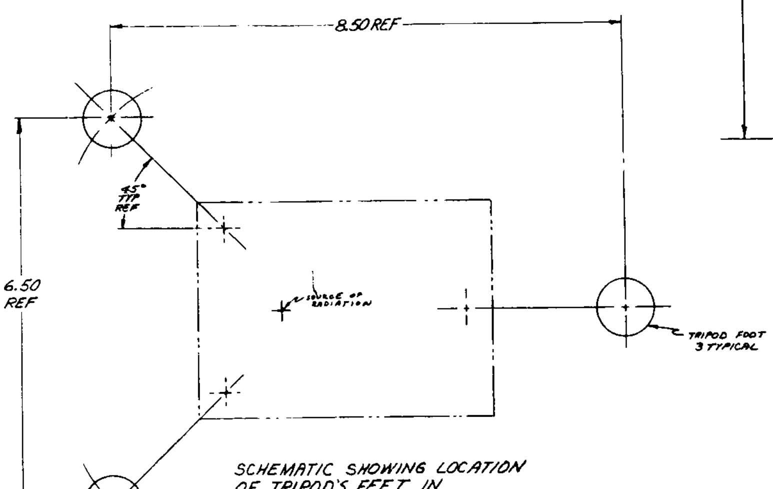


16.25 REF
SK64034
NOTE:
SEE DWG SK64033 & SK64035
FOR FURTHER DETAILS
CONCEPT DRAWING
UNLESS OTHERWISE SPECIFIED
Hamilton Standard
CHEST PACK
EXTERNAL ASSY
J
SK64034
SCALE 1/1
N.D.T. IN SPACE

FIGURE 4-22.

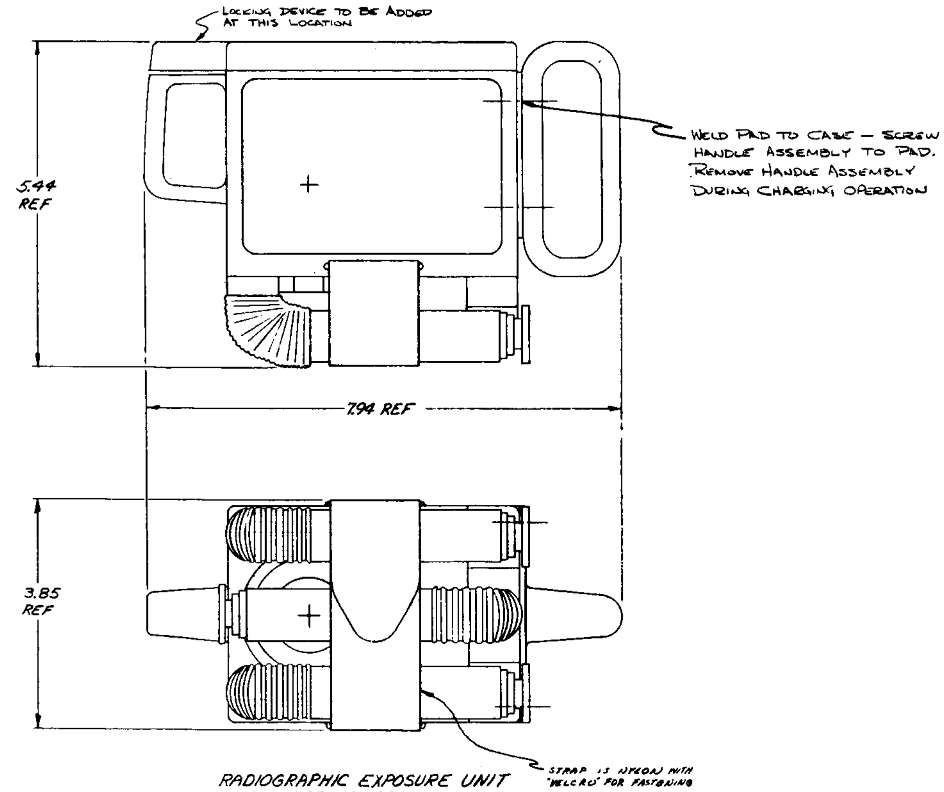


RADIOGRAPHIC EXPOSURE UNIT
IN OPERATING POSITION



SCHEMATIC SHOWING LOCATION
OF TRIPOD'S FEET IN
OPERATING POSITION
(ON RADIOGRAPHIC EXPOSURE UNIT)

FIGURE 4-



RADIOGRAPHIC EXPOSURE UNIT
IN SHIPPING POSITION

SK64035

80-2

CONCEPT DRAWING

UNLESS OTHERWISE SPECIFIED:
MARK PART IDENTIFICATION; ASSEMBLY AND DETAILS.
DIMENSIONS ± ~, ANGLES ± ~
FILLET RADII ~ TO ~ SURFACES HAVING A COMMON AXIS CONCENTRIC WITHIN ~ TIR. BREAK ALL EDGES ~ TO ~ SURFACE FINISHES ~ RMS MAX.

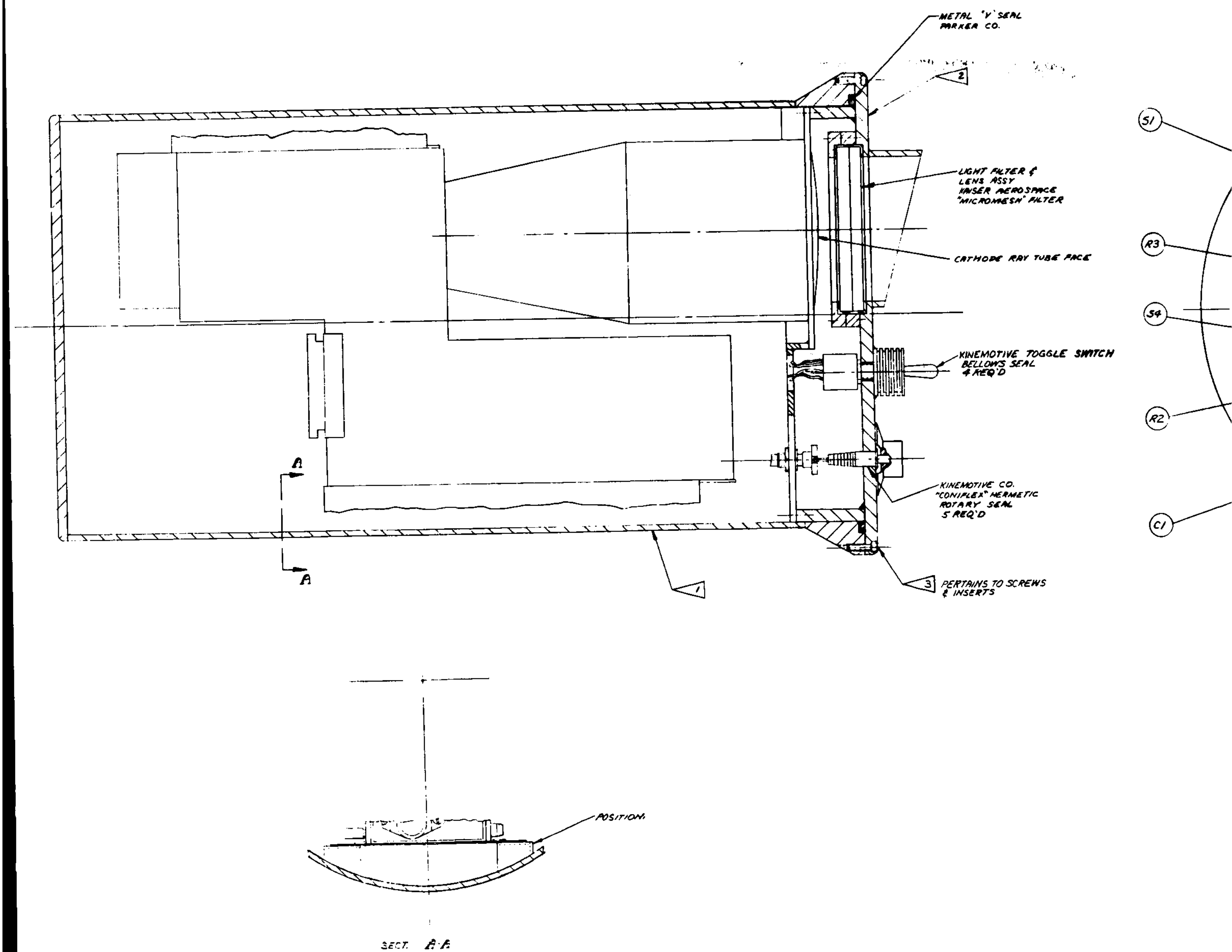
MATERIAL	~	DRAWN	Phillips Lee	10-7-66	DES/ENG
		CHECKED			PROJECT
HARDNESS		DRAFTING			

Hamilton Standard
WINDSOR LOCKS, CONNECTICUT • U.S.A.

RADIOGRAPHIC EXPOSURE UNIT

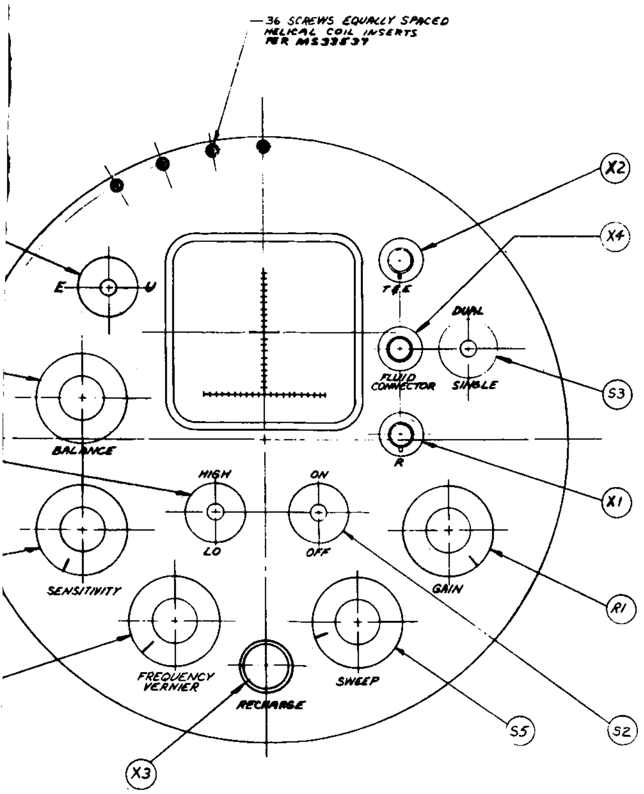
SIZE J SK64035
SCALE: 1/1 SHEET 1 OF 1

N.D.T. IN SPACE


METAL 'Y' SEAL
PARKER CO.
2
LIGHT FILTER & LENS ASSY KAISER AEROSPACE "MICROMESH" FILTER
CATHODE RAY TUBE FACE
KINEMOTIVE TOGGLE SWITCH BELLOWS SEAL 4 REQ'D
KINEMOTIVE CO. "CONIFLEX" HERMETIC ROTARY SEAL 5 REQ'D
3 PERTAINS TO SCREWS & INSERTS
1
A
A
S1
R3
S4
R2
C1
POSITION
SECT. A-A

FIGU

81-1



X4	COUPLANT PASS-THRU (ULTRASONICS)	QUICK DISCONNECT
X3	BATTERY RECHARGE CONNECTOR	3 PIN MALE CHASSIS CONN. HERMETIC SEAL; REMOVABLE CAP
X2	PROBE CONNECTOR (ULTRASONICS & EDDY CURRENT)	COAX CONNECTOR HERMETIC SEAL TYPE
X1	PROBE CONNECTOR (ULTRASONICS)	COAX CONNECTOR HERMETIC SEAL TYPE
S5	SWEEP SELECTOR, (ULTRASONIC UNIT)	4 POSITION, ROTARY SELECTOR SWITCH
S4	FREQUENCY SELECTOR EDDY CURRENT	TOGGLE SWITCH
S3	DUAL-SINGLE (ULTRASONIC PROBE MODE SELECTOR)	TOGGLE SWITCH
S2	ON-OFF SWITCH SYSTEM POWER	TOGGLE SWITCH
S1	EDDY CURRENT/ULTRASONICS SELECTOR	TOGGLE SWITCH
R3	BALANCE CONTROL (EDDY) VERT. POS. CONTROL (ULTRASONIC)	POTENTIOMETER
R2	SENSITIVITY CONTROL (EDDY CURRENT)	POTENTIOMETER
R1	GAIN CONTROL (ULTRASONICS)	POTENTIOMETER
C1	FREQUENCY VERNIER (EDDY CURRENT)	VARIABLE CAPACITOR
DESIG	FUNCTION	TYPE

SK 64033

4. SEE SK64034 & SK64035 FOR FURTHER INFORMATION.

3 SCREWS & INSERTS TO BE STAINLESS STEEL

2 COVER MAT'L - AISI 347

1 CANNISTER MAT'L - AA5052

CONCEPT DRAWING

UNLESS OTHERWISE SPECIFIED:
MARK PART IDENTIFICATION; ASSEMBLY AND DETAILS.
DIMENSIONS ± — ANGLES ± —
FILLET RADII — TO — SURFACES HAVING A COMMON AXIS CONCENTRIC WITHIN — TIR. BREAK ALL EDGES — TO —. SURFACE FINISHES — RMS MAX.

Hamilton Standard

CHEST PACK
— INTERNAL ASSY —

MATERIAL	NOTED	DRAWN	[illegible]		DES/ENG		SIZE J	SK64033
		CHECKED			PROJECT	[illegible]		
HARDNESS		DRAFTING					SCALE: 1/1	SHEET 1 OF 1

IN SPACE N.D.T.

E 4-24

81-2

1. Distance between adjacent edges of controls (not less than "2", where possible).

2. Size of control knobs and handles (not less than 3/4", in diameter).

3. Swing of toggle switch devices (not less than a 30^{0} arc between toggle switch positions).

4. The amount of resistance incorporated in toggle switches (minimum: 10 ounces, maximum: 40 ounces)

5. The amount of resistance incorporated in rotary knobs (torque minimum: value determined by jarring, vibrations to be met, torque maximum: for fingertip operation = 4 1/2 inch-ounces).

6. Logical layout of controls to facilitate the sequential or simultaneous operation and economy of panel space.

The choice of the chest pack concept was prompted by the results of the recent Gemini 11 extravehicular activity. The requirement to minimize the work load of the astronaut was pointed out by the astronaut's rapid over exertion on this trip. The chest mounted instrument package eliminates the need to hand carry the equipment and aids significantly in the ease of operation of the ultrasonics/eddy current equipment.

In the operation of the unit the astronaut will have all the materials needed; self-contained and available from the chest pack. (See concept Figure 25) The probe cable assembly and wrist band (see artist conception Figure 26) will be located in a corrugated nylon pouch (the larger of 4 ounces shown in SK64034), located on the right hand side of the cylindrical container. The smaller pouch on the righthand side will contain the semi-glove mounted fingertip probes. The other two pouches, located on the left-hand side, will be used for film packs and defect marking materials.

The two off-the-shelf instruments selected have a total of 19 controls, five on the eddy current unit and 14 on the ultrasonic unit.

When this same equipment is human engineered for space applications, the total number of controls increases to 22. Three more are required on the ultrasonic unit. Integration of the ultrasonic unit and eddy current unit into a single instrument and human engineering reduces the controls to only 9. Five controls are still required for eddy current but the ultrasonic portion needs only 4. Preliminary studies for flight hardware indicate a possible further reduction to six controls.

The CRT display, control knobs, and probe connections are all located on the top panel affording full visual contact for the astronaut. A special lens assembly light filter will be installed in front of the CRT face to eliminate washout in direct sunlight. This Kaiser Aerospace "micro mesh" filter has the characteristic of preventing 95% of unwanted

FIG. 25 IN SPACE INSPECTION

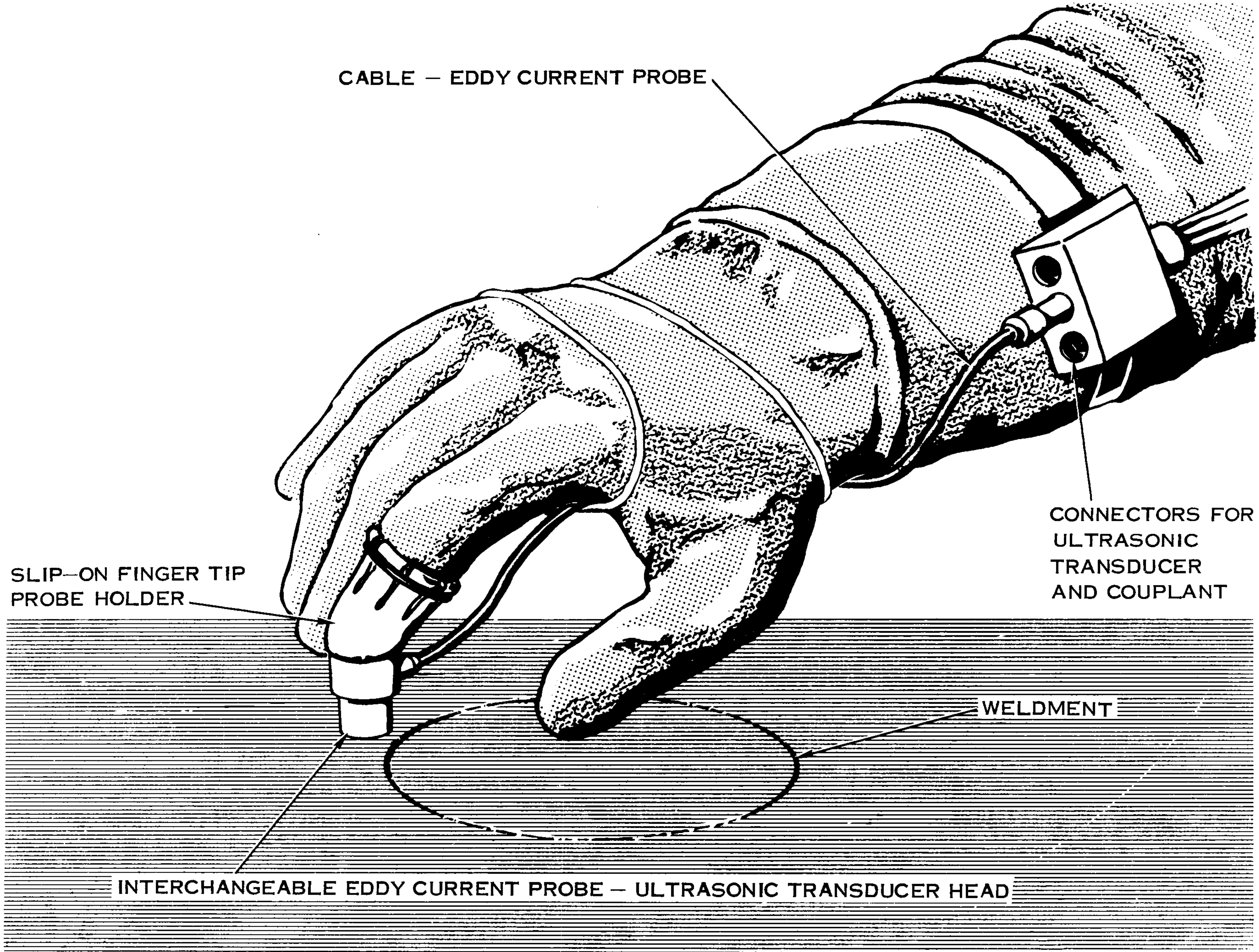


FIGURE 26 PROPOSED PROBE CONCEPT

incident light from striking the display surface. In addition, the lens assembly has an anti-reflecting quarter-wave coating to minimize reflections from the front surface of the glass. This coating reduces specular reflection to less than 0.5% of the intensity of the incident light. The viewing angle of the "micro mesh" filter is a 30^{o} cone or $\pm$ 15^{o} from normal.

The demonstration unit has been designed to function in a vacuum or 100% oxygen environment. To accomplish this all the electronic components, switches, battery, and CRT are located in a hermetrically sealed container. In addition, the ultrasonic fluid couplant will be stored in compartments within this container. The container has been designed as a pressurized can (14.7 to 17 psia). The can will be filled with inert nitrogen gas. A small pressure switch and indicating light will be provided to detect loss of container pressure. The inert/gas/atmosphere will serve to readily conduct heat from the electronic components to the vessel wall and to greatly decrease internal spark potentials. The cylindrical shape of the container was chosen to:

1. provide greater strength under pressure

2. eliminate potential sharp edges

3. facilitate use of sealing devices (readily available in circular configuration)

4. to provide as light a weight package as possible while still conforming to the chassis configuration of the off-the-shelf Sperry "UCD".

5. facilitate fabrication time (availability of cylindrical stock)

Vacuum sealing of the instrument package will be provided using approved techniques. The cover plate (control panel face) will be removable for repeated access to the breadboard package. A V-shaped special vacuum seal ring will be used as the cover plate seal. Toggle switch actuation will be vacuum sealed using a welded bellows seal assembly. The rotary actuated potentiometers will be protected from vacuum by using special bellows rotary hermetic seals. The sealing techniques are illustrated on drawing SK64033. All connectors shown will be hermetic types having glass to metal sealed header pass-throughs.

The total estimated weight of the demonstration ultrasonics/eddy current package (not including the radiographic unit) is 26.0 pounds.

The estimated volume of this unit is 0.9 cubic feet.

B. <u>Drawing SK64034</u>

This drawing shows the external configuration of the demonstration package. The Viso radiographic is shown, conceptually, attached to the ultrasonic/eddy current chest pack. The actual configuration of the demonstration unit will have such a mounting provision for the radiographic unit.

3.5. Probe Design

Two major concepts in probe design were considered for astronaut use for in-space nondestructuve testing. The first concept was that of a multiple probe, hand held unit, which could be utilized for the inspection of such fabricating methods as electron beam welding, adhesive bonding, brazing and TIG welding. Figure 27 shows a multiple headed unit containing an eddy current and ultrasonic probe plus the couplant contained in a squeezeable tube. This initial concept appeared desirable due to the fact that all probes were centralized and that the probes would not have to be changed while in a zero-gravity condition. This condition could present problem areas in that transducers or probes could potentially be "lost in-space". Closer evaluation of this method revealed areas which noteably affect astronaut fatigue and thereby jeopardize success of the mission. Examination of the pressurized glove revealed that the glove tends to assume a position of maximum volume; i.e., a position where all the digits are fully extended. Under these conditions grasping a control or handle in a pressurize glove would require expenditure of energy. The continuous grasping of a handle, as would be required in the performance of various inspection application, would require a continuous expediture of energy and thereby contribute to overall astronaut fatigue. Also considered in the multiple probe concept was the problem of maintaining intimate contact with the surface of the material to be inspected. It was noted that, while grasping the handle of the multiple probe unit, any twisting of the hand, wrist, forearm or upper arm would break the contact required for inspection. Motions of the hand or arm in any direction could also break the contact area.

In considering the forward and backward plus the sideways movement required during angle beam ultrasonic inspection it was decided that a finger tip mounted probe would be more applicable to this program. It was noted that the wrist and hand enclosed in a pressurized glove has a maximum pronation-supination of $\pm$ 50° to $\pm$ 70° and that the wrist has a maximum flexion of 40° to 65°. The ability of the wrist to bend and twist provides a compensating device with which to overcome body movements caused by zero-gravity conditions. Also with the transducer or eddy-current probe mounted on the finger-tip the tactile feedback or sense of touch is increased to the point where maintaining contact and providing motion to the transducer is greatly enhanced. The relationship between body mass and probe mass is so great that mobility of the finger tip must be considered as a prime design factor. The nature of contact testing demands this sense of touch, motion and positioning if defects are to be located and identified as to type and size. Another advantage to the fingertip-concept is that the hand with the probe can be utilized to set the controls on the unit and also help when the astronaut is positioning himself prior to and during the inspection process.

Interchangeability of probes will be accomplished by a connector, similar to a B.N.C. This connector will serve only as a positive mechanical connection between the probe and the fingertip holder. Tethers will be provided to guard against loss of transducers.

Weight of the fingertip design has been estimated at approximately 1/3 the weight of the multiple-probe unit. This is an important consideration due to the nature of the integrated package design concept. Overall unit mass must be considered as a parameter contributing to astronaut fatigue.

The liquids which appear most promising as an ultrasonic couplant belong to the silicone fluid family. They exhibit sound transmission characteristics similar to SAE 30 oil. Many of the silicones have been exposed to radiation doses up to 10^6 rads, at pressures down to 10^{-5} mm, and exhibited no penetration change or deterioration.

In considering couplants for ultrasonic applications, one potential problem area stands out. This problem is contamination of the astronauts hands, suit, and controls. During the process of inspection it is possible, and probable, that the astronaut will come in contact with the couplant which is essentially a lubricant. Should the lubricant be spread to the control panel knobs and the astronauts environmental system controls, the ability to manipulate the knobs with slippery pressurized gloves would be difficult. A possible solution to this problem would be to have the astronaut wear thin, lightweight, expendable gloves only during the inspection process. These gloves could be stripped off and thrown away after inspection or during an emergency situation. This would be most feasible for vehicular inspection under pressurized situation. Initial concept of selecting a fluid for inspection outside the capsule considered use of a liquid which would vaporize at a moderate rate and thereby present no contamination problems. Considering this same vaporizing fluid for inspection inside the capsule and in 100% oxygen atmosphere would indicate the possibility of contamination of the total cabin, mechanical controls, and electrical instrumentation.

It is therefore apparent that a different couplant for inside and outside use may be required.

Preliminary concepts of applying the couplant have been (1) mounting a squeeze type applicator on the hand held multiple probe unit and (2) utilizing a fluid membrane over the transducer with a small orifice in the membrane to facilitate slow seepage on to the work piece. The transducer will be so designed as to utilize the couplant as a fluid delay line and also allow the use of focused transducers. The proposed system would feed the fluid from a reservoir in the instrument package to the fluid delay portion of the probe which in turn would flow through the membrane and act as a couplant. Placement of the couplant fluid reservoir within the instrument package will provide a heat sink for maintaining reasonable temperatures on certain electrical components plus the heated couplant will tend to balance out thermal variations at the probe, partially during maximum shadow time orbits.

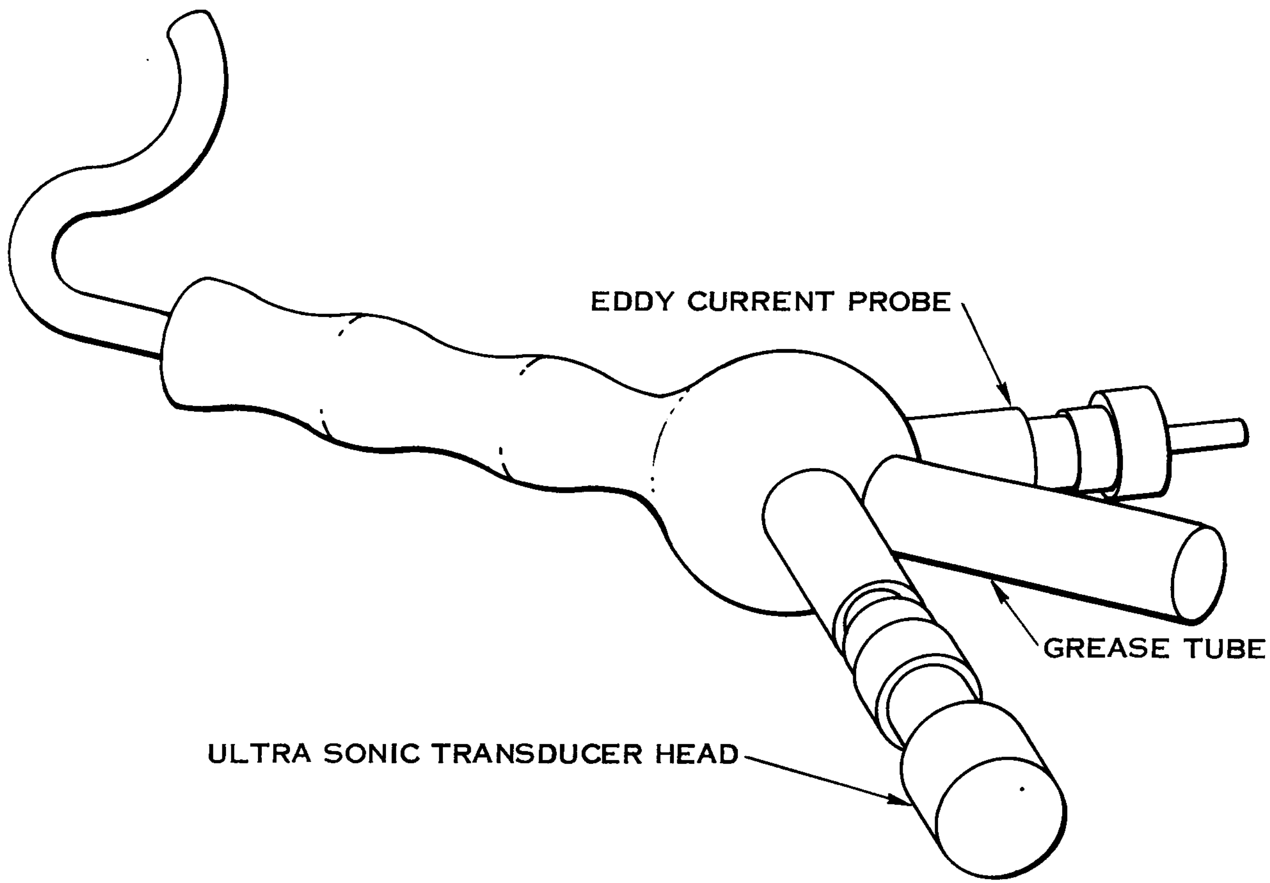


FIGURE 27. MULTIPLE HEAD PROBE CONCEPT

4.0 TEST PLAN - PHASE II

4.1 Procurement of Off-the Shelf Instruments

To accomplish the above proposed design concept modifications for a breadboard demonstration instrument the following basic off-the-shelf equipment will be procured. Sperry portable ultrasonic instrument, model UCD; Picker portable radio isotope camera, Viso model 601, and Uresco portable eddy current unit, model FC300S. Note, that only a few of the electrical circuits from the Uresco eddy current unit will be utilized, however procurement is required in order that preliminary defect and environmental studies can be made.

Modification to the equipment will be accomplished in accordance with final designs which will be completed during the first month following Phase I concept approval. Testing and evaluation will be indicated as follows.

4.2 Fabrication of Standard Defects

A necessary part of any nondestructive test plan is fabrication of standard defects. The earlier review of proposed in-space fabrication has indicated that most of the representative defect specimens required by the program are directly applicable. However, to get maximum integration with the currently planned Apollo applications program and projected early in-space nondestructive test requirements,some changes are recommended. The following lists the test specimens recommended and will be fabricated within the limits of feasibility for use in subsequent equipment and processes evaluations:

a. Surface cracks - a minimum of 0.125" length X 0.010" depth in electron beam butt-welded 2014 aluminum alloy plates of 0.50" - thickness. The surface finish to be a maximum of 250 rms.

   Note: Due to lack of absolute control and precise measurement of natural cracks in a weld without distructive evaluation, an artificial eloxed groove of the above length and depth may be substituted as a standard.

b. Surface porosity -- porosities of 2% minimum size, based on material thickness, in electron beam butt - welded 2014 aluminum plates of 0.50" thickness.

c. Lack of penetration -- a minimum 20% lack of penetration in electron beam butt-weld 2014 aluminum alloy plates .250 and .50" thickness. The surface finish to be a maximum of 250 rms.

d. Lack of penetration - - a minimum 0.125" in length for through electron beam welded lap joint,simulating 0.020" thickness patch on 0.020" thick face sheet of honeycomb core.

e. Adhesive bonded aluminum honeycomb panel -- one inch diameter debond area and a one inch diameter crushed core area. Face sheets are 0.020" thickness and core is 1/4" - aluminum alloy .004P, 1/2" thick.

f. Lack of bond in brazed tubing -- a minimum of 20 percent of braze area, the defects being oriented circumferentially and longitudinally within the tube to coupling brazed joint between a 1/4" AISI 347 or AISI 1321 stainless tube of 0.028" wall thickness and a 3/4" long X 0.459" O.D. coupling.

Note: Representative actual tube to coupling joints of a similar material may be supplied by NASA for furnace brazing and establishment of above defects or as supplied with known built-in defects. These will be considered as a satisfactory substitution.

g. Temper variations -- two 2014 aluminum alloy plates of 0.125 thickness X 4" wide X 4" long shall be prepared. One shall be of T6 temper and one shall be in the "O" condition.

4.3 Concept and Design Evaluation Testing

This preliminary study for application of nondestructive testing to space has indicated the areas which must be evaluated prior to design of flight hardware as well as those areas where evaluation can only be conducted on totally designed flight hardware. The following concept and design evaluation testing plan covers only the former areas. These areas are outlined below:

Ultrasonics:

(1) Establish resolution of standard defects by demonstration instrument.

(2) Select and evaluate couplant for compatibility with hard vacuum and O_2 environments.

(3) Evaluate probe materials compatibility with hard vacuum and O_2 environments.

Eddy Current:

(1) Establish resolution of standard defects by demonstration instrument.

(2) Evaluate performance in simulated environments using Uresco FC300S as basis of comparison.

(3) Evaluate probe materials compatibility with environments using Uresco FC300S as basis of comparison.

Ultrasonic - Eddy Current System:

(1) Evaluate both methods of inspection as a single demonstration packaged instrument in hard vacuum.

(2) Evaluate both methods of inspection of demonstration packaged instrument in 100 per cent O_2 atmosphere.

(3) Conduct survey of EMI during both methods of operation of demonstration packaged instrument under room ambient conditions.

Radiography:

(1) Establish resolution of standard defect using Viso Model 601 Camera and ytterbium 169 source.

(2) Evaluate polaroid film compatibility and reliability in hard vacuum.

(3) Develop technique chart for use of Viso camera and ytterbium 169 source.

(4) Establish safety procedures considering astronaut use of Viso camera and ytterbium 169 source.

Radioisotope Package Operation:

(1) Conduct actual exposure of standard defect in hard vacuum using modified Viso 601 camera and ytterbium 169 source.

Because the proposed concept and design evaluation testing actually constitutes applied development, even though standard terrestrial equipment and processes are being used, all specific requirements defined for this section are considered as guidelines. That is, within the capabilities of existing Hamilton Standard facilities and equipment, tests will be conducted as close to the levels defined as is reasonable and possible.

4.3.1 Ultrasonics

On receipt of the procured off-the-shelf Sperry UCD unit, calibration checks will be made against standard test blocks in order to define the original sensitivity and resolution capability. This will serve as the basis for evaluating the effect of proposed modification and simulated space type environment on performance.

Couplant tests shall consist of selecting a liquid which has good sonic properties, is inert in an oxygen atmosphere and will survive in a vacuum in a temperature range of -100°F to +160°F. Compatibility tests shall simulate those conditions of temperature and pressure the liquid shall see during flight storage conditions. Duration of the test shall be a maximum of 168 hrs. Operating tests shall consist of evaluation of the sonic properties of the liquid while in a thermal vacuum of at least 5 X 10^{-5}mm Hg and at temperature extremes of -20°F to +140°F. Duration of operating tests shall be four (4) hours maximum.

The ultrasonic transducers shall be exposed to temperatures ranging from -100°F to +160°F. This shall be a maximum of 168 hours survival, non-operating test. Operating tests shall be conducted at a thermal vacuum of 5 X 10^{-5}mm Hg in a temperature range of -20°F to +140°F for a maximum of four (4) hours. This test shall consist of fixturing transducers to a test block with the couplant selected and observing any changes in characteristics of the received signal.

Following the above tests and utilizing the subject transducers, reinspect the test standards at terrestrial condition using established techniques. This reinspection shall indicate either gains or losses in sensivity due to the environments.

4.3.2 Eddy Current

Since only specific circuits of the Uresco eddy current tester will be used in the modified ultrasonic-eddy current demonstration equipment, calibration tests will of necessity ultimately have to be run on the modified equipment. However, in order to study the environmental effects of hard vacuum and temperature on performance; a standard of reference is required. The portable Uresco eddy current tester, IACS test blocks and standard defects will form this basis. On procurement of the instrument, calibration tests will be conducted to establish sensitivity and resolution capability.

Although a fluid couplant is not required in eddy current testing, the effect of hard vacuum, 100 per cent O_2 atmosphere and extreme temperatures are unknown. Test conditions for the eddy current probe will include a thermal vacuum at 10^{-5} torr over the temperature range of -20 to +140°F for times up to 4 hrs. maximum considering AA2014-T6 AA2014-0, and a Std. IACS block. Compatibility and influence on performance of the eddy current probe in a 100 per cent O_2 atmosphere will be established during the ultrasonic-eddy current package evaluation only. This test is discussed below. The thermal soak evaluation simulating storage conditions will be run for 168 hrs (1 week) at -100°F minimum and 160°F maximum in air at normal atmosphere conditions. Re-examination of the standard IACS test blocks and defects will establish the degree of gain or loss in sensitivity.

4.3.3 Ultrasonic-Eddy Current Package Evaluation

Evaluation testing of the preliminary concept ultrasonic-eddy current demonstration instrument will include leakage testing, operation in vacuum and 100 per cent O_2 atmosphere plus a E.M.I. survey. Prior to vacuum testing the completely assembled instrument will be pressurized with helium. By mass spectrometer techniques, a reasonable leak free rate will be established. Since a go-no-go pressure indicator light will be included in the internal chamber of the ultrasonic-eddy current instrument, this will then be the only means by which further leakaged will be evidenced.

The vacuum tests will be conducted at 10^{-6} to 10^{-7} torr under room ambient temperature conditions. With the ultrasonic system in operation and the transducer coupled to a standard defect by a fixture, the vacuum chamber will be pump down to the range of 10^{-6} to 10^{-7} torr. Changes in the observed osciloscope indication will be one measure of the effect of hard vacuum on sensitivity and resolution capability. With the eddy current system in operation and the probe fixtured to a standard. the vacuum chamber will be pumped down to the range of 10^{-6} to 10^{-7} torr. Changes in the observed amplitudes of the osciloscope indication will be one measure of the effect of hard vacuum on the sensitivity and resolution capability. Following these tests the instrument will be check out on standards at room ambient conditions and visually examined for evidence of change or damage.

The oxygen compatibility tests will be run similar to the vacuum tests, with the exception that the final atmosphere will be 100 per cent O_2 at room temperature and a pressure of 3-5 psia. Evaluation of change and damage will also be based on rechecking of standards at room ambient conditions.

The electromagnetic interference measurements will be of a survey type. Radiated interference and susceptibility will be investigated in screen room testing at a commercial testing facility. Low and high frequency levels of both the ultrasonic and eddy current methods will be considered. Specification MIL-I-6181D will be used as the guide-line for the permissible levels of interference and susceptability. Tests for conducted interference and susceptibility will not be performed as they are not applicable to this unit as it is self-powered from the integral battery.

4.3.4 Radiography

Before effective and safe utilization of isotope radiography can be achieved in space certain basic evaluations must be undertaken. The following tests have been planned to develop this information.

Polaroid Film Sensitivity

Three types of Polaroid film are being considered P/N-55 (ASA 50 speed), P57 (ASA 3000 speed) and type TLX.

The first series of tests will be directed toward selection of the Polaroid film having the greatest defect sensitivity and film exposure latitude. The evaluation will consist of comparing each of the films, which have been exposed to both conventional x-ray and the ytterbium 169 radiation, with Kodak type M film which has been exposed in a similar manner but processed under standarized conditions.

Sensitivity Tests

Variables: Film: Polaroid P/N 55, P57 and TLX; Kodak M
Radiation Source: 25 Curce Ytterbium 169, conventional x-ray.
Radiation Exposure Time: Three levels each thickness.
Material Thickness: 1/8", 3/8" and 1/2" AA2014 aluminum.

Constants: Environment: room ambient temperature
Radiation Parameters: Focal spot size, fixed focal distance, KV-ma for conventional x-ray

Standard of Comparison (1) Kodak M developed 5 minutes at 68°F using General Electric Solutions.

(2) AA2014 Aluminum step block with 2% actual thickness penetrameters at 1/8" 3/8" and 1/2".

Environment Compatibility

The second series of tests will be addressed to compatibility in the terrestial space and vehicular environments. Preliminary analysis indicates that processing of Polaroid film in a vacuum without supplimentary thermal and atmospheric control is highly unlikely. The temperature control necessary for obtaining a reproducible image plus direct sublimation and in turn icing of the moist gelatin processing chemicals due to the high vacuum are the two major factors. To establish film storage requirements the following test will be conducted:

Space Storage Evaluation

Initially off-the-shelf film will be evaluated in order to establish the order of magnitude of the space storage problem. To minimize cost this will be accomplished using a partial Latin Square statistical approach. Following storage tests one square consisting of nine randomly mixed variables will be evaluated by exposure to convential x-rays under standardized conditions and to a specific defect level established in the sensitivity tests. Conventional x-rays will be used since the ytterbium 169 source strength would be changing during the length of the storage testing thus requiring compensation in the time of exposure. The second square, however, will be exposed to the ytterbium 169 source to assure that preferential sublimation of some of the photo emulsion chemicals, which are sensitive to the radiation wave length of the isotope, has not occurred. The pertinent variables and constants are listed below.

Variables: Atmosphere: 10^{-2}, 10^{-5}- 10^{-7} torr
Film: Polaroid P/N 55, P57 and TLX

Constants: Temperature: ambient room
Conventional x-ray at standard defect at set KV-ma, F.F.D. time and processing following removal from storage tests.
Isotope: ytterbium 169
Standard of Comparison: Results on limit of sensitivity from previous tests.

Should leakage occur in the sealed metal foil developer pads or significant degradation of the film sensitivity occur, then evaluation of hermetically sealed film packs will have to be explored. The approach currently being considered is to provide a supplementary envelope which would be vacuum sealed at a pressure not detrimental to the chemicals yet would require structural support for only a low differential pressure when exposure to the total vacuum of space. A low gas permeable organic film is the preferred material since in the relatively soft radiation range of ytterbium 169 addition of higher density materials will impose further restrictions on sensitivity. Once an envelope design and material is selected re-run of the above Latin Square storage tests would be accomplished.

Should no leakage or significant degradation of the film sensitivity occur the film reliability must be established. Considering either the off-the-shelf film or the modified version, whichever is proven necessary, storage testing and certain radiation exposure evaluation must be accomplished on a large enough sample size of film to establish a significant level of confidence. To accomplish this, thirty films from at least three lots of the type selected will be stored at 10^{-7} torr for a period of 168 hrs (1 week) at which time they will be exposed to conventional x-ray under standardized conditions. Thirty films randomly selected from the same lots will also be stored at room ambient conditions for the an equal length of time and then simularly exposed to the conventional x-ray radiation. Following development in a conventional Polaroid pack, evaluation will be based on (1) evidence of leakage of the chemical pad prior to development (2) loss of resolution of the standard defect image and (3) the uniformity of film development. Successful completion of all tests would define the reliability required for 99 per cent confidence that 90 per cent of the exposures would be successful.

Parallel evaluation of the films latitude to storage under the extreme temperatures anticipated in the terrestial space environment will be required. Of particular concern is the combined effect of the elevated temperature and vacuum on the sublimation and/or conversion of the processing chemicals. The low temperature does not appear to be a problem unless loss of adhesion of the chemical film was encountered. To determine the severity of the problem, three films per test condition will be subjected to temperatures of -100 and +60 degrees F. Note only the elevated temperature test will be conducted in vacuum. Following exposure to standard x-ray conditions and subsequent development, analysis as to the effect on performance will be made.

A review and analysis of all of the above data should indicate (1) the best Polaroid film to use (2) the relative sensitivity and reliability of this film for use in vacuum and (3) the extent of additional development required to fully utilize this film in flight tests and what may be required if development of the film outside the vehicle is ultimately deemed necessary.

Technique Operation Chart

In order for the astronaut to effectively use the Viso camera and ytterbium 169 source for routine inspection he must have simple operational instructions. This must consider: working distance from object being radiographed, strength of source, age of source, type film, material, material thickness and time of exposure. Planned testing will be directed toward establishing a technique chart covering the range of materials and thicknesses represented by the standard defects being fabricated for this program. This data will be analyzed considering different methods or means of presenting the parameters which will provide ease of operation.

Safety Evaluation

Because of the low strength level, 25 curies and nearly monocromatic wave length of the ytterbium 169, source handling and safety precautions become minimal. The modified breadboard camera Picker, Viso 601, will effectively be rated for 100 curies. As a result checking in this respect will be limited to a routine leakage survey of the camera with the 25 curie source in the safe position emission. Preliminary evaluation of working radiation levels will be accomplished under normal room ambient conditions. While the camera is emitting radiation normal to the surface of 12" X 12" X 1/4" aluminum and 12" X 12" X 1/8" steel plates respectively, a survey of back scatter radiation measurements will be made at 10 degree increments though one quadrant and at the 0, 30 and 60 degree positions in the corresponding quadrant. Intensities at a minimum of two radii, representing boundary limits of where the astronaut would be while activating the camera will be measured. Graphic presentation of this data will be as shown in Figure 28.

Radio Isotope Package Operation

The preliminary concept demonstration radiographic unit, a modified Viso 601 camera with 25 curie ytterbium source, will be operational tested in a vacuum of 10^{-6} to 10^{-7} torr at room temperatures. The equipment will be fixtured to permit activation of the radioisotope source. The source to standard test specimen distance will be fixed. The film will be the best polaroid type selected from previous work. Development will be accomplished at room ambient conditions. Evaluation of results will be based on comparison of sensitivity and resolution capability from previous tests.

4.3.5 Demonstration Test - NASA-MSC

The prototype in-space nondestructive testing unit will be demonstrated at NASA-MSC at the completion of this program. The demonstration will be conducted in the eight-foot

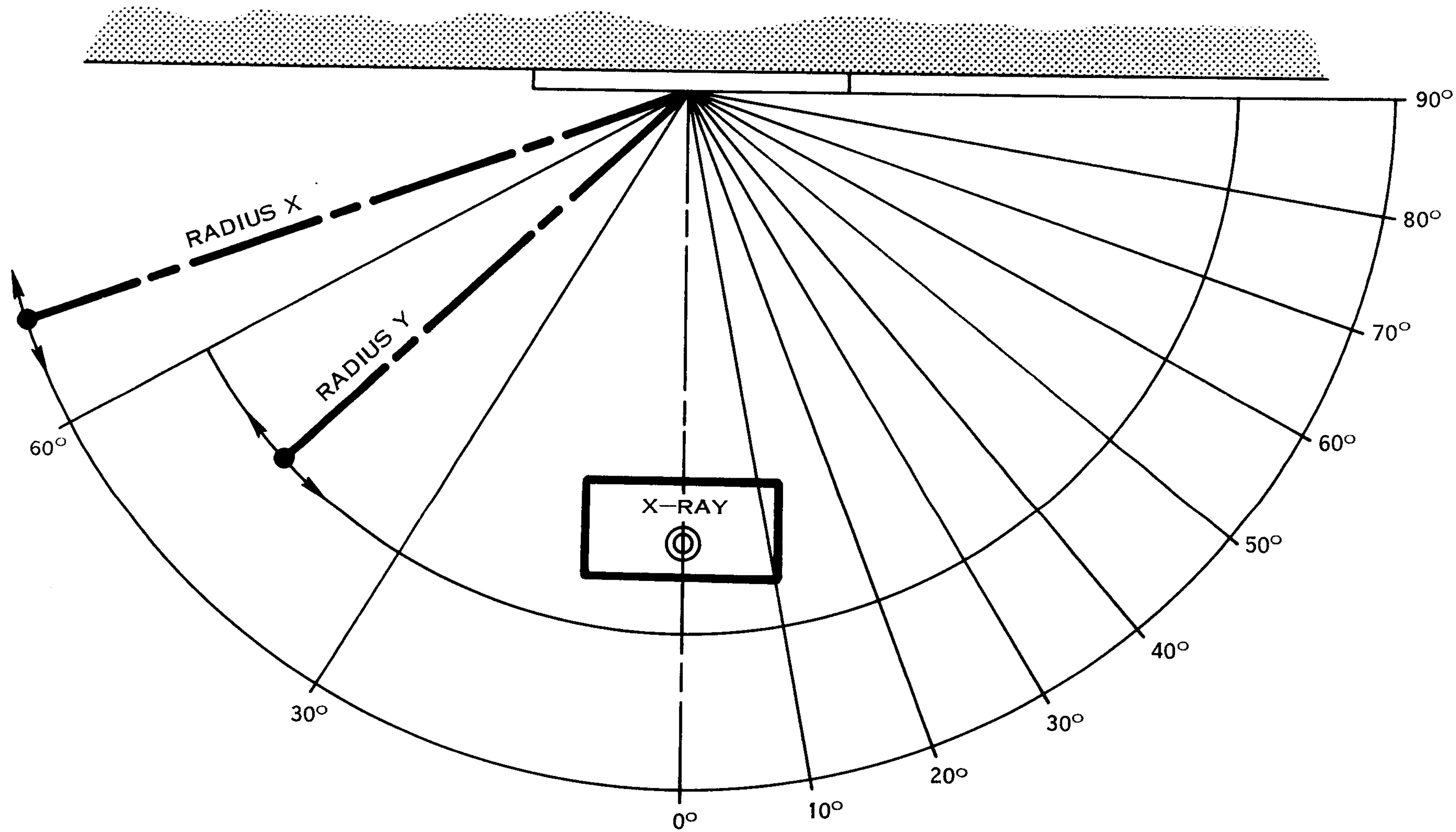


FIGURE 28 BACK SCATTER RADIATION SURVEY

diameter altitude chamber, furnished without charge by NASA under the following conditions:

(1) Pressure level of 1.5×10^{-2} torr

(2) Ambient temperature of 68 to $70^{\circ}F$

(3) No thermal vacuum or solar radiation simulation

The test subject will be supplied by Hamilton Standard and be qualified to operate the prototype NDT package. A test plan defining specific defects and the appropriate inspection technique will be prepared by Hamilton Standard and submitted for NASA-MSC further breakdown. This will be supplied at least one month prior to actual demonstration. For the test demonstration, NASA will provide the following equipment and/or services:

(1) Pressure profile of chamber internal pressure

(2) Four thermocouples to monitor equipment or ambient temperature if requested

(3) Video tape recording of demonstration

(4) Space suit for Hamilton Standard Division test subject

(5) Biological monitoring of test subject, including respiration, O_2, CO_2 and EKG.

(6) Breakdown of test plan to movement level

(7) Test subject debriefing and short test profile report.

Bibliography

1. R. D. Engquist, D. B. Nord, Study of Space Environment Fabrication and Repair Techniques, May 1966.

2. F. R. Schollhammer, G. A. Lind, Final Report on Electron Beam Welding in Variable Environments, AFML-TR-65-172. June 1965.

3. R. D. Hibben, "USAF Studying Space Tool Development", Aviation Week and Space Technology, September 13, 1965.

4. L. G. Foxwell, F. R. Schollhammer, J. E. Smith, Electron Beam Welding in Variable Environments, 1R-8-243(I).

5. C. G. Goetzel, J. B. Rittenhouse, J. B. Singletary, Space Materials Handbook - Second Edition, ML-TDR-64-40A.

6. J. A. Chamberlin, "Manned Spacecraft Design - The Problems Solved", Astronautics and Aeronautics, March 1966.

7. Hua Lin, "Structures and Materials in the Long Duration Manned Spacecraft," Astronautics and Aeronautics, March 1966.

8. D. Bogdanoff, M. T. Braun, M. Kushner, Composite Structures for Orbiting Space Stations, SAE Preprint No. 910D.

9. P. E. Purser, M.A. Faget, N. F. Smith, Manned Spacecraft: Engineering Design and Operation, 1964.

10. G. Peters, J. Whitaker, J. Hanny, Extravehicular Space Adhesive System AFAPL-TR-66-59.

11. R. A. Long, G. Gievert, F. Wilson, Final Report on Exothermic Joining in Vacuum AML-TDR-64-7.

12. J. F. Lundeburg, D. H. Lee, G. T. Burch Jr., "Impact Penetration of Manned Space Stations," Journal of Spacecraft and Rockets, February 1966.

13. P. M. Knox, Expandable Space Structure - Semirigid or Nonrigid Structures for Space Applications, ML-TDR-64-281.

14. T. L. Hoffman, Utilization of Elastic Recovery Materials for the Development of Crew Transfer Tunnels, Airlocks, and Space Maintenance Hangers, AFAPL-TR-66-2.

15. F. W. Forbes, Expandable Structures for Space Applications, July 30, 1964.

16. R. H. Christensen, R. J. Bellinfante, Some Considerations in the Fatigue Design of Launch and Spacecraft Structures, NASA CR-242.

17. Expandable Structures Design Handbook Part I RTD-TDR-63-4275

18. Expandable Structures Design Handbook Part II, RTD-TDR-63-4275

19. R. Rochen, R. C. Clark, N.S. Hassen, W. J. McKillip, Aerospace Expandable Structures and Maintenance Support Devices, Vol 1, AFAPL-TR-65-40.

20. F. K. Daisey, Space Power Tool RTD-TDR-63-4227, April 1964.

21. Applied Exothermics, Whittaker Corp., February 1966.

22. Norman E. Weare, Exothermic Brazing - A New Convience for Permanent Tube Connections, Whittaker Corp.

23. O. F. Danielson, "Materials for Manned Spacecraft", AIAA/ASME Seventh Structures and Materials Conference

24. G. Strouhol, D. M. Curry, J. M. Janney, "Thermal Protection System Performance of the Apollo Command Module," AIAA/ASME Seventh Structures and Materials Conference.

25. D. G. Younger, R. M. Edmiston, R. G. Cram, Nonrigid and Semirigid Structures for Expandable Spacecraft, ASD-TDR-62-568, November 1962.

26. R. L. Stark, F. B. Tatom, L. M. Bhalla, H. W. Hsu, Lunar Dust and Debris Hazards Associated with the Manned Flying System, NASA-CR-61106, October 1965.

27. S. I. Firstman, Current Concepts and Issues of In-Space Support, August 1964.

28. J. F. Shea, M. I. Yavymovych, The Manned Orbital Laboratory, August 1964.

29. Evelyn Falkowsky, Saturn Illustrated Chronology, April 1957 to June 1964, NASA-TM-X-54992.

30. M. J. Rafflensperger, Future Manned Earth Orbital Activities, NASA-TM-X-56227.

31. G. P. Chubb, An Evaluation of Proposed Applications of Remote Handling In Space. AMRL-TR-64-98, October 1964.

32. R. L. Keeney, Environmental Effects on Space Structures, April 1964.

33. High-Vacuum Technology, Testing and Measurement Meeting, NASA-TM-X-1268 June 1965.

34. L. R. Lankston, J. A. Wrede, Encapsulated Foam for Aerospace Support Application, AFAPL-TR-64-139,December 1964.

35. J. S. Islinger, Research and Development of Design Concepts for Sealing Applications in Aerospace Vehicle Cabins, ASD-TR-61-696,February 1962.

36. C. O. Keffer, Experimental Investigation of Packaging and Deployment Characteristics of an Inflatable Toroidal-Space-Station Configuration, NASA-TM-X-1079

37. Study on Methods of Structurally Evaluating Expandable Structures Having Potential Space Applications,August 1963.

38. D. J. Hamman, E. M. Wyler, Space Environment Effects on Materials and Components, Vol. 2, April 1964.

39. L. L. Allen, P. L. Vitkus, J. L. Ham, F. J. Brock, Adhesion and Cohesion of Metals,April 1965.

40. P. M. Winslow, D. Horiwitz, D. V. McIntyre, Adhesion and Cohesion of Metals in the Space Environment

41. P. R. Hill, E. Schnitzer, "Rotating Manned Space Station", Astronautics, September 1962.

42. R. A. Berglund, "AEMT Space Station Design", Astronautics, September 1962.

43. R. S. Osborne, C. O. Keffer, G. Look, "Materials and Structures for Space Stations", Astronautics, September 1962.

44. A Report on the Research and Technological Problems of Manned Rotating Spacecraft, NASA-TN-D-1504.

45. Aviation Week and Space Technology,August 22, 1966.

46. I. W. Russell, N. S. Hanssen, W. J. McKillip, Development of an Inflatable Self-Rigidizing Space Shelter and Solar Collector from Honeycomb Sandwich Material, APL-TOR-64-29

47. W. J. Normyle, "NASA Adapting S-4B for Space Station," Aviation Week and Space Technology, September 5, 1966.

48. Aviation Week and Space Technology, July 18, 1966.

49. W. J. Normyle, "Air Force Given Manned Space Role," Aviation Week and Space Technology, August 30, 1965.

50. B. J. Loret, Optimization of Manned Orbital Satellite Vehicle Design with respect to Artificial Gravity, ASD-TR-61-688,December 1961.

51. D. J. Withey, Repair of Leaks in an Aerospace Environment, ASD-TOR-62-1015, February 1963.

52. F. J. Schollhammer, Hamilton Standard Division of United Aircraft Corp., Private Communication.

53. Norman E. Weare, Whittaker Corp. Private Communication

54. R. D. Engquist, Hughes Aircraft Co., Private Communication

55. R. C. McMaster, Nondestructive Testing Handbook, 1959.

56. W. D. Rummel, "Characterization and Evaluation of 2014 Aluminum Alloy by Eddy Current Conductivity Techniques", Materials Evaluation, June 1966.

57. D. Hagemaier, G. J. Base, Analysis of the Heat Affected Zone in 2014-T6 Wedments by Nondestructive (Eddy Current) Methods

58. Radiography in Modern Industry,Eastman Kodak Co.

59. Technical Operations, Inc. - Product Bulletin

60. N. A. Normann, M. D., Hamilton Standard Division, United Aircraft Corp., Private Communication

APPENDIX A

Appendix I Component Qualification Analysis
UCD REFLECTOSCOPE

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
1	DD151	Cap	CE	150pf 1000V Ceramic Disk	None	2		
2	CM15-E431G	Cap	AR	430pf 500V ± 5% Mica	None	CM15ED431GM3	EM	MIL-C-5C/1E (SA-6, -7)
3	IPJ-202-G	Cap	AR	.002μf 100V ± 2% Polystyrene	None	3		
4	IPJ-103-G	Cap	AR	.01μf 100V ± 2% Polystyrene	None	3		
5	IPJ-393-G	Cap	AR	.039μf 100V ± 2% Polystyrene	None	3		
6	DD102G	Cap	CE	.001μf 1000V Ceramic Disc	None	SCK60BX102K	ER	50M60187
7	DD151	Cap	CE	150pf 1000V Ceramic Disc	None	2		
8	DD470	Cap	CE	47pf 600V Ceramic Disc	None	SCK60BX470K	ER	50M60187
9	DD511	Cap	CE	510pf 1000V Ceramic Disc	None	2		
10	IPJ-502-G	Cap	AR	.005μf 500V Polystyrene ± 2%	None	3		
11	IPJ-503-G	Cap	AR	.05μf 300V Polystyrene ± 2%	None	3		
12	IPJ-504-G	Cap	AR	.5μf 300V Polystyrene ± 2%	None	3		
13	DD100	Cap	CE	100pf 1000V Ceramic Disc	None	SCK60BX100K	ER	50M60187
14	DD391	Cap	CE	390pf 600V Ceramic Disc	None	2		
15	1N914	Diode	GE	N/A	LEM PPL			
16	1N270	Diode	SY	N/A	None	1N270	FA	Fact III Level 4AB
17	1N270	Diode	SY	N/A	None	1N270	FA	Fact III Level 4AB
	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
19	1N90	Diode	GE	N/A	None	1N90	FA	Fact III Level 4AB

UCD REFLECTOSCOPE

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
20	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
21	1N914A	Diode	TE	N/A	None	1N914A	FA	Fact III Level 4AB
22	1N914A	Diode	TE	N/A	None	1N914A	FA	Fact III Level 4AB
23	1N90	Diode	GE	N/A	None	1N90	FA	Fact III Level 4AB
24	1N2326	Diode	RCA	N/A	None	1N2326	FA	Fact III Level 4AB
25	1N1313A	Diode	HO	N/A	None	1N1313A	FA	Fact III Level 4AB
26	1N914	Diode	GE	N/A	LEM PPL			
27	1N270	Diode	SY	N/A	None	1N270	FA	Fact III Level 4AB
28	RS6	Diode	HO	N/A	None	RS6	FA	Fact III Level 4AB
29	133-010-03	Conn	AM	10 Pin Male	None	1		
30	2N1309	Trans.	AP	N/A	None	2N1309	FA	Fact III Level 4AB
31	2N1309	Trans.	AP	N/A	None	2N1309	FA	Fact III Level 4AB
32	2N1308	Trans.	AP	N/A	None	2N1308	FA	Fact III Level 4AB
33	2N1309	Trans.	AP	N/A	None	2N1309	FA	Fact III Level 4AB
34	2N585	Trans.	RCA	N/A	None	2N585	FA	Fact III Level 4AB
35	2N1309	Trans.	AP	N/A	None	2N1309	FA	Fact III Level 4AB
36	2N384	Trans.	RCA	N/A	None	2N384	FA	Fact III Level 4AB
37	2N1308	Trans.	AP	N/A	None	2N1308	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
38	2N1605	Trans.	RA	N/A	None	2N1605	FA	Fact III Level 4AB
39	2N461	Trans.	RCA	N/A	None	2N461	FA	Fact III Level 4AB
40	2N698	Trans.	GE	N/A	None	2N698	FA	Fact III Level 4AB
41	2N698	Trans.	GE	N/A	None	2N698	FA	Fact III Level 4AB
42	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
43	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
44	EB Series	Res	AB	7.5K 1/2W ± 5%	LEM PPL			
45	EB Series	Res	AB	100Ω 1/2W ± 5%	LEM PPL			
46	EB Series	Res	AB	180K 1/2W ± 5%	LEM PPL			
47	EB Series	Res	AB	15K 1/2W ± 5%	LEM PPL			
48	EB Series	Res	AB	2K 1/2W ± 5%	LEM PPL			
49	EB Series	Res	AB	12K 1/2W ± 5%	LEM PPL			
50	EB Series	Res	AB	2K 1/2W ± 5%	LEM PPL			
51	EB Series	Res	AB	1.8K 1/2W ± 5%	LEM PPL			
52	EB Series	Res	AB	33Ω 1/2W ± 5%	LEM PPL			
53	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
54	EB Series	Res	AB	5.6K 1/2W ± 5%	LEM PPL			
55	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
56	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
57	EB Series	Res	AB	1.2K 1/2W ± 5%	LEM PPL			
58	EB Series	Res	AB	1.5K 1/2W ± 5%	LEM PPL			
59	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
60	EB Series	Res	AB	240Ω 1/2W ± 5%	LEM PPL			
61	EB Series	Res	AB	15K 1/2W ± 5%	LEM PPL			
62	MTC-1	Pot	MA	10K 1/4W 500V Max	None	1		
63	MTC-1	Pot	MA	50K 1/4W 500V Max	None	1		
64	MTC-1	Pot	MA	10K 1/4W 500V Max	None	1		
65	EB Series	Res	AB	6.2K 1/2W ± 5%	LEM PPL			
66	EB Series	Res	AB	510Ω 1/2W ± 5%	LEM PPL			
67	MTC-1	Pot	MA	2.5K 1/4W 500V Max	None	1		
68	EB Series	Res	AB	910Ω 1/2W ± 5%	LEM PPL			
69	EB Series	Res	AB	24K 1/2W ± 5%	LEM PPL			
70	EB Series	Res	AB	200K 1/2W ± 5%	LEM PPL			
71	EB Series	Res	AB	51K 1/2W ± 5%	LEM PPL			
72	EB Series	Res	AB	2K 1/2W ± 5%	LEM PPL			
73	EB Series	Res	AB	24K 1/2W ± 5%	LEM PPL			
74	EB Series	Res	AB	4.7K 1/2W ± 5%	LEM PPL			

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
75	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
76	MTC-1	Pot	MA	2K 1/4W 500V Max	None	(1)		
77	EB Series	Res	AB	7.5K 1/2W ± 5%	LEM PPL			
78	EB Series	Res	AB	20K 1/2W ± 5%	LEM PPL			
79	EB Series	Res	AB	470K 1/2W ± 5%	LEM PPL			
80	EB Series	Res	AB	820K 1/2W ± 5%	LEM PPL			
81	EB Series	Res	AB	30K 1/2W ± 5%	LEM PPL			
82	EB Series	Res	AB	30K 1/2W ± 5%	LEM PPL			
83	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
84	MTC-1	Pot	MA	2K 1/4W 500V Max	None	(1)		
85	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
86	EB Series	Res	AB	22K 1/2W ± 5%	LEM PPL			
87	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
88	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
89	MTC-1	Pot	MA	25K 1/4W 500V Max	None	(1)		
90	EB Series	Res	AB	30K 1/2W ± 5%	LEM PPL			
91	EB Series	Res	AB	47K 1/2W ± 5%	LEM PPL			
92	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			

ITEM NO	COMPONENT DATA				QUALIFICATION DATA	SUBSTITUTION DATA		
	PART NUMBER	PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
93	EB Series	Res	AB	270 Ω 1/2W ± 5%	LEM PPL			
94	DQB005B40	Cap	AR	330pf 300V Style 15	None	(3)		
95	30GAD68	Cap	SP	.0068μf 3KV Ceramic	None	(2)		
96	5C9	Cap	SP	.22μf 25V ± 20% Ceramic	None	(2)		
97	5C9	Cap	SP	.22μf 25V ± 20% Ceramic	None	(2)		
98	NLW10-15	Cap	CDE	10μf 15V +10-15% -40° to 85°C	None	(2)		
99	DQB003C23	Cap	AR	100μf 500V Style 25 Mica	None	(3)		
100	NLW10-15	Cap	CDE	Same as 100	None	(2)		
101	DQB003C23	Cap	AR	Same as 101	None	(3)		
102	NLW10-15	Cap	CDE	Same as 100	None	(2)		
103	5C9	Cap	SP	Same as 98	None	(2)		
104	ET8805	Cap	ER	.01μf 100V ± 80%-20%	None	(2)		
105	5C7	Cap	SP	.1μf 25V ± 20%	None	CKR06CW104K	ER	
106	DD200	Cap	CE	20pf 600V ± 10% Ceramic	None	(2)		
107	5C7	Cap	SP	Same as 107	None	CKR06CW104K	ER	
108	5C7	Cap	SP	Same as 107	None	CKR06CW104K	ER	
109	NLW10-15	Cap	CDE	Same as 100	None	(2)		
110	Type SM	Cap	RMC	.005μf 500V +80-20% Ceramic	None	(2)		
111	NLW10-15	Cap	CDE	Same as 100	None	(2)		

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
112	NLW10-15	Cap	CDE	Same as 100	None	2		
113	DD102-G	Cap	CE	$.001\mu f$ 1000V $\pm$ 10%	None	2		
114	NLW10-15	Cap	CDE	Same as 100	None	2		
115	NLW10-15	Cap	CDE	Same as 100	None	2		
116	NLW10-15	Cap	CDE	Same as 100	None	2		
117	5C9	Cap	SP	Same as 98	None	2		
118	5C9	Cap	SP	Same as 98	None	2		
119	30GAD68	Cap	SP	$.0068\mu f$ 3KV	None	2		
120	NLW10-15	Cap	CDE	Same as 100	None	2		
121	DD102-G	Cap	CE	Same as 115	None	2		
122	DD203-G	Cap	CE	$.02\mu f$ 500V	None	2		
123	H3	Cap	CDE	$.02\mu f$ 50V	None	2		
124	1N98A	Diode	GE	N/A	None	1N98A	FA	Fact III Level 4AB
125	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
126	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
127	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
128	1N64G	Diode	GE	N/A	None	1N64G	FA	Fact III Level 4AB
129	1N90	Diode	GE	N/A	None	1N90	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
130	VR8.5A	Diode	SA	N/A	None	VR8.5A	FA	Fact III Level 4AB
131	VR8.5A	Diode	SA	N/A	None	VR8.5A	FA	Fact III Level 4AB
132	1N90	Diode	GE	N/A	None	1N90	FA	Fact III Level 4AB
133	1N914	Diode	GE	N/A	LEM PPL			
134	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
135	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
136	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
137	1N456A	Diode	TE	N/A	None	1N456A	FA	Fact III Level 4AB
138	1N995	Diode	SY	N/A	None	1N995	FA	Fact III Level 4AB
139	1N995	Diode	SY	N/A	None	1N995	FA	Fact III Level 4AB
140	1N456	Diode	TE	N/A	None	1N456	FA	Fact III Level 4AB
141	133-010-03	Conn	AM	10 Pin Male	None	1		
142	70F474A1	Choke	MI	470μh Q=59 790KC Fe Core	None	1		
143	70F103A1	Choke	MI	1mh Q=50 790KC Fe Core	None	1		
144	50A3375	Coil	SR	157μh	None	1		
145	50A3376	Coil	SR	157μh	None	1		
146	70F474A1	Choke	MI	470μh Q=59 790KC Fe Core	None	1		
147	2082-13	Choke	CA	680μh	None	1		

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
148	69F104A1	Choke	MI	100μh Q=50 790KC Fe Core	None	1		
149	70F154A1	Choke	MI	150μh Q=55 790KC Fe Core	None	1		
150	70F154A1	Choke	MI	150μh Q=55 790KC Fe Core	None	1		
151	50A3375	Coil	SR	157μh	None	1		
152	70F474A1	Choke	MI	470μh Q=59 790KC Fe Core	None	1		
153	50A3375	Coil	SR	157μh	None	1		
154	70F474A1	Choke	MI	Same as 154	None	1		
155	2N706	Trans	MO	N/A	LEM PPL			
156	2N2089	Trans	AP	N/A	None	2N2089	FA	Fact III Level 4AB
157	2N2089	Trans	GE	N/A	None	2N2089	FA	Fact III Level 4AB
158	2N706	Trans	AP	N/A	None	2N706	FA	Fact III Level 4AB
159	2N2089	Trans	AP	N/A	None	2N2089	FA	Fact III Level 4AB
160	2N698	Trans	GE	N/A	None	2N698	FA	Fact III Level 4AB
161	2N2512	Trans	AP	N/A	None	2N2512	FA	Fact III Level 4AB
162	2N697	Trans	RCA	N/A	None	2N697	FA	Fact III Level 4AB
163	2N2512	Trans	AP	N/A	None	2N2512	FA	Fact III Level 4AB
164	2N698	Trans	GE	N/A	None	2N698	FA	Fact III Level 4AB
165	2N398B	Trans	RCA	N/A	None	2N398B	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
166	2N1226	Trans	RCA	N/A	None	2N1226	FA	Fact III Level 4AB
167	EB Series	Res	AB	82K 1/2W ± 5%	LEM PPL			
168	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
169	EB Series	Res	AB	3.3K 1/2W ± 5%	LEM PPL			
170	MTC-1	Pot	MA	50K	None	(1)		
171	EB Series	Res	AB	1.5 Meg 1/2W ± 5%	LEM PPL			
172	GB Series	Res	AB	4.7 Meg 1W ± 5%	LEM PPL			
173	EB Series	Res	AB	2.2K 1/2W ± 5%	LEM PPL			
174	EB Series	Res	AB	7.5K 1/2W ± 5%	LEM PPL			
175	EB Series	Res	AB	10Ω 1/2W ± 5%	LEM PPL			
176	EB Series	Res	AB	500Ω 1/2W ± 5%	LEM PPL			
177	EB Series	Res	AB	750Ω 1/2W ± 5%	LEM PPL			
178	EB Series	Res	AB	75Ω 1/2W ± 5%	LEM PPL			
179	EB Series	Res	AB	10Ω 1/2W ± 5%	LEM PPL			
180	EB Series	Res	AB	750Ω 1/2W ± 5%	LEM PPL			
181	EB Series	Res	AB	100Ω 1/2W ± 5%	LEM PPL			
182	EB Series	Res	AB	1.8K 1/2W ± 5%	LEM PPL			
183	EB Series	Res	AB	5.1K 1/2W ± 5%	LEM PPL			

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
184	EB Series	Res	AB	1.8K 1/2W ± 5%	LEM PPL			
185	EB Series	Res	AB	5.1K 1/2W ± 5%	LEM PPL			
186	EB Series	Res	AB	330Ω 1/2W ± 5%	LEM PPL			
187	EB Series	Res	AB	24K 1/2W ± 5%	LEM PPL			
188	MTC-1	Pot	MA	5K	None	1		
189	EB Series	Res	AB	5.1K 1/2W ± 5%	LEM PPL			
190	EB Series	Res	AB	820Ω 1/2W ± 5%	LEM PPL			
191	EB Series	Res	AB	100K 1/2W ± 5%	LEM PPL			
192	EB Series	Res	AB	100K 1/2W ± 5%	LEM PPL			
193	EB Series	Res	AB	6.8K 1/2W ± 5%	LEM PPL			
194	EB Series	Res	AB	100K 1/2W ± 5%	LEM PPL			
195	MTC-1	Pot	MA	100K	None	1		
196	EB Series	Res	AB	5.1K 1/2W ± 5%	LEM PPL			
197	EB Series	Res	AB	820Ω 1/2W ± 5%	LEM PPL			
198	EB Series	Res	AB	24K 1/2W ± 5%	LEM PPL			
199	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
200	EB Series	Res	AB	1.6K 1/2W ± 5%	LEM PPL			
201	MTC-1	Pot	MA	50K	None	1		

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
202	EB Series	Res	AB	10Ω 1/2W ± 5%	LEM PPL			
203	EB Series	Res	AB	75Ω 1/2W ± 5%	LEM PPL			
204	EB Series	Res	AB	750Ω 1/2W ± 5%	LEM PPL			
205	EB Series	Res	AB	10Ω 1/2W ± 5%	LEM PPL			
206	EB Series	Res	AB	470Ω 1/2W ± 5%	LEM PPL			
207	EB Series	Res	AB	75Ω 1/2W ± 5%	LEM PPL			
208	EB Series	Res	AB	100Ω 1/2W ± 5%	LEM PPL			
209	EB Series	Res	AB	1.8K 1/2W ± 5%	LEM PPL			
210	EB Series	Res	AB	11K 1/2W ± 5%	LEM PPL			
211	EB Series	Res	AB	22Ω 1/2W ± 5%	LEM PPL			
212	50A3384	Pot	MA	50K	None	1		
213	50A3382	Pot	MA	25K	None	1		
214	50A3381	Pot	MA	2.5 Meg	None	1		
215	GB Series	Res	AB	330K 1W ± 5%	LEM PPL			
216	50A3380	Pot	MA	500K	None	1		
217	EB Series	Res	AB	1K 1/2W ± 5%	LEM PPL			
218	92-689	Trans F.	Aladdin	Power Transformer	None	1		
219	30GAD33	Cap	SP	.0033 f 3KV ± 10% ceramic	None	2		

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
220	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
221	30GAD33	Cap	SP	.0033μf 3KV $\pm$ 10% ceramic	None	2		
222	30GAD33	Cap	SP	.0033μf 3KV $\pm$ 10% ceramic	None	2		
223	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
224	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
225	BR4-450	Cap	CDE	4mf 450V -10 + 50% AL. Elec.	None	3		
226	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
227	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
228	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
229	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
230	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
231	30GAS10	Cap	SP	.01μf 3KV $\pm$ 10% ceramic	None	2		
232	US63-HP	Diode	IRC	N/A	None	1		
233	US63-HP	Diode	IRC	N/A	None	1		
234	US48-HP	Diode	IRC	N/A	None	1		
235	CER-72	Diode	SL	N/A	None	1N2072	FA	Fact III Level 4AB
236	EB Series	Res	AB	100K 1/2W $\pm$ 5%	LEM PPL			
237	EB Series	Res	AB	100K 1/2W $\pm$ 5%	LEM PPL			
238	EB Series	Res	AB	1 Meg	LEM PPL			

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
239	EB Series	Res	AB	1 Meg 1/2W ± 5%	LEM PPL			
240	EB Series	Res	AB	1 Meg 1/2W ± 5%	LEM PPL			
241	BR-100-50	Cap	CDE	100mf 50V -10 +50% AL Can	None	3		
242	BR-500-50	Cap	CDE	500mf 50V -10 +50% AL Can	None	3		
243	BR-500-50	Cap	CDE	500mf 50V -10 +50% AL Can	None	3		
244	BR-100-50	Cap	CDE	100mf 50V -10 + 50% AL Can	None	3		
245	5C7	Cap	SP	.1μf 25V	None	CKR06CW104K	ER	
246	5C15	Cap	SP	2.2μf 25V ± 20%	None	2		
247	5C15	Cap	SP	Same as 248	None	2		
248	5C7	Cap	SP	Same as 247	None	CKR06CW104K	ER	
249	1N91	Diode	GE	N/A	SA-5, 6, 7, 9			
250	2082-12	Choke	CA	470μh	None	1		
251	2082-12	Choke	CA	470μh	None	1		
252	EB Series	Res	AB	51Ω 1/2W ± 5%	LEM PPL			
253	EB Series	Res	AB	51Ω 1/2W ± 5%	LEM PPL			
254	EB Series	Res	AB	27Ω 1/2W ± 5%	LEM PPL			
255	EB Series	Res	AB	330Ω 1/2W ± 5%	LEM PPL			
256	50B1330	Inv. Transf	MG	Step Down	None	1		

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ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTETISTIC		P/N	MFG	PROCUREMENT METHOD
257	DM19-122	Cap	AR	1200μf 500V $\pm$ 10%	None	(2)		
258	DQJ001C01	Cap	AE	.0033μf 600V Ceramic	None	(2)		
259	1N90	Diode	GE	N/A	None	1N90	FA	Fact III Level 4AB
260	2N1774	Scr Diode	GE	N/A	None	2N1774	FA	Fact III Level 4AB
261	133-010-03	Conn.	AM	10 Pin Male	None	(1)		
262	50A3372	Coil	SR	1MC	None	(1)		
263	50A3373	Coil	SR	2.25MC	None	(1)		
264	50A3374	Coil	SR	5MC	None	(1)		
265	EB Series	Res	AB	820μ 1/2W $\pm$ 5%	LEM PPL			
266	EB Series	Res	AB	9.1K 1/2W $\pm$ 5%	LEM PPL			
267	HB Series	Res	AB	100K 2W $\pm$ 5%	LEM PPL			
268	VC3D	Res	CS	1Ω 3W $\pm$ 5%	None	(2)		
269	DQK001B04	Cap	AR	20-125pf 600V Ceramic Trimmer	None	(1)		
270	UG-657-μ	Conn	AM	Female Coaxial 50Ω impedance	None			
271	UG-657-μ	Conn	AM	Female Coaxial 50Ω impedance	None			
272	EB Series	Res	AB	200Ω 1/2W $\pm$ 5%	LEM PPL			
273	50A2514	Pot	AB	10K	None	(1)		

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
274	50A3391	Pot	AB	100K	None	1		
275	50A3387	Pot	AB	5K	None	1		
276	TC-3	Switch	AHH	SPDT	None	4		
277	50A3390	Switch	CE	SPDT	None	4		
278	50A3388	Switch	CE	SPDT	None	4		
279	SC3511-PZ0	CRT	SY		None	3		
280	50A3393	Switch	CE	SPDT	None	4		
281	8XYS	Batt	Y.E.	8.7 Volts, Silcad Cells	None	1		
282	BYD6D51	Cap	CDE	.01μf 600V Ceramic	None	2		
283	BYD6D51	Cap	CDE	Same as 286	None	2		
284	NLW100-15	Cap	CDE	100μf 15V	None	3		MIL-C-62C/1A
285	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
286	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
287	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
288	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
289	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
290	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT DATA
291	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
292	2A50	Diode	SL	N/A	None	1N5171	FA	Fact III Level 4AB
293	1N135ZE	Diode	SA	N/A	None	1N1352E	FA	Fact III Level 4AB
294	RH-10	Res	DL	$6\,\Omega$ 10W $\pm$ 1%	None	3		
295	50B1314	Transf	MG	Step Down	None	1		
296	NLW100-15	Cap	CDE	100μf 15V	None	3		
297	1N3017B	Diode	MO	N/A	None	1N3017B	FA	Fact III Level 4AB
298	143-010-01	Conn	AM	10 Socket Female	None	4		
299	143-010-01	Conn	AM	10 Socket Female	None	4		

Appendix I Component Qualification Analysis
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ITEM NO	COMPONENT DATA				QUALIFICATION DATA	SUBSTITUTION DATA		
	PART NUMBER	PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
1	5001-10	Switch	GH	10 Position	None	4		
2		Coil		Inductor	None	1		
3	HFA100A	Var Cap	Hamarlund	4pf to 100pf	None	1		
4	DM15-211	Cap	AR	210pf 500V Mica ± 5%	None	DM15-211	EM	
5	DM15-301	Cap	AR	300pf	None	DM15-301	EM	
6	DM15-391	Cap	AR	390pf	None	DM15-391	EM	
7	DM15-481	Cap	AR	480pf	None	DM15-481	EM	
8	DM15-571	Cap	AR	570pf	None	DM15-571	EM	
9	DM15-661	Cap	AR	660pf	None	DM15-661	EM	
10	DM15-751	Cap	AR	750pf	None	DM15-751	EM	
11	DM15-841	Cap	AR	840pf	None	DM15-841	EM	
12	DM15-931	Cap	AR	930pf	None	DM15-931	EM	
13	DM15-102	Cap	AR	1000pf 500V Mica ± 5%	None	DM15-102	EM	
14	DM15-151	Cap	AR	150pf 500V Mica ± 5%	None	DM15-151	EM	
15	DM15-151	Cap	AR	150pf 500V Mica ± 5%	None	DM15-151	EM	
16		Cap	Hopkins	0.1μf paper mylar	None	2		
17	CRE623	Cap	AE	100μf 25 V Electrolytic	None	3		
18	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
19	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
20	2N3643	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
21	2N3643	Trans	FA		SA-5, 6, 7, 9	2N3643	FA	Fact III Level 4AB
22		RES	OH	68K 1/2W ± 10% CC	SA-6, 7			
23		RES	OH	4.7K 1/2W ± 10% CC	SA-6, 7			
24		RES	OH	6.8K 1/2W ± 10% CC	SA-6, 7			
25		RES	OH	6.8K 1/2W ± 10% CC	SA-6, 7			
26		RES	OH	1.2K 1/2W ± 10% CC	SA-6, 7			
27		RES	OH	150K 1/2W ± 10% CC	SA-6, 7			
28		RES	OH	270K 1/2W ± 10% CC	SA-6, 7			
29		RES	OH	270K 1/2W ± 10% CC	SA-6, 7			
30		RES	OH	15K 1/2W ± 10% CC	SA-6, 7			
31		RES	OH	820K 1/2W ± 10% CC	SA-6, 7			
32		RES	OH	220K 1/2W ± 10% CC	SA-6, 7			
33	420M	Meter	Triplett	0-500μA D.C.	None	1		
34	510	Pot	Spectrof	100Ω 10 turn Linear	None	4		
35	A47-5000	Pot	CS	5000 Ω 1/2W	None	1		
36		RES	OH	39Ω 1/2W ± 5% CC	SA-6, 7			
37		RES	OH	39Ω	SA-6, 7			
38		RES	OH	3.9K	SA-6, 7			

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
39		RES	OH	18K 1/2W ± 5% CC	SA-6, 7			
40		RES	OH	3.9K 1/2W ± 5% CC	SA-6, 7			
41		RES	OH	2.7K 1/2W ± 5% CC	SA-6, 7			
42		RES	OH	3.9K 1/2W ± 5% CC	SA-6, 7			
43		RES	OH	2.7K 1/2W ± 5% CC	SA-6, 7			
44		RES	OH	47Ω 1/2W ± 5% CC	SA-6, 7			
45		RES	OH	47Ω 1/2W ± 5% CC	SA-6, 7			
46		RES	OH	1.2K 1/2W ± 5% CC	SA-6, 7			
47		RES	OH	680K 1/2W ± 5% CC	SA-6, 7			
48		RES	OH	2.7K 1/2W ± 5% CC	SA-6, 7			
49		RES	OH	2.7K 1/2W ± 5% CC	SA-6, 7			
50		RES	OH	3.9K 1/2W ± 5% CC	SA-6, 7			
51		RES	OH	180K 1/2W ± 5% CC	SA-6, 7			
52	P42DK	Cap	Hopkins	.4μf 200V ± 10% Paper Mylar	None	2		
53	P42DK	Cap	Hopkins	.4μf 200V ± 10% Paper Mylar	None	2		
54	M968P	Cap	Electron	.05μf 150V ± 20% Paper Mylar	None	2		
55	5A4	Diode	1R		None	5A4	FA	Fact III Level 4AB
56	5A4	Diode	1R		None	5A4	FA	Fact III Level 4AB
57	1N914	Diode	GE		LEM PPL	1N914	FA	Fact III Level 4AB

ITEM NO	PART NUMBER	COMPONENT DATA			QUALIFICATION DATA	SUBSTITUTION DATA		
		PART NAME	MFG	ELECTRICAL CHARACTERISTIC		P/N	MFG	PROCUREMENT METHOD
58	1N914	Diode	GE		LEM PPL	1N914	FA	Fact III Level 4AB
59	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
60	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
61	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
62	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
63	2N3638A	Trans	FA		None	2N3638A	FA	Fact III Level 4AB
64	TRX146X	Batt	MA	8.4V	None	1		
65	TRX146X	Batt	MA	8.4V	None	1		
66	TRX146X	Batt	MA	8.4V	None	1		