## PASSIVE THERMAL SESSION TOPICS

### Passive Thermal Paper Session #1 (Wednesday 7:30AM to 11:30AM)

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THERMAL OPTIMIZATION AND ASSESSMENT OF A LONG DURATION CRYOGENIC PROPELLANT DEPOT

Ryan Honour, Kennedy Space Center
Robert Kwas, Kennedy Space Center
Gary O’Neil, Kennedy Space Center
Bernard Kutter, United Launch Alliance

ABSTRACT

A Cryogenic Propellant Depot (CPD) operating in Low Earth Orbit (LEO) could provide many near term benefits to NASA space exploration efforts. These benefits include elongation/extension of spacecraft missions and reduction of launch vehicle up-mass requirements. Some of the challenges include controlling cryogenic propellant evaporation and managing the high costs and long schedules associated with new spacecraft hardware development. This paper describes a conceptual CPD design that is thermally optimized to achieve extremely low propellant boil-off rates. The CPD design is based on existing launch vehicle architecture and its thermal optimization is achieved using current passive thermal control technology. Results from an integrated thermal model are presented showing that this conceptual CPD design can achieve propellant boil-off rates well under 0.05% per day, even when subjected to the LEO thermal environment.
DEVELOPMENT OF THE GPM OBSERVATORY THERMAL VACUUM TEST MODEL

Kan Yang, Goddard Spaceflight Center
Hume Peabody, Goddard Spaceflight Center

ABSTRACT

A software-based thermal modeling process was documented for generating the thermal panel settings necessary to simulate worst-case on-orbit flight environments in an observatory-level thermal vacuum test setup. The method for creating such a thermal model involved four major steps: (1) determining the major thermal zones for test as indicated by the major dissipating components on the spacecraft, then mapping the major heat flows between these components; (2) finding the flight equivalent sink temperatures for these test thermal zones; (3) determining the thermal test ground support equipment (GSE) design and initial thermal panel settings based on the equivalent sink temperatures; and (4) adjusting the panel settings in the test model to match heat flows and temperatures with the flight model. The observatory test thermal model developed from this process allows quick predictions of the performance of the thermal vacuum test design.

In this work, the method described above was applied to the Global Precipitation Measurement (GPM) core observatory spacecraft, a joint project between NASA and the Japanese Aerospace Exploration Agency (JAXA) which is currently being integrated at NASA Goddard Space Flight Center. From preliminary results, the thermal test model generated from this process shows that the heat flows and temperatures match fairly well with the flight thermal model, indicating that the test model can simulate fairly accurately the conditions on-orbit. However, further analysis is needed to determine the best test configuration possible to validate the GPM thermal design before the start of environmental testing later this year. Also, while this analysis method has been applied solely to GPM, it should be emphasized that the same process can be applied to any mission to develop an effective test setup and panel settings which accurately simulate on-orbit thermal environments.
RETURN TO MERCURY: AN OVERVIEW OF THE MESSENGER SPACECRAFT THERMAL CONTROL SYSTEM DESIGN AND UP-TO-DATE ON-ORBIT FLIGHT PERFORMANCE

Carl J. Ercol, The Johns Hopkins University Applied Physics Laboratory

ABSTRACT

AT 01:00 UTC on March 18, 2011, MESSENGER (MErcury Surface, Space ENvironment, GEochemistry, and Ranging) became the first spacecraft to achieve orbit around the planet Mercury. Designed and built by The Johns Hopkins University Applied Physics Laboratory in conjunction with the Carnegie Institution of Washington, MESSENGER was launched on August 3, 2004 and has recently completed its primary mission of a 1-year orbital phase to study Mercury. Currently, MESSENGER has successfully completed the yearlong primary mission and is in the initial phase of a yearlong extended mission to gather more science with a shortened orbit period. Prior to orbit injection at Mercury the spacecraft completed a 7-year cruise phase that has included a flyby of the Earth (August 2005), two flybys of Venus (October 2006 and June 2007), and three flybys of Mercury (January and October, 2008 and October 2009). The January 2008 Mercury flyby marked the first spacecraft visit since Mariner 10 (1975) and made MESSENGER the first spacecraft to encounter Mercury when near the planet’s perihelion. This paper will provide an overview of the thermal design challenges for both the cruise and orbital phases and the flight temperature and power data that verify the performance of the thermal control subsystem over the mission to date.
Increasingly, aerospace structures are designed and fabricated using composite materials. For the thermal analyst, this means that models often must incorporate materials with orthotropic, temperature-dependent properties on complex geometries. It is especially important that thermal models account for the typically low transverse conductivity through composite panels, since this can lead to significant temperature differences across panels. Complicating the problem, aerospace structures frequently consist of hundreds of distinct composite layups applied to arbitrary geometry. Structural analysis tools such as Nastran offer analysts convenient methods for defining composite layups and applying them to the analysis mesh (e.g., with PCOMP cards), but commercial thermal analysis packages currently do not feature such conveniences. ATA Engineering, Inc. (ATA) has developed a method to automatically import an arbitrary number of composite layups defined in Nastran into a Thermal Desktop model and apply them to the analysis mesh. This method was used successfully during system-level analysis of a crewed spacecraft that featured over 200 distinct composite layups, each with several dozen material layers. For the thermal model, each of the layups was automatically translated into an effective orthotropic material and applied at the correct orientation to solid elements representing the structure. The present paper discusses the method with a comparison to results from detailed panel models (which would be unsuitable for system-level analysis), and introduces a refinement that will account for radiation across honeycomb layers as a parallel heat transfer path incorporated into the effective temperature-dependent orthotropic properties. The method shows excellent steady-state and transient agreement with the detailed models at a fraction of the computational cost, making it ideal for creating system-level models of aerospace structures featuring composite panels.
The CRYOTE concept is for a low cost orbital testbed designed to perform cryogenic fluid management experiments in a micro-gravity environment. In lieu of this concept, a ground test article was developed to characterize heat loads in a 1G environment. The purpose of this study is to anchor thermal and fluid system models to CRYOTE ground test data. The CRYOTE ground test article was jointly developed by Innovative Engineering Solutions, United Launch Alliance and NASA KSC. The test article is constructed out of a titanium alloy tank, composite skirt, an external secondary payload adapter ring, thermal vent system, multi layer insulation and various data acquisition instrumentation. In efforts to understand heat loads throughout this system, the GTA is subjected to a series of tests in a vacuum chamber at Marshall Space Flight Center using nitrogen. By anchoring analytical models against test data, higher fidelity thermal environment predictions can be made for future flight articles demonstrating critical cryogenic fluid management technologies such as system chilldown, transfer, pressure control and long term storage. Significant factors that influence heat loads include radiative environments, multi-layer insulation performance, tank fill levels and pressures and even contact conductance coefficients. This research demonstrates how analytical thermal/fluid networks are established and includes supporting rationale for specific thermal responses seen during testing.
DESIGN AND ANALYSIS OF JMAPS INSTRUMENT THERMAL CONTROL SYSTEM

Triem T. Hoang, TTH Research Inc.
Donald E. Wilson, TTH Research Inc.
Robert W. Baldauff, Praxis Inc.
Kenneth M. Weldy, U.S. Naval Research Laboratory

ABSTRACT

The Joint Milli-Arcsecond Pathfinder Survey (JMAPS) is a Department of the Navy space astronomy mission. The goal of the mission is to update the bright star catalogs for the purpose of attitude determination. The JMAPS instrument consists of a single-aperture optical telescope assembly (OTA), Focal Plane Assembly (FPA) and supporting electronics. The FPA needs to be maintained below 193K with a stability requirement of less than 10mK. Also, the stringent optical distortion budget levies an additional stability requirement of less than 0.15K on the OTA over a period of five days. Lastly, the JMAPS jitter specification compels the instrument thermal control system (TCS) to be passive (i.e. containing no mechanical moving parts). This paper presents the design, analytical simulations, and results of some component testing of the TCS to demonstrate how it meets the challenging JMAPS thermal requirements.
CUBE FLUX METHOD TO GENERATE SPACECRAFT THERMAL ENVIRONMENTS

Siraj A. Jalali, Ph.D., P.E., Oceaneering Space Systems

ABSTRACT

Spacecrafts are exposed to various environments that are not present at the surface of the earth, like plasmas, neutral gases, x-rays, ultraviolet (UV) irradiation, high energy charged particles, meteoroids, and orbital debris. The interaction of these environments with spacecraft cause degradation of materials, contamination, spacecraft glow, charging, thermal changes, excitation, radiation damage, and induced background interference. The damaging effects of natural space and atmospheric environments pose difficult challenges for spacecraft designers. ISS/Shuttle thermal model was used to develop a program to determine environment around an orbiting spacecraft. The method was applied to compare environments around the ISS/Shuttle in Earth and Mars orbits. The method was also applied on a Satellite in Lower Earth Orbit (LEO) and Geosynchronous Orbit (GEO) and results were compared.

To determine the thermal environments around the ISS/shuttle 1 cubic foot arithmetic cubes were placed 1 foot above the surfaces where thermal environments were needed. The ISS/Shuttle was placed in Earth and Mars orbits with required beta, attitudes, and altitude. The applicable solar, Albedo, and IR fluxes were applied on the model depending upon summer or winter solstice. Model was analyzed such that absorbed solar fluxes and surface temperatures of all cube surfaces were obtained. A routine (HTFLXCAL) was developed to calculate Infrared fluxes for all cube surfaces using cube absorbed solar fluxes and surface temperatures. The solar and infrared fluxes at a cube location were used to calculate orbital sink temperatures at that location. The sink temperatures at a cube location for tools, spacecraft surfaces, or space suit are extreme temperatures those components will be exposed to at that location.

The cube flux method has been developed previously also, but the method presented here is efficient and simpler since the space vehicle model and flux generation routine (HTFLXCAL) are run from Thermal Desktop® in a single run, and Solar and IR fluxes for all cube locations are generated. The sink temperatures generation routine for required materials using Solar and IR fluxes is also part of the main routine.
TFAWS2012-PT-08

THERMAL MODELING OF THE SOLAR PROBE CUP FOR THE SOLAR PROBE PLUS MISSION

Mark Freeman, Smithsonian Astrophysical Observatory (SAO)

ABSTRACT

The Solar Probe Cup (SPC) is one instrument in the Solar Wind Electrons, Alphas, and Protons (SWEAP) suite for the Solar Probe Plus (SPP) mission, and the only one with direct exposure at the closest solar approach of about 8.5 solar radii from the sun’s surface. At this distance, the solar flux is approximately 700,000 W/m². The SPC (which is a Faraday cup) must be made from materials that withstand temperatures in the range of 1000-1700°C. We will discuss the materials used and some of the special testing required for those materials (for thermal performance and other reasons), as well as some special modeling issues with such a high-temperature instrument. The first round of material testing has taken place, and some results will be discussed, as will our plans for future material and thermal model testing. The instrument is currently in Phase B, a large component of which is Technology Development in the areas mentioned above, since no previous Faraday cup has been exposed to anything approaching this thermal environment. The current plan is for a Demonstration Model to be tested to TRL-6 by December 2013.
MODELING OF HEAT TRANSFER AND EROSION OF REFRACTORY MATERIAL DUE TO ROCKET PLUME IMPINGEMENT

Mr. Michael F. Harris, QinetiQ
Dr. Bruce T. Vu, Kennedy Space Center

ABSTRACT

CR Tech’s Thermal Desktop-SINDA/FLUINT software was used in the thermal analysis of a flame deflector design for Launch Complex 39B at Kennedy Space Center, Florida. The analysis of the flame deflector takes into account heat transfer due to plume impingement from expected vehicles. The heat flux from the plume was computed using computational fluid dynamics provided by Ames Research Center in Moffet Field, California. The results from the CFD solutions were mapped onto a 3-D Thermal Desktop model of the flame deflector using the boundary condition mapping capabilities in Thermal Desktop. The ablation subroutine in SINDA/FLUINT was then used to model the erosion of the refractory material, such as Fondu Fyre.
CONDUCTIVE WHITE THERMAL CONTROL POLYIMIDE FILMS WITH ATOMIC OXYGEN DURABILITY

Garrett D. Poe, Ph.D., NeXolve Corporation

ABSTRACT

Polyimides are attractive materials for passive thermal control due to their low mass, solar radiation resistance, and cryogenic flexibility. However, traditional polyimides are highly colored and as such are highly absorptive, intrinsically insulative, and have little or no intrinsic durability to atomic oxygen (AO). These inherent property shortfalls are usually overcome with the use of fillers and coatings. For example, the orange color can be overcoated with vapor deposited aluminum (VDA) or other metals to reduce the amount of heat absorbed; conductive carbon black is additives can be incorporated into the bulk of the polyimide used to render the material electrostatically dissipative; and the film can be protected from atomic oxygen with coatings. However, these approaches have associated shortfalls which present additional engineering challenges. For example, VDA has a low emissivity; conductive carbon black raises the thermal absorptivity; and coatings, when cracked or scratched, are no longer effective at protecting from AO. In response to these inherent limitations of existing thermal control materials, NeXolve has sought to overcome these challenges with Thermalbright and CORINbright thermal control polyimides, which are conductive white polyimides that are intrinsically AO durable and available in roll form. Thermal and mechanical properties will be presented, as well as MISSE flight data.