Autonomous Aerobraking: Thermal Analysis and Response Surface Development

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A high-fidelity thermal model of the Mars Reconnaissance Orbiter was developed for use in an autonomous aerobraking simulation study. Response surface equations were derived from the high-fidelity thermal model and integrated into the autonomous aerobraking simulation software. The high-fidelity thermal model was developed using the Thermal Desktop software and used in all phases of the analysis. The use of Thermal Desktop exclusively, represented a change from previously developed aerobraking thermal analysis methodologies. Comparisons were made between the Thermal Desktop solutions and those developed for the previous aerobraking thermal analyses performed on the Mars Reconnaissance Orbiter during aerobraking operations. A variable sensitivity screening study was performed to reduce the number of variables carried in the response surface equations. Thermal analysis and response surface equation development were performed for autonomous aerobraking missions at Mars and Venus.

Nomenclature

AADS	=	autonomous aerobraking development software
ALC_p	=	aluminum honeycomb core specific heat, J/kg-K
ALk	=	aluminum honeycomb core thermal conductivity, W/m-K
b_0	=	response surface intercept coefficient
b_i	=	response surface equation main effect term coefficients
b_{ii}	=	response surface equation quadratic term coefficients
b_{ij}	=	response surface equation 2nd order interaction term coefficients
b_{ijk}	=	response surface equation 3rd order interaction term coefficients
C_H	=	heat transfer coefficient
CCD	=	central composite design
CR	=	contact resistance, W/m^2-K
DP	=	drag pass duration, sec
DOE	=	design of experiments
DSMC	=	direct simulation Monte Carlo
FSC_p	=	M55J composite facesheet specific heat, J/kg-K
FSE	=	M55J composite facesheet emissivity
FSk	=	M55J composite facesheet thermal conductivity, W/m-K
GRETA	=	generic response-surface equation thermal analysis program
i	=	summation index
ITJE	=	improved triple junction solar cell emissivity
IT	=	initial solar panel temperature, °C
k	=	number of factors
l	=	number of levels
m	=	point on the spacecraft for which a response surface equation has been derived
M	=	DOE matrix dimension
MD	=	solar cell mass distribution, kg
MGS	=	Mars Global Surveyor
MOI	=	mars orbit insertion

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MRO	=	Mars Reconnaissance Orbiter
n	=	number of factors
N	=	DOE matrix dimension
OFM	=	outboard panel M55J facesheet mass distribution, kg
P	=	orbit period, hr
Q_s	=	solar and planetary heat flux, W/cm^2
R^2 *adj*	=	coefficient of determination, R squared adjusted
RHO	=	atmospheric density, kg/km^3
RSE	=	response surface equation
T_m	=	temperature (°C) of the mth point on the solar array
V	=	periapsis velocity, km/s
X_i	=	independent variable
ΔV	=	change in velocity, km/s
ρ_∞	=	freestream density, kg/km^3
σ	=	standard deviation

I. Introduction

THERE are several challenges associated with placing a spacecraft in orbit around any planetary body. Often, mission design trade studies are made to maximize payload and minimize propellant mass. A mass efficient technique in terms of propellant use that has been used successfully by past missions is aerobraking[1,2,3,4]. After propulsively establishing a high-eccentricity, long-period orbit, aerobraking reduces an orbital period and eccentricity to a desired science orbit by passing through the upper atmosphere multiple times and using the drag on the spacecraft to reduce velocity. Atmospheric drag reduces the periapsis velocity of the spacecraft, thereby lowering the apoapsis altitude and velocity on each pass through the atmosphere. A larger drag, results in a larger change in velocity (ΔV) for a given orbit pass. The spacecraft passes through the upper atmosphere at hypersonic speeds and as a result is subjected to aerodynamic heating. The aerodynamic heating causes the temperature of both the internal and external spacecraft components to increase during the drag pass. The atmospheric drag and the aerodynamic heating are both functions of the atmospheric density and spacecraft velocity. One of the fundamental trades in performing an aerobraking maneuver is to achieve the largest ΔV possible while keeping all spacecraft components within defined temperature limits. As the spacecraft passes deeper into the atmosphere, the atmospheric density increases which results in a larger drag and a larger ΔV. However, an increase in atmospheric density causes a corresponding increase in the aerodynamic heating and hence, an increase in the spacecraft temperatures. Typically, most science orbiters are designed to minimize structural mass in order to maximize the science payload. The materials used in the construction of these spacecraft have finite temperature limits which cannot be exceeded without loss of structural integrity or functional performance. The temperature limits on these materials introduce a constraint to the aerobraking process and dictate how large the aerodynamic heating can become and thus, how much ΔV can be obtained on a given drag pass.

Aerobraking was first demonstrated by the Magellan spacecraft in orbit around Venus[1]. Mars Global Surveyor (MGS), Mars Odyssey, and Mars Reconnaissance Orbiter (MRO) all successfully performed aerobraking maneuvers around Mars[2,3,4]. The one character all of these missions had in common was that during the drag pass, the maximum temperature limit of the spacecraft was the most limiting factor in determining how many drag passes would be needed to arrive at the final science orbit. In particular, it was the temperature of the solar array of the spacecraft that was the most limiting. Because of orbit-to-orbit variations in atmospheric density and uncertainty in its prediction, the temperature of the solar array for an upcoming drag pass cannot be accurately predicted. In addition to the uncertainties associated with the atmospheric density, uncertainties also exist in the high-fidelity thermal model used to make the temperature predictions. The uncertainties in the thermal model can be classified into three groups; environmental, material property, and modeling. The environmental group encompasses the external inputs to the thermal model which include, heat transfer coefficient distribution, solar heating, etc. The material property group includes the uncertainties in the thermophysical properties of the materials used in the construction of the spacecraft. The modeling group is somewhat abstract and includes uncertainties introduced by modeling a physical object with a nodalized, lumped capacitance representation. This group includes modeling constructs such as contact resistance and mass distribution.

Traditionally, the aerobraking operations phase has required many teams (navigation, atmospheric scientist, mission designers, thermal analyst, etc.) to constantly monitor the mission and spacecraft. The aerobraking operations phase can last between 3 to 6 months and automating this process would reduce workload, cost, and risk of human error. Automation may also increase aerobraking mission flexibility by providing the means with which to choose maneuvers that are not limited to times occurring during a workday[5]. In addition, all aerobraking operations have relied on surrogate variables such as maximum dynamic pressure or maximum incident heat flux for mission control in lieu of the driving constraint which is solar array temperature. The thermal analysis performed on the MRO was unique in that a new thermal analysis technique was developed to account for the uncertainties in the analysis and improve the accuracy of the temperature predictions. The new technique called thermal response surface analysis was demonstrated during aerobraking operations[6]. The thermal response surface analysis technique provides the means with which to use onboard temperature measurements to make maneuver decisions. The purpose of this paper is to describe the thermal model and response surface development as well as the response surface equation integration into an autonomous aerobraking simulation.

II. Thermal Model Development

A high-fidelity thermal model, originally developed in MSC PATRAN®[7] and Thermal Desktop®[8] for MRO aerobraking operations[9], was modified to develop the response surface equations for this autonomous aerobraking simulation. Originally, Thermal Desktop was used to compute the view factors to space and the solar heating. The PATRAN model was used to compute the temperatures during the drag pass, utilizing the view factor and solar heating data from Thermal Desktop, and the aerodynamic heating from the direct simulation Monte Carlo (DSMC) code as boundary conditions. The original high-fidelity PATRAN thermal model was used as a starting point because model was already correlated to flight data.[10] One of the objectives of the autonomous aerobraking study was to consolidate the thermal analysis models into one universal model which would compute the view factors, solar heating inputs and solar array temperatures. To accomplish this objective, the original MRO thermal model, shown in Fig. 1, was converted to Thermal Desktop and correlated to MRO flight data.[11] The results of the correlation effort compare well to flight data. An example of the correlation results is provided in Fig. 2 and Fig. 3 for orbit pass 262.

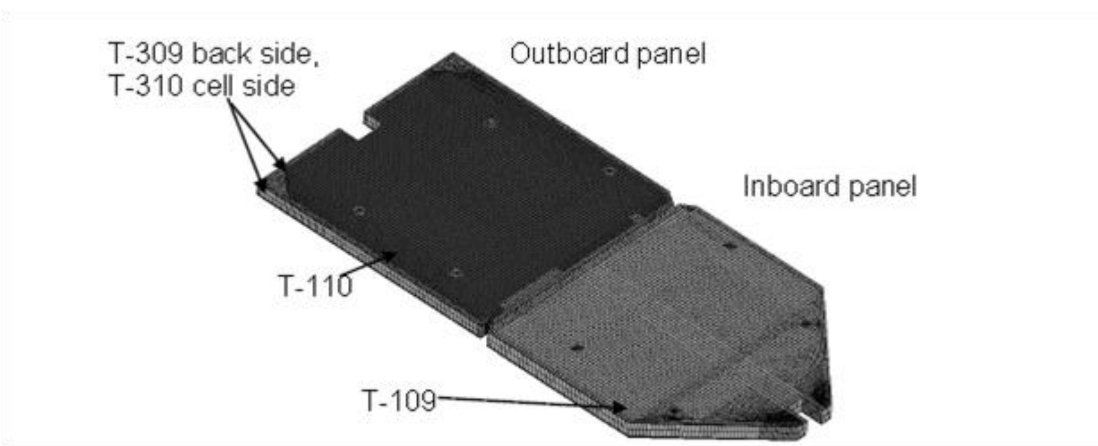

Fig. 1 Original MRO solar array model and sensor locations.

After the MRO model was converted to Thermal Desktop and was correlated to flight data, several modifications were made to utilize the model as a tool for autonomous aerobraking and response surface development. First, the model was parameterized to allow variation in the key environmental, material property, and modeling variables needed for response surface development. This parameterization involved creating symbols within the model that either explicitly define the value of specific variables, or, as in most cases, establishes a multiplier or bias to known values to represent the defined uncertainty of the variable.

The next modification of the model is made to enable autonomous running of multiple analyses in parametric mode with multiple variables, where the user can select a desired number of variables and change the values between a defined upper and lower limit. Currently, Thermal Desktop has no design of experiment (DOE) capabilities; the code only has the built-in ability to run in parametric mode while varying a single variable. For response surface equation development of the MRO model, it is necessary to vary between twelve and fifteen parameters. Therefore, custom logic and operation blocks are added to the Thermal Desktop model that allows for multiple cases being run with variation of a user-defined number of variables. Additionally, these logic blocks allow specification of the total number of cases to run as well as the nominal, the high, and the low values of each variable.

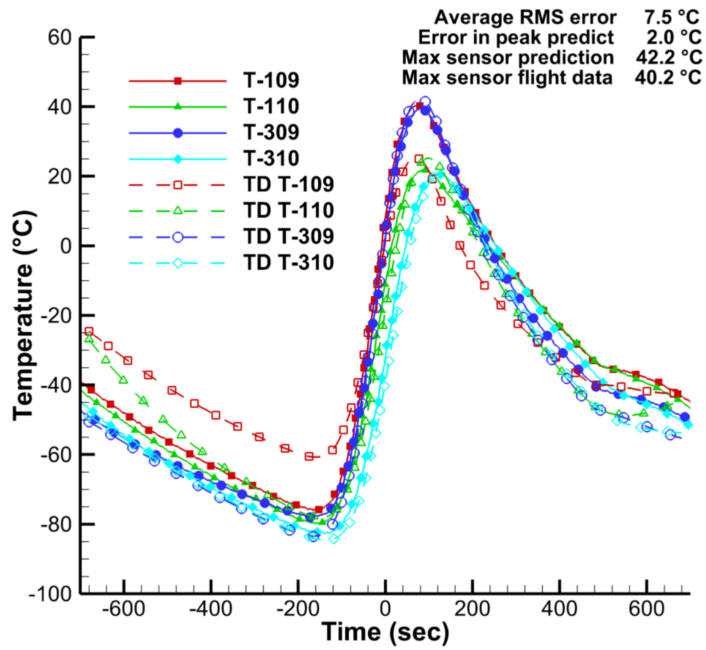

Fig. 2 Correlation of the Thermal Desktop model to flight temperature data for drag pass 262 [11]

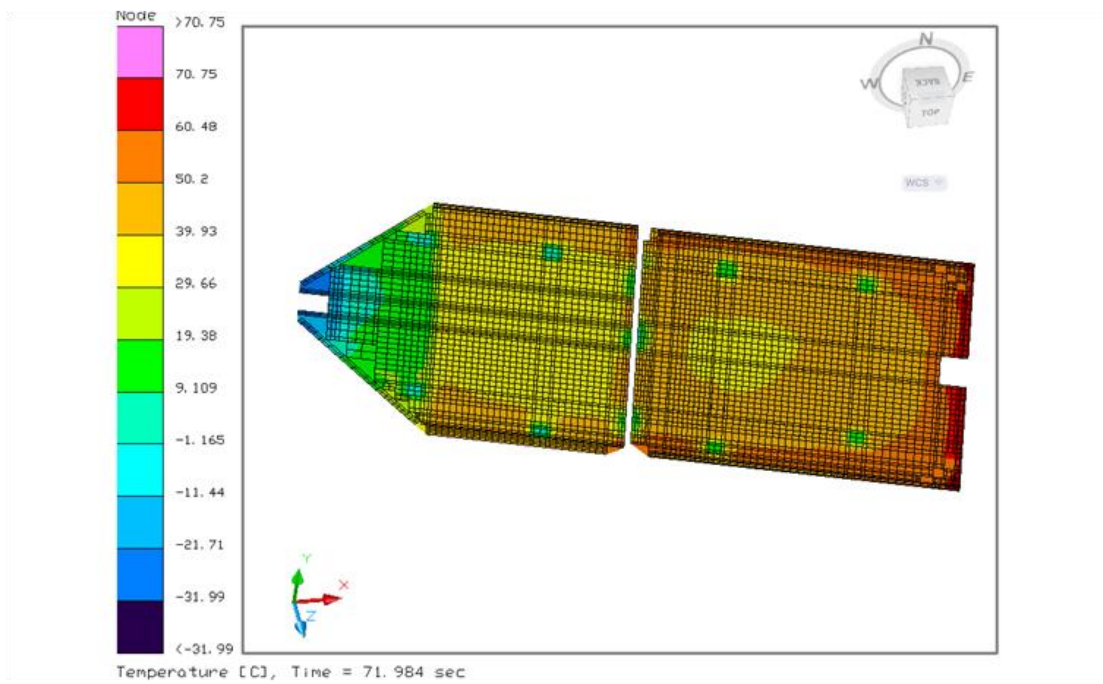

Fig. 3 Peak temperature distribution for drag pass 262 (°C)

The logic block also provides the ability to input a matrix of numbers that define the values of each parameter for each run. For a DOE, this matrix would be N by M elements, where N represents the number of cases in the study, and M represents the number of variables being investigated. The values in the matrix consist of either a 0 or ± 1, where, in the case of the MRO model, 0 indicates that the nominal value of the variable used in the study, and ± 1 indicates that the $\pm 3\ \sigma$ value is used. The variables are coded to range between -1 and +1 so that they are all on the same scale. This matrix is then input to an array data block, within the Thermal Desktop logic manager. While this approach limits the user to only the nominal, high and low values, minimal effort would be required to populate this matrix with any values between -1 and +1, based on either a uniform or Gaussian distribution, and the variable set according to the corresponding value, thus allowing the user to run Monte Carlo analyses, but that aspect is beyond the scope of this study.

III. Design of Experiments, Sensitivity Study and Response Surface Development

For an autonomous aerobraking mission, it is impractical, from a time perspective, with current onboard spacecraft computer technology to run a high-fidelity thermal model onboard the spacecraft. For autonomous aerobraking, the spacecraft must be able to compute the temperatures within seconds, minutes at most. One solution to satisfy this calculation speed requirement is to develop a response surface model for the temperatures which is derived from the high-fidelity thermal model. A response surface model is typically a polynomial equation that can be used to determine how a given response is affected by a set of quantitative independent variables or factors over a specified range. In the case of a high-fidelity thermal model the response is the temperature at a discrete point. The general form of the response surface equation representing the thermal response of the spacecraft solar arrays is given in Eq. (1)[12].

$$T_m = b_0 + \sum_{i=1}^{n} b_i x_i + \sum_{i=1}^{n} b_{ii} x_i^2 + \sum_{i=1}^{n-1} \sum_{j=i+1}^{n} b_{ij} x_i x_j + \sum_{i=1}^{n-2} \sum_{j=i+1}^{n-1} \sum_{k=j+1}^{n} b_{ijk} x_i x_j x_k \quad (1)$$

Eq. (1) captures the main effects, 1st and 2nd order interactions and captures non-linearities with the quadratic terms and 3rd order interaction terms. Main effects are how the response of the system changes as a single factor changes. Interactions occur when the effect of one factor on the response depends on the level of another factor.[13]

Without *a priori* knowledge of how the temperatures calculated via a thermal analysis of a complex system will respond to variations and uncertainty in the input parameters, analysts are forced to include every variable they can think of in the development of a response surface representation of the thermal analysis. One way to generate the data necessary to create a response surface is to perform a DOE. A DOE is a systematic way of varying the design variables so that the data obtained can be analyzed to yield valid and objective conclusions.[13] In the case of the thermal analysis for autonomous aerobraking, the objective is to create a response surface model of the high-fidelity thermal model. As the number of variables or factors, as they are called in statistics, increases, the number of runs required for the DOE and hence, required to define the response surface increases dramatically. For example, in a full factorial design, which is a DOE that includes all possible combinations of the factors, if there are three levels for each factor and ten factors, then the number of required runs of the thermal analysis model would be 59,049, or l^k, where l is the number of levels and k is the number of factors. A level is defined as a discrete value for a particular factor, hence three levels represents three discrete values for a factor. Typically, when three levels are used the minimum, maximum, and midpoint values are used.

There are other types of DOEs that reduce the number of runs, but the trade off is that not every combination of the factors is represented. A face-centered central composite design (CCD) for example is one type of DOE that reduces the number of runs. A face-centered CCD is made up of three parts; center points, axial points, and fractional factorial points. For the same example of ten factors at three levels, if a face-centered CCD is chosen with two center points and a ¼ fractional factorial contribution, the number of runs required of the thermal model would be reduced to 278. The variation in the number of required runs as a function of the number of analysis variables for a full factorial design and a face-centered CCD are compared in Fig. 4.

The trends in Fig. 4 indicate that the number of factors being used to create the response surface should be minimized in order to minimize the number of required runs of the thermal model. In practical terms, if the thermal model takes 2 hours for one run, the 10 factor face-centered CCD requiring 278 runs would take over 23 days running on a single computer to generate the data required to create the response surface. For autonomous aerobraking, updates to the thermal response surface may be required so minimizing the number of required runs, and hence, the time necessary for an update are essential. Additionally, reducing the number of factors reduces the amount of data that needs to be passed back-and-forth and maintained within the autonomous aerobraking simulation software.

To accomplish the goal of minimizing the number of factors, a sensitivity study can be performed to determine which factors initially selected are significant contributors to the solar array temperature response. Creating a screening DOE is a way to examine which of the factors main effects and which interactions are important. A screening DOE is similar to a CCD, except that a screening DOE does not include axial points, may or may not include center points, and the fraction factorial portion is much, much smaller. If a factor is deemed insignificant, it does not mean that particular factor contributes nothing to the response; it just means that particular factors variation is insignificant.

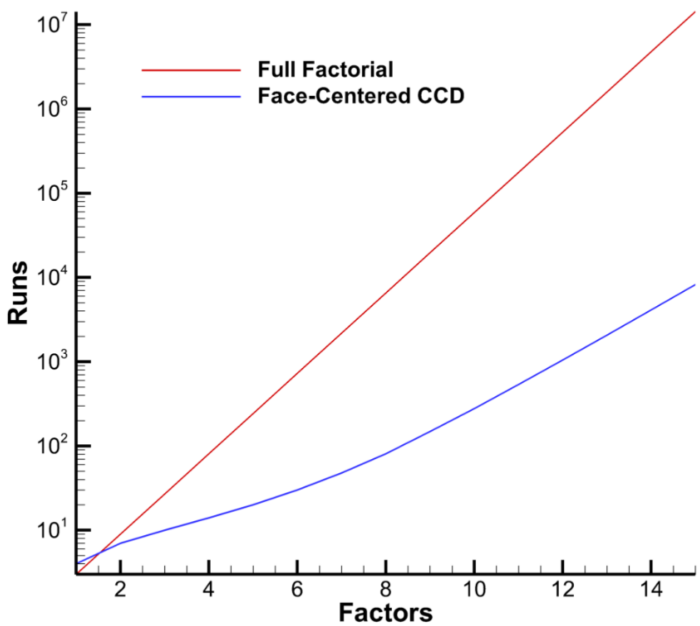

Fig. 4 Comparison of required runs for different DOEs

For this study, the MRO spacecraft is used to simulate autonomous aerobraking around both Mars and Venus. The thermal model described in Section II is used for both the Mars and Venus mission scenarios. The only differences in the model come from the external heating environments. At Mars, the solar heating input is relatively low and the affect of solar occultation on the initial temperatures is large. The atmospheric density and corresponding aerodynamic heating encountered during the drag pass are also relatively low, but due to the low initial temperatures prior to the drag pass, only the aerodynamic heating dominates the thermal response during the drag pass. At Venus, the solar heating inputs are relatively high and the affect of solar occultation in lowering the initial temperatures is lessened. The density and corresponding aerodynamic heating are also relatively high and combined with the solar heating both dominate the thermal response during the drag pass. The differences in the corresponding thermal response for both mission scenarios necessitate that a screening sensitivity study be performed for each mission scenario.

Starting with the initial list of factors used in the actual MRO aerobraking thermal response surface analysis[6], a screening DOE was generated using the JMP® statistical software.[14] The factors and their definitions are given in Table 1. The factors can be classified into three general categories: environmental, material property, and modeling. For these 15 factors, the screening DOE only required 129 runs, 128 from the fraction factorial part and 1 center point.

The JMP software performed an analysis of variance on the resulting temperatures calculated for each case in the DOE matrix. The statistical p-value was an indication as to whether the variation in the factor contributes significantly to the analysis. P-values less than 0.05 typically indicate a significant contribution. For the Mars autonomous aerobraking mission, the main effects for factors that had p-values greater than 0.05 are summarized in Table 2. If the only concern was the main effects, all six of these factors could be eliminated from the subsequent DOE and would not be carried in the response surface equation. However, the interactions between factors must also be examined. In the Mars mission scenario, interactions between all but two of the factors had p-values less than 0.05 when interacting with other factors. The only factors that could be dropped were the drag pass duration, and the solar cell emissivity, hence the face-centered CCD DOE for generating the response surface equation for the Mars mission scenario will contain 13 factors.

Table 1. MRO analysis variables

Category	Factor	Abbreviation
Environmental	Drag pass duration	DP
	Density	RHO
	Heat transfer coefficient	C_H
	Periapsis velocity	V
	Initial solar array temperature	IT
	Orbital heat flux	Q_s
Material Property	M55J graphite emissivity	FSE
	ITJ solar cell emissivity	ITJE
	M55J graphite thermal conductivity	FSk
	M55J graphite specific heat	FSC_p
	Aluminum honeycomb core thermal conductivity	ALk
	Aluminum honeycomb core specific heat	ALC_p
Modeling	Outboard solar panel mass distribution	OFM
	Solar cell layer mass distribution	MD
	Contact resistance	CR

Table 2. Factor screening for Mars mission scenario

Factor	Abbreviation	p-value
Drag pass duration	DP	0.8100
Orbital heat flux	Q_s	0.5987
ITJ solar cell emissivity	ITJE	0.6443
M55J graphite thermal conductivity	FSk	0.7929
Outboard solar panel mass distribution	OFM	0.4642
Contact resistance	CR	0.7929

Since different environmental conditions are encountered for the Venus mission scenario, the screening sensitivity must be performed again. Also, the drag pass duration was replaced by the orbital period. This new factor was used since it was deemed a better representation of the variation in the orbit geometry, which was the original intent of the drag pass duration factor. Following the same procedure as in the Mars mission scenario, an identical screening DOE was generated and the resulting data analyzed. For the Venus autonomous aerobraking mission, the main effects for factors that had p-values greater than 0.05 are summarized in Table 3.

Table 3. Factor screening for Venus mission scenario

Factor	Abbreviation	p-value
Orbital period	P	0.1097
Periapsis velocity	V	0.7999
M55J graphite specific heat	FSC_p	0.5526
M55J graphite thermal conductivity	FSk	0.5232
Aluminum honeycomb core thermal conductivity	ALk	0.9832
Aluminum honeycomb core specific heat	ALC_p	0.5684
Solar cell layer mass distribution	MD	0.5291
Outboard solar panel mass distribution	OFM	0.5496
Contact resistance	CR	0.5081

For Venus, some of the factors that are found to be insignificant are the same as the Mars mission scenario, however, there are others that are insignificant for Venus. but were significant for Mars, and vice-versa. The difference arises due to how different the missions are in terms of their environment and underscores the need to repeat the screening study for every mission scenario. Both scenarios illustrate the need to examine the interaction between factors. It was found that all but two factors had significant interactions with other factors. For Venus, the periapsis velocity and the contact resistance are dropped; hence the face-centered CCD DOE for generating the response surface equation for the Venus mission scenario will also contain 13 factors.

A face-centered CCD with 13 factors was generated using the JMP statistical software. The CCD had 26 axial points, 10 center points and 128 point from the fractional factorial contribution. JMP automatically reduces the fraction used to compute the fractional factorial contribution as the number of factors increases; in this case the fraction was 1/64th. The temperatures calculated for each of the 164 total runs for both Mars and Venus was analyzed using JMP where a least squares fit was constructed using the stepwise regression option in JMP. The result of the regression is a quadratic equation, one unique to the Mars mission scenario and one unique to the Venus mission scenario. The coefficient of determination or R^2 adjusted value was measured and used to determine how well the assumed functional form of the response measures the variability of the supplied data. In this case, the R^2 adjusted value measured how well the quadratic response surface represented the variability in the temperatures generated by the DOE cases. In the Mars mission scenario, the resulting response surface equation had an R^2 adjusted value of 0.9948. For the Venus mission scenario, the R^2 adjusted value is 0.9991. An R^2 adjusted value greater than 0.9 was desirable, but was not sufficient to determine the goodness of fit of the response surface.

To get a clear picture of how well the response surface equation is fitting the response data from the DOE runs, a plot of the actual versus predicted values, a plot of the residual versus predicted values, and the model fit distributions must be examined. The actual versus predicted plot shows the temperatures calculated by the thermal model for the cases described in the DOE plotted against the temperatures calculated by the quadratic response surface equation and is given in Fig. 5 for the Mars mission scenario.

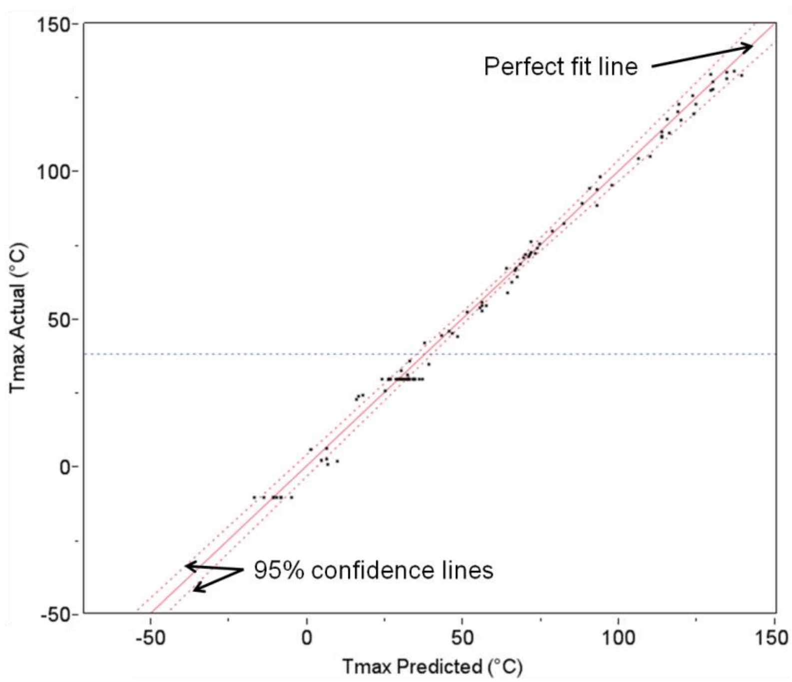

Fig. 5 Mars mission scenario actual temperatures versus predicted temperatures

The centerline of the plot represents a perfect fit of the data; the plot shows that the data points lie close to the center line which indicates a good fit. The residual is the error in the fitted model and is the difference between the actual temperature calculated by the thermal model and the temperature calculated by the response surface equation. The residual for the maximum solar panel temperature versus the predicted maximum temperature is plotted in Fig. 6.

In general, the data points are randomly scattered in Fig. 6 indicating a good fit of the temperature data. However, there are two areas on both Fig. 5 and Fig. 6 where the data points are clustered together; this clustering indicates that one of the factors may be dominating the response. For aerobraking, the peak temperatures are highly influenced by the peak density which is the primary reason for this clustering. One way to alleviate the occurrence of clustering is to break the density up into smaller intervals and develop a different response surface equation for each interval as in Ref 6. For simplicity in implementing the response surface equations into the autonomous aerobraking simulation, a goal is to try to have a single response surface equation. As a result of the goodness of fit analysis, a recommendation is that the density be broken up into three ranges and three separate response surface equations used.

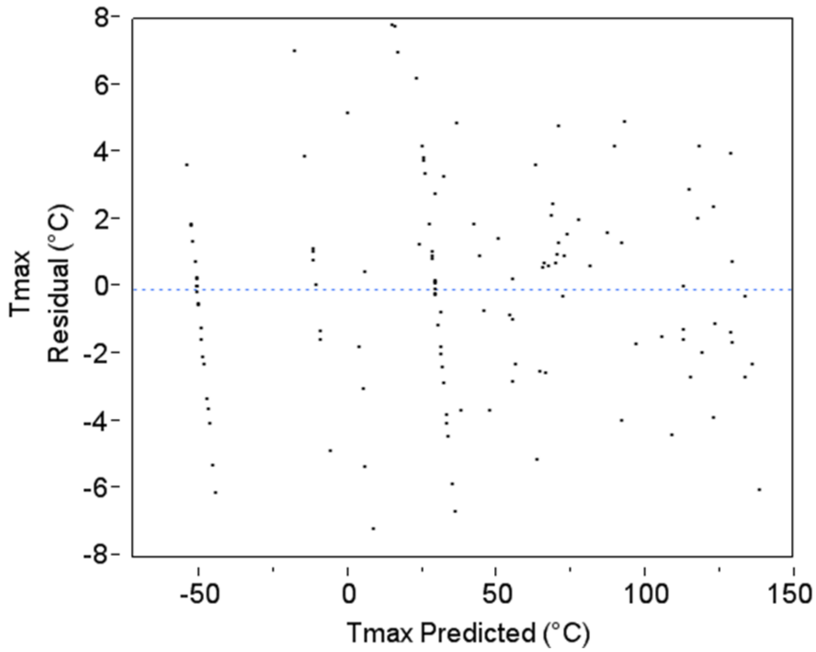

Fig. 6 Mars mission scenario maximum solar panel temperature residual versus predicted maximum temperature

One final check of the goodness of fit is to examine the model fit and model representation error distributions. Both model error distributions should approximate a normal distribution with mean around zero and standard deviation less ≤ 1.0. The model fit error is how well the response surface fits the temperature data in the DOE. The model fit error distribution for the maximum temperature for the Mars mission scenario is plotted in Fig. 7. The distribution is approximately normal and has a mean of 0.0158 and a standard deviation of 1.0359. The standard deviation is slightly above 1.0, but is sufficiently close to 1.0 to conclude that the model is accurate.

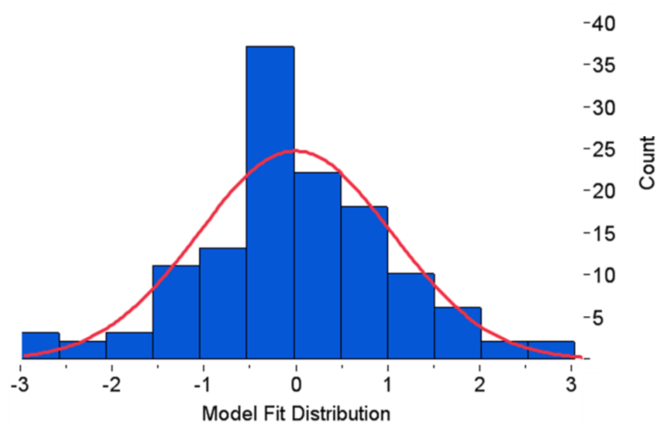

Fig. 7 Mars mission scenario model fit error distribution

The model representation error is how well the response surface fits actual temperatures calculated by the thermal model for points other than those on the DOE. For the Mars mission scenario, the model representation error for the maximum temperature is plotted in Fig. 8. The distribution is approximately normal with a mean of -0.1103 and standard deviation of 0.6177. Hence, it can be concluded that the response surface equation is an accurate representation of the high-fidelity thermal model.

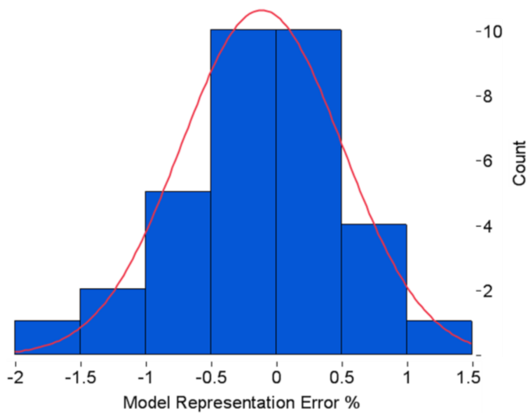

Fig. 8 Mars mission scenario model representation error distribution

The model fit and model representation errors are accounted for in the response surface equation when the temperature calculation is made from within the autonomous aerobraking simulation. Another error is also added as a bias to the temperature calculated by the response surface. This error is present because the high-fidelity thermal model will typically not be correlated to the aerobraking flight temperature data. This error is typically unknown until the first couple of drag passes are made and the flight temperatures and predicted temperatures compared. Therefore, a short calibration period is required but this can be accomplished during walk in which makes up the first initial orbits where the spacecraft periapsis is gradually lowered into the aerobraking altitude corridor.

One important aspect of response surface modeling that must be emphasized is that the response surface equation is only valid over the range for which it was defined. It must be stressed that even a small amount of extrapolation in any factor included in the equation can produce results that are invalid.

IV. Autonomous Aerobraking Simulation

A generic response-surface equation thermal analysis (GRETA) computer program was written for use in the autonomous aerobraking development software (AADS). There are two versions, one written as a standalone program which includes the ability to run Monte Carlo simulations, the other for use with AADS which does not have a Monte Carlo simulation. AADS accesses the GRETA routines via an external function call. This architecture is beneficial in that the response surface equation coefficients or GRETA routines can be updated independently of AADS. The main feature of GRETA is that GRETA will accept any number of variables and hence any number of response surface coefficients so long as the program follows the form of Eq. (1). GRETA will also allow the user to modify any set of factors and thus calculate a new response. Additionally, GRETA allows the user to input a value for the response and calculate the value of one specific factor, holding all others constant. For autonomous aerobraking, the ability to calculate the value of a factor is crucial. For autonomous aerobraking the response is the temperature and the factor which needs to be determined is the atmospheric density. During the autonomous aerobraking simulation a temperature within the temperature corridor is sent by AADS to GRETA and the density is calculated. Hence, the temperature can be used to control the spacecraft during aerobraking. Using the temperature represents a major step forward since the temperature is measured directly onboard the spacecraft and can be used to determine what temperature is input to GRETA for the next orbit pass. The temperature and corresponding density for the Mars Mission run out is shown in Fig. 9 and Fig. 10.

Similar simulations were run using the Venus response surface equation and similar results were obtained, however, since the MRO spacecraft was used, the temperature results were unrealistically high and will not be shown in this paper to avoid confusion. The reason the temperatures were unrealistically high comes from the fact that the solar heating was almost 4.5 times higher at Venus as compared to Mars in addition to a higher aerodynamic heating. The MRO spacecraft was not designed to aerobrake at Venus and hence, the generated thermal response was not consistent with a spacecraft specifically designed for Venus aerobraking. For the autonomous aerobraking simulation at Venus, for demonstration purposes, the maximum temperature obtained from the thermal analysis was scaled to match the maximum temperature calculated for a proposed Venus aerobraking spacecraft; a spacecraft which had a more robust thermal design and had solar panels tailored to minimize the aerodynamic heating.

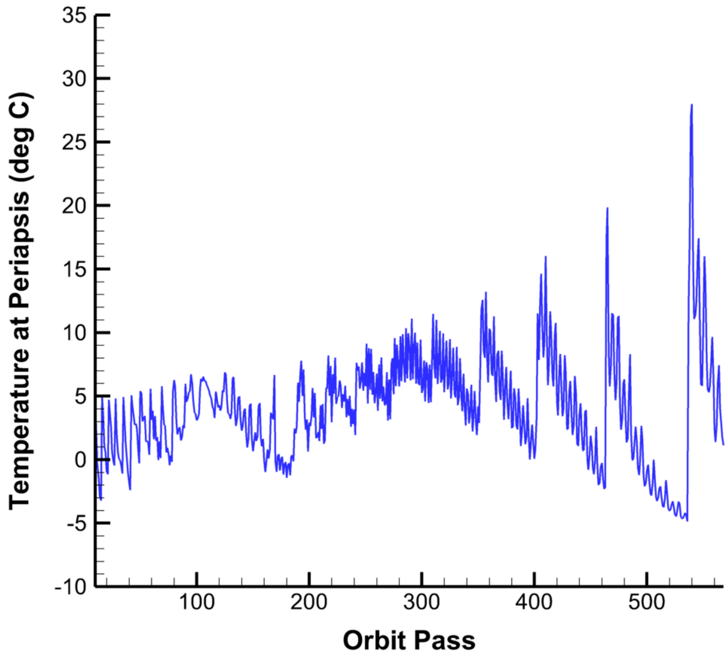

Fig. 9 Periapsis temperature for a Mars mission scenario

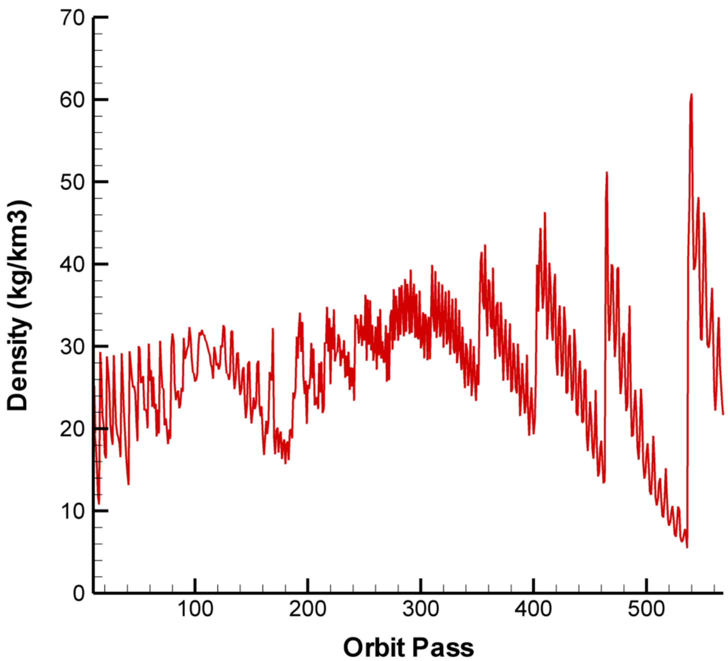

Fig. 10 Periapsis density for a Mars mission scenario

V. Summary

The original high-fidelity thermal model using both PATRAN and Thermal Desktop was described and converted for analysis in Thermal Desktop. The new Thermal Desktop model was successfully correlated to flight data obtained from the MRO mission. The response surface development and the response surface equation integration into an autonomous aerobraking simulation were described and implemented. Analysis variable screening was performed and it was determined that for each mission scenario, two different variables could be dropped from the subsequent response surface equation derivation. A goodness of fit analysis was performed confirming the response surface equations were adequate representations of the high-fidelity thermal model. The generic response surface equation thermal analysis program was developed and demonstrated within the autonomous aerobraking development software.

Acknowledgments

The authors would like to thank Ruth Amundsen for converting the MRO thermal model to Thermal Desktop and making the comparison between the original PATRAN thermal model. This work was sponsored by the NASA Engineering and Safety Center (NESC). This assessment can be found in the final report NESC-RP-09-00605 in November 2011. In particular we would like to thank Steven Rickman the NESC Passive Thermal Control Technical Fellow for his review of this work.

References

[1] Carpenter, A. S., "The Magellan Aerobraking Experiment: Attitude Control Simulation and Preliminary Flight Results", AIAA Paper 93-3830, August 1993.

[2] Lyons, D., Beerer, J., Esposito, P., Johnston, M. D., and Willcockson, W., "Mars Global Surveyor: Aerobraking Mission Overview", *Journal of Spacecraft and Rockets*, Vol. 36, No 3, 1999, pp. 307-313.

[3] Smith, J. C., and Bell, J. L., "2001 Mars Odyssey Aerobraking", *Journal of Spacecraft and Rockets,* Vol. 42, No. 3, 2005, pp. 406-415.

[4] Lyons, D., "Mars Reconnaissance Orbiter: Aerobraking Reference Trajectory", AIAA Paper 2002-4821, August 2002.

[5] Prince, J. L., Dec, J. A., and Tolson, R. H., "Autonomous Aerobraking Using Thermal Response Surface Analysis", *Journal of Spacecraft and Rockets,* Vol. 46, No 2, 2009, pp 292-298.

[6] Dec, J. A., "Probabilistic Thermal Analysis During Mars Reconnaissance Orbiter Aerobraking", AIAA Paper 2007-1214, January 2007.

[7] MSC/PATRAN User Manual, MacNeal-Schwendler Corporation, Version 2010, February 2010.

[8] Thermal Desktop User Manual, Cullimore and Ring Technologies, Inc., Version 5.3, January 2010.

[9] Dec, John A., Gasbarre, Joseph F., and Amundsen, Ruth M., "Thermal Modeling of the Mars Reconnaissance Orbiter's Solar Panel and Instruments During Aerobraking," 07ICES-64, 37th International Conference On Environmental Systems, Chicago, Illinois, July 2007.

[10] Amundsen, Ruth M., Dec, John A., Gasbarre, Joseph F., "Thermal Model Correlation for Mars Reconnaissance Orbiter", 07ICES-17, 37th International Conference on Environmental Systems, Chicago, Illinois, 2007.

[11] Amundsen, Ruth M., "Aeroheating Mapping to Thermal Model for Autonomous Aerobraking Capability", 22nd Annual Thermal Fluids and Analysis Workshop, Newport News, Virginia, 2011.

[12] Breyfogle, F. W., *Implementing Six Sigma: Smarter Solutions Using Statistical Methods*, 2nd Ed., John Wiley & Sons, Inc., Hoboken, NJ, 2003.

[13] *NIST/SEMATECH e-Handbook of Statistical Methods*, http://www.itl.nist.gov/div898/handbook/, 2011.

[14] JMP, Version 8. SAS Institute Inc., Cary, NC, 1989-2011.